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Space Propulsion Hazards Analysis Manual (SPHAM)

Volume 2: Appendices

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Space Systems Company
P. O. Box 179
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Space Division, Air Force Systems Command
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California 93523-5000

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19. ABSTRACT (Continue on reverse if necessary and identify by block number)

The Space Propulsion Hazards Analysis Manual (SPHAM) is a compilation of methods and data directed at hazards analysis and safety for space propulsion and associated vehicles, but broadly applicable to other environments and systems. It includes methods for compiling imposed requirements and deriving design requirements. It describes in detail the steps to constructing accident scenarios for formal risk assessment. It discusses the approaches to developing probabilities for events in scenarios, and probabilities for scenarios. It illustrates data analysis from experience data for the purpose of probability modeling. The SPHAM provides methods for predicting blast, fragmentation, thermal, acoustic and toxicity post-accident environments. The SPHAM describes in overview fashion a large number of qualitative and quantitative analytical methods available to perform hazards analysis complete with guidelines for application. Examples are FMEA, Fault-tree and Energy Analysis. It describes methods to organize analysis by type, phase, or subsystem. Examples are interface hazards analysis, preliminary hazards analysis, and ordnance hazards analysis. Qualitative and quantitative risk assessments are described. The formal processes for hazards analysis and safety for various...

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22a. NAME OF RESPONSIBLE INDIVIDUAL John W. Marshall	22b. TELEPHONE (Include Area Code) (805) 275-5642 / 22c. OFFICE SYMBOL RKPL

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Cont'd

Block 19 (continued): agencies and departments of the government and DOD are described. The appendices to SPHAM contain voluminous data on available references in the form of an annotated bibliography, summary of the hazardous nature of 27 commodities common to space propulsion, and system description for a variety of space launch vehicles, upper stage vehicles, and spacecraft. (AW)

Accession For		
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DTIC TAB	☐	
Unannounced	☐	
Justification		
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Availability Codes		
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PREFACE

This Space Propulsion Hazards Analysis Manual was prepared by Martin Marietta Astronautics Group, Denver, Colorado, under contract F04611-84-C-0003. It was sponsored jointly by the Air Force and the National Aeronautics and Space Administration. Program management and contract administration were provided by the Air Force Astronautics Laboratory, Edwards Air Force Base, California. Technical direction was provided principally by both the Astronautics Laboratory and the Eastern Space and Missile Center, Patrick Air Force Base, Florida. The project manager was Mr. John W. Marshall, Air Force Astronautics Laboratory.

This manual is intended to be a source of information, methods and data useful to hazards analysis for space propulsion and space vehicles. It is not intended to be used as a regulatory document, nor is it to be construed as a complete, definitive and authoritative work.

The complete Space Propulsion Hazards Analysis Manual (SPHAM) consists of these two volumes that are bound separately to facilitate their handling and use as reference material:

Volume I SPHAM Technical Chapters
Volume II SPHAM Appendices

ACKNOWLEDGEMENTS

This manual reflects the work of many dedicated individuals from industry and government. We wish to give special recognition to those people whose contributions helped to make this manual a first of its kind:

Mr. Louis J. Ullian, ESMC, Patrick AFB, FL
Mr. John W. Marshall, AFAL, Edwards AFB, CA
Mr. Bill Riehl (retired), NASA-MSFC, Huntsville, AL
Mr. John Atkins, RTI, Cocoa Beach, FL
Mr. Robert Fletcher, NASA-JSC, Houston, TX
Mr. Tom Kerr (retired), NASA-HQS, Washington, DC
Mr. Wayne R. Frazier, NASA-HQS, Washington, DC
Mr. Bobby R. Quisenberry, GDC, San Diego, CA

We also wish to acknowledge the following people for their contributions as Martin Marietta employees:

Mr. Bob Lomax	Mr. Rich Barthelow
Mr. Doug Banning	Mr. Ed Kirk
Ms Gloria Bradway	Mr. Art Major
Ms Kate McCarthy	Mr. Joe Mangino

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Appendix A
Annotated Bibliography

APPENDIX A

ANNOTATED BIBLIOGRAPHY

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ACKNOWLEDGEMENTS

We would like to thank the following for their contribution of document reference lists and/or information (documents, memoranda or notes) from their personal files: Mr. Tom Kerr from NASA Headquarters, Mr. Robert Brown and Mr. Bob Fletcher from Johnson Space Center, NASA; Mr. Wilbur Riehl from Marshall Space Flight Center, NASA; Capt. Betschart from Space Division, USAF; Mr. William Riley from WSMC Vandenberg Air Force Base; Mr. Louis Ullian from ESMC Patrick Air Force Base; Mr. John Marshall from AFAL (formerly AFRPL) Edwards Air Force Base; Dr. Walter Queen and his colleagues from the DOD Explosive Safety Board; and Dr. Jerry Ward and Dr. Michael Swisdak, Jr., from the Naval Surface Weapons Center (NSWC).

APPENDIX A
FOREWORD

This annotated bibliography documents are hazard analysis and failure mode data obtained from an extensive literature search and field survey of the rocket propulsion and aerospace community. The acquired data provides information on the potential hazard environment and the credible failure scenarios associated with space vehicle liquid and solid propellant systems and subsystems. Each bibliography entry has a reference number, rank, reference identification and reference application.

The reference number is the number assigned to the document for listing in the bibliography.

Rank is a number from a system that ranks each document from 1 to 10, where 10 is the material most relevant to the SPHAM program.

The reference identification is intended to provide all information necessary to obtain the document; e.g., title, author, organization, document number, date and AD number, if available. The reference identification is most complete for technical reports, journal articles, seminar minutes and conference proceedings. It is less complete for personal memoranda, viewgraph presentations and, in some cases, material abstracted from documents without a reference.

The reference application includes:

1) a brief description of the document;
2) list of the hazardous materials that are discussed;
3) description of the failure scenarios;
4) type of post-accident environment that is discussed; e.g., blast, fire, fragmentation, toxicity or acoustics;
5) methodology used for failure mode analysis (quantitative or qualitative);
6) the location (or person) from which the reference document was obtained, and
7) the initials of the person that reviewed the document.

A cross-reference matrix provides a ready tool to identify the documents that contain information on specific vehicles, hazardous materials and post-accident environments. Each document is identified by its reference number that corresponds to its entry in the Annotated Bibliography. The rank or value rating of the document is also listed. Note that this matrix summarizes vehicles, and marks with an "X" information on hazardous materials, and post-accident environments. Failure scenarios and methodologies are obtained from the annotated bibliography entries themselves.

APPENDIX A

SYMBOLS AND ABBREVIATIONS

The following symbols and abbreviations are used in the annotated bibliography and the cross-reference matrix.

ANNOTATED BIBLIOGRAPHY

Ref No. = Reference Number
NA = Not Applicable
NI = Not Identified
*/O = Internal Audit Symbols
AFAL = Air Force Astronautics Lab
AFRPL = Air Force Rocket Propulsion Laboratory
A-50 = Aerozine 50
DOD = Department of Defense
DSP = Defense Systems Program
DSCS = Defense Satellite Communications Systems
DTIC = Defense Technical Information Center
FMEA = Failure Modes and Effect Analysis
GIDEP = Government Industry Data Exchange Package
GPS = Global Positioning System
ICBM = Intercontinental Ballistics Missile
ISPM = International Solar Polar Mission
IUS = Inertial Upper Stage
JANNAF = Joint Army Navy, NASA, Air Force
JSC = Johnson Space Center
KSC = Kennedy Space Center
LH_2 = Liquid Hydrogen
LO_2 = Liquid Oxygen
MMH = Monomethyl Hydrazine
MMU = Manned Maneuvering Unit
N_2O_4 = Nitrogen Tetroxide
N_2H_4 = Hydrazine
NASA = National Aeronautics and Space Administration
OTV = Orbit Transfer Vehicle
PAM-D = Payload Assist Module-Delta Type
RTG = Radioisotrope Thermoelectric Generator
SPHAM = Space Propulsion Hazard Analysis Manual
SRB = Solid Rocket Booster
SRM = Solid Rocket Motor
STS = Space Transportation System
TDRS = Tracking and Data Relay Satellite
TOS = Transfer Orbit Stage
TRS = Teleoperator Retrieval System
USAF = United States Air Force Base
VAFB = Vandenberg Air Force Base
WSMC = Western Space and Missile Center

CROSS-REFERENCE MATRIX

X = Item(s) included in Document
RK = Rank
DOC = Document
He = Helium
N_2 = Nitrogen
LHe = Liquid Helium
LH_2 = Liquid Hydrogen
LOX = Liquid Oxygen
S1.1 = Solid 1.1 (7)
S1.3 = Solid 1.3 (2)
N_2H_4 = Hydrazine
UDMH = Unsymmetrical Dimethyl Hydrazine
MMH = Monomethyl Hydrazine
N_2O_4 = Nitrogen Tetroxide
NH_3 = Ammonia
CO_2 = Carbon Dioxide
CO = Carbon Monoxide
H_2O_2 = Hydrogen Peroxide
LF_2 = Liquid Fluorine
LCH_4 = Liquid Methane
NF_3 = Nitrogen Trifluoride
Hg = Mercury
RTG = Radioisotope Thermoelectric Generator
A50 = Aerozine 50
TNT EQ = TNT Equivalent
BLST = Blast
FRGMNT = Fragment
TOXIC = Toxicity
ACOUST = Acoustics

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
001	7	Titan II Hazard Management Guide, for Fuels, Propellants and Chemical, 00-ALCP 144-4 HQ ODGEN Air Logistics Center, Hill AFB, UT, 1 July 1983 Dept. of the Air Force Ogden Logistic Center	Propellant Mishap and Corrective Action	Liquid UDMH/Hydrazine N_2O_4	Fuel or Oxidizer Spill/ Leak Topside or Holding Trailer	Thermal Toxicity Blast	Qualitative Quantitative	In-House	GB *
002	6	MIL-STD-1574A (USAF) 15 August 1979	System Safety Program for Space and Missile Systems	NA	NA	NA	Qualatative	In-House	GB
003	6	Suppressive Shields Structural Design and Analysis Handbook, HNDM-1110-1-2, U.S. Army Corps of Engineers, Huntsville Division, Nov. 1977	Explosive Environments Inside Containment Structures and Structural Response to the Contained Blast (more for the use of design engineering)	TNT Chemical Explosives Converted to TNT Equivalence	NI	Blast Thermal Fragmentation	Quantitative	DTIC	GB *
004	5	Draft Report Simulation of Area Weapons Effects Safety Criteria, U.S. Army Armament Research & Development Command Safety Office, Nov. 1981. S. Hoxha J. Elliott	Defines Quantified Safety Criteria for Realistic Simulation of Incoming Projectlies (Artillery/ Mortar)	Artillery Mortar	NA	Blast Fragmentation Acoustics	Qualitative Quantitative	El Segundo Operations, CA	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
005	8	The Characterization and Evaluation of Accidential Explosions, NASA CR 134779 R. A. Strehlow R. D. Siewert 1975 NASA-Lewis Cleveland, OH, 44135	Provides a review of Current Charaterization and Evaulation of Accidential Explosions	TNT Liquid Propellants	Explosions due to Pressure Vessel Failure Spill/ Leak of Material	Blast Fragmentation Thermal	Quantitative Quantitative	NTIC	GB *
006	9	Workbook for Predicting Pressure Wave and Fragment Effects of Exploding Propellant Tanks and Gas Storage Vessels, N76-19296, Sept 1977 W. E. Baker Southwest Research Institute San Antonio, TX	Provides Method to Predict Damage and Hazard from Explosions of Propellants and Burst of Pressure Vessels.	NI	Confinement by Missile Confinement by Ground Surface High Velocity Impact	Blast Fragmentation	Qualitative	NTIS	GB *
007	9	Explosion Effects and Properties Part 1 - Explosion Effects in Air, NSWC/WOL/TR75-116, Oct 1975 M. M. Swisdak, Jr.	Basically Provides Methodologies for Blast Effects (Confined, Underground and Air)	Solid 94/6 Ammonium Nitrate TNT Equivalence	Overpressure Air Blast Static Pressure	Blast	Qualitative	DTIC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
008	10	Liquid Propellant Explosive Hazards Final Report No. 1, AFRPL-TR-68-92, December 1968 (Project Pyro) A. Willoghby, C. Wilton, J. Mansfield	Provides Description and Results of Actual Testing Performed to Determine (Predict) Explosive Yield.	Liquid Oxygen-RP-1 Liquid Oxygen-Liquid hydrogen Nitrogen Tetroxide-50% UDMH and 50% Hydrazine	High Velocity Impact Explosive Donor Confinement by Missile 100 ft Drop Cold Propellants	Blast Thermal Fragmentation	Other	DTIC	GB *
009	10	Liquid Propellant Explosive Hazards, Final Report Vol. 2 AFRPL-TR-68-92, Dec. 1968 (Project Pyro) A. B. Willoughby and C. Wilton J. Mansfield	Present the Results for Blast, Thermal and Fragmentation	Liquid Oxygen-RP-1 Liquid Oxygen-Liquid Hydrogen $N_2O_4$50% UDMH/50% Hydrazine	High Velocity Impact Explosive Donor Confinement by Missile Confine-	Blast Thermal Fragmentation	None	DTIC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
009	Cont				ment by Ground Surface (Vertical & Horizontal)				
010	10	Liquid Propellant Explosive Hazards, Final Report Vol 3 AFRPL-TR-68-92 Dec. 1968 (Project Pyro) A. Willoghby, C. Wilton, et. al	Present a Generic Methodology for Predicting the Blast and Thermal Environments Resulting from Explosions.	Liquid Oxygen-RP-1 Liquid Oxygen-Liquid Hydrogen $N_2O_4$50% UDMH/50% Hydrazine	High Velocity Impact Exposive Donor Confinement by Missile Confinement by Ground Surface (Vertical & Horizontal)	Blast Thermal Fragmentation	Quantitative Qualitative	DTIC	GB *
011	8	Estimating Airblast Characteristics for Single Point Explosions	Provides Methods for Determining Blast Wave Characteristics	Nuclear TNT Equalivance	NI	Blast Acoustics	Quantitative	ANSI	GB

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
011	Cont	in Air, with a Guide to Evaluation of Atomspheric Propagation and Effects, ANSI 52.20, 1983	for Single Point or Spherical Explosions						
012	1	Scenario Analyses of Industrial Accidents, Sixth International System Safety Conference, Houston Texas, September 1983	Not Applicable Deals with Personnel Accidents	NA	NA	NA	NA	NA	GB
013	2	How to use System Safety Technics during the Crisis Time, System Safety Conference, September 1983	(This Report was Abstract only) Might Provide Methodology for Assessing the Accident	NA	NA	NA	NA	NA	GB
014	7	Minutes of Vehicle Level Working Group Meeting held at Los Alamos National Labatory March 31, 1982 Space Shuttle, IUS & Galileo R. H. Brown	This Minutes Talk about Tests that will be taking place and Assessment of Overpressure tests that have been completed.	Radio isotope Thermo-electric Generators (RTG) Solid Rocket Booster	Fall Over Pitch Over Pressure Rupture of SRB Cases	Blast Fragmentation	NI	JSC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
014	Cont			Liquid Oxygen-Liquid Hydrogen					
015	10	Final Integrated Accident Risk Assessment Report for T34D/IUS, MCR-78-22, Part I, August 1982	Evaluation of the accident risk being assumed when the T34D/IUS are subjected to final assembly, checkout, preparation for launch, and launch from ETR.	Solid Rocket Motors 50/50 UDMH & Hydrazine & N_2O_4	Electrical Ground and In-Flight Malfunction	Thermal Toxicity Blast	Quantitative Qualitative Other	In-House	GB *
016	9	Final Integrated Accident Risk Assessment Report for T34D/IUS, MCR-78-22, Part II, August 1982 T. W. Knapp	Comprehensive Evaluation of the Accident Risk to be Assumed when the DSCS II & DSCS III are Integrated to the T34D/IUS at launch pad.	Solid Rocket Motors	Premature SRM ignition, RCS Tank Rupture and Leakage	Blast Thermal Toxicity	Qualitative	In-house	GB *
017	2	Radiofrequency Hazard Analysis of the Thiokol Model 2134A Safe & Arm Device using an Initiator no more Sensitive than the NASA	This Report Presents the results of a Worst Case RF Hazard Analysis of the Thiokol Model 2124A S&A Device. (Might	NA	Induced Current to initiate firing	NA	Quantitative Other	Franklin Research Center, PA	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
017	Cont	Standard Iniciator, F-C5696-3, June 1983 R. H. Thompson	be good for an Engineer who is trying to Select a Safe and Arm Device)						
018	2	Radiofrequency Testing of MX Exploding Bridgewire (EBW)	Initators of the Exploding Bridgewire Type were Tested, for RF Susceptibility (Good for Design Engineer)	NA	Induced Current Resulting in EBW Firing	NA	Other	Franklin Research Center, PA	GB
019	2	Direct Current and Radiofrequency Sensitivity Testing of two Electroexplosive Device (EED) F-A5321 J. Stewart R. Thompson	Testing of Electro-Explosive Device	NA	Induced Current Resulting in EED Firing	NA	Other	Franklin Research Center, PA	GB *
020	2	Limited RF Testing of Thiokol Supplied Electroexplosive Devices. F-C5293 J. Stuart R. Thompson	Testing of RF Sensitivity of EED's.	NA	NA	NA	Other	Franklin Research Center, PA	GB *
021	8	Product Safety Management and Engineering, Prentice-Hall International Series in	Provides Methodologies in Accident Prevention	NA	NA	Blast Thermal Fragmentation Toxicity	Quantitative Qualitative	Borrowed from Dick Olson SD.	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
021	Cont	Industrial and System Engineering, 1980 (PAM-D) Skynet-4 W. Hammer, PE							
022	8	Handbook of System and Product Safety, Prentice Hall, 1972 W. Hammer, PE	Guide for System and Product Safety	NA	Premature Separation of Payload or SRM & RCS Ignition	NA	Qualitative	Dick Olson SD.	GB O
023	8	Skynet 4, Accident Risk Assessment Report, British Aerospace, Space and Communications Division, RPT/UKS/ 45689/BAE	Assessment of the accident risk involved in all operational phases of the Skynet 4 cargo element.	Solid and Liquid Propellants	Premature Deployment and RCS or SRM Ignition	Blast Thermal Toxicity	Qualitative	Borrowed From Art Major	GB *
024	8	Appendix A7 to Accident Risk Assessment Report, Space Program P80-1, April 1982 Rockwell International F04701-77-6-0065 W. H. Reetz	Analysis of the accident potential of the P80-1 payload including ION propulsion.	Liquid GH_2 Hydrazine SRM Methane Liquid Neon Mercury	Spills/ Leaks Imploding Tank Domes	Blast Toxicity Thermal	NA	Borrowed from Art Major	GB *
025	4	Storable and Cryogenic Propellant Safety for Space Tugs, AD HOC Tug	Evaluation of the relative safery aspects of Earth Stor-	Liquid LH_2 LO_2	Spills/ leaks Ingnition	Thermal Blast Toxicity	Qualitative	Sent by Joe Mangino	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
025	Cont	Safety Group, National Aeronautics and Space Administration, Wash., D.C. Sept 1974	ables and cryogenic propellants as they affect space Tug operations and interfaces with the space shuttle, personnel and ecology.	MMH N_2O_4 UDMH N_2H_4 Aerozine JO	of propellants Imploding tank domes Impact				
026	1	Environmental Protection, Safety, and Health Protection Program for DOE Operations, DOE 5480.1A, 8-81	Basically has reference source for explosive, nuclear and fire standards and some procedures.	NA	NA	NA	NI	Sent by Joe Mangino	GB *
027	10	Large Solid-Propellent Boosters Explosive Hazards Study Program (Project Sophy) AFRPL-TR-65-211, AD476617 O. R. Irwin et. al	1) Enable prediction of hazards associated with T&H and launching of SRM 2) Determine critical diameter of typical class II propellants 3) Determine parameters for partial or complete denotation	Class II Solid Composite Propellants	NA	Blast	Quantitative	DTIC	GB *
028	7	Air-Launched Missile Motor Behavior, ADB061411, Vol 1 AFRPL-TR-81-45	Assess the risk in some hazard scenarios of the tactical environment (Tactical	1.3/B and 1.1/A Solid Propellants	Explosion	Blast Fire Fragmentation	Other	DTIC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
028	Cont	(Limited) Hercules, Inc. J. Richard B. Leining & J. H. Thacher, et. al	Rocket Motors)						
029	7	Air Launched Missile Motor Behavior Handbook Vol. II, AFRPL-TR-81-45, ADB061412 (Limited) R. B. Leining, J. H. Thacher Hercules, Inc. Aerospace Div. Bacchus Works, Magna UT, 84044	This Handbook provides a rational methodology for assessing explosive risk over the operational life cycle of an air launched tactical rocket motor (hazards, methods and scenarios)	1.1 and 1.3 Solid Propellants	Rapid Heating Crash Landing	Fire Explosion Blast-Overpressure	Qualitative Quantitative	DTIC	GB *
030	7	Air-Launched Missile Motor Behavior, AFRPL-TR-78-54, Oct 1978(Limited) R. B. Leining & J. H. Thacher, Hercules, Inc.	Presents Hazard Analysis Methodolgy	NA	Ignition of Fuel Spill	Blast Thermal Toxicity	Qualitative	DTIC	GB *
031	6	The Air Force Manual for Design and Analysis of Hardened Structures, AFWL-TR-74-102, ADB004152 (Limited)	The objectives of this manual are to present methods and procedures for the estimation of nuclear weapon effects on pro	Nuclear Weapons	Surface Burst Air Burst Underground	Thermal Fragmentation	Quantitative	DTIC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
031	Cont	R. E. Crawford, et. al	tective structures; and design/analysis for such structures		Burst				
032	3	Material Compatibility With Space Storable Propellants, Design Handbook N72-26678 P. E. Uney D. A. Fester	Material compatibility with propellants	Liquid Hydrazine Monomethyl-Hydrazine Nitrogen Tetroxide Flourine Oxygen	Corrosion	NI	Qualitative	In House	GB *
033	10	Liquid Propellant Rocket Abort Fire Model, SC-RR-70-454 Model with Atlas/Centaur Model with Pyro Data B. E. Bader et. al	Analytical development of a thermal flux model of a launch abort for a liquid fuel rocket	Liquid Propellants	Explosive Donor	Thermal	Quantitative Qualitative	NTIS	GB *
034	6	Evaluation of Explosives Storage Criteria, Computer Program Users Manual, AD870963 J. D. Donahue	Computer program for evaluating quantity distance criteria for blast damage	Explosives	Explosion	Blast Fragmentation	Quantitative	DTIC	GB *
035	10	Assembly and Analysis of Fragmentation Data for Liquid Propellant Vessels, NASA CR-134538,	Fragmentation effects of bursting liquid propellant data. (This book used the	In-Flight Impact	Confinement by missile. Confine-	Fragmentation	Quantitative Qualitative	NASA	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
035	Cont	AD-870-963 Missile Project Pyro C. D. Miller et. al	same steps the SPHAM is using; gathering and analyzing). Recommend this document be used for final SPHAM.		ment by ground surface.				
036	8	Bomb Crater Damage to Runways, AFWL-TR-72-183, AD907456L P. S. Westine	Presents empirical methods for estimating true crater size and extent of concrete cracking from bomb explosions.	Explosive Charges Solid TNT	Explosion	Fragmentation	Quantitative	DTIC	GB *
037	8	Target Response to Explosive Blast, AD715475, Sept. 1970 G. H. Custard	Response of vehicle and plate glass to explosion.	TNT	Explosion initiated	Blast Fragmentation	Other	DTIC	GB *
038	8	Evaluation of Explosive Storage Safety Criteria, ADB71-194 G. Custard, J. Donahue, et. al	Provides a basis for estimating damage to structures from air-blast forces	TNT	Explosion	Blast	Quantitative	DTIC	GB *
039	9	Development of STS/ Centaur Failure Probabilities Liftoff to Centaur Separation, Technical Report No.	Presents results of an analysis performed to determine STS/ Centaur catastrophic failure probabilities	Liquid MMH N_2O_4 HE LO_2	ET Puncture & Explosion in Payload	Blast Thermal	Quantitative	Sent by John Marshall	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
039	Cont	82-1404	from liftoff to Centaur seperation from Orbiter. (Galileo)	LH_2					
040	7	STS, SRM Propellant Detonation Probability Thiokol, Inc. Wasatch Division, 7030/RL-83-25 D. M. Benn	Rational on whether or not a SRM will detonate at a velocity of 800 ft/sec at sea	Solid Class 1.3 propellant	Impact	Explosion	Quantitative	Hugh Baker	GB *
041	7	Research on the Hazards Associated with the Production and handling of Liquid Hydrogen, 1961 Bureau of Mines 5707 M. G. Zabetakis, et. al	Identified hazards associated with liquid hydrogen handling	Solid LH_2	Spill	Blast Thermal	Qualitative Quantitative	In House	GB *
042	7	Hydrogen Safety Manual, Advisory Panel on Experimental Fluids and Gases Lewis Research Center, NASA TMX-52454 Lewis Research Center	Presents safety practices and standards for the handling of hydrogen (Design, Operational, and accidents)	Liquid hydrogen	Leakage/ spills	Blast Thermal Fragments	Quantitative Qualitative	Joe Mangino	GB *
043	9	Cargo Element Integrated accident Risk Assement Report for the Space Test, Program P80-1, Martin Marietta, El Segundo/STS	Assesses the risk associated with the P80-1/STS from launch to separation	Solid/ Liquid SRM Hydrazine GN_2 Mercury LH_2	Premature Firing & Collision with Orbiter	NI	Qualitative	Art Major	GB *

NA Not Applicable
NI Not Identified
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043	Cont	MMC-F04701-77-C-0183 R. W. Dueck		LO_2					
044	10	System Safety Engineering and Management, Wiley Series on Systems Engineering and Analysis System Safety Engineer, Rockwell	Presents methodology for the identification and control of hazards throughout the life cycle of a project.	NA	NA	NA	Quantitative Qualitative	Dick Olson	GB O
045	8	Preliminary Integrated Accident Risk Assesment Report for Navstar Block II Satellite and PAM-D Upper Stage Cargo Element, Rockwell International, SSD81-004-1-1	Evaluation of the risks assumed during the assembly checkout test and operation of the NAVSTAR block II Satellite including the PAM-D. Identifies	Solid & Liquid SRM Monopropel- and hydrazine	Valve Malfunction on Ground & in Flight	Blast Thermal Acoustics Toxicity	Quantitative	In-House	GB *
045	Cont		potential hazards occuring in the Orbiter Payload Bay until the cargo element is deployed						
046	7	Thermodynamic Properties of Nitrogen Gas from Sound Velocity Measurement 1979 E176-0675 R-1187 and GIDEP How Safe is Hydrogen? R-1178, Jan., 1979	Provides methodology for predicting the fireball and thermal radiation parameters for a hydrogen explosion	Hydrogen	Combustion in closed vessel	Fireball Thermal Radiation	Quantitative Qualitative	Received from Ken Morrison	GB

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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047	10	GIDEP-High Explosive Equivalency Tests of Large Solid Propellant Motors, NWC-TP-4643, Sept., 1968	Evaluate the explosive potential of large class 2 and class 7 solid propellants	Solid Class 2 & 7 Propellants C-4 Booster	Explosion Initiated	Blast Peak Over-Pressure Postive Phase-Impulse Fragments	Quantitative Other	Received from Ken Morrison	GB *
048	7	Explosive Hazard Classification Data, D290-10015-1 Boeing, May, 1982 C. W. Hurd	Hazard classification to determine proper handling and storage for explosive items in the IUS	Solid Propellants Hydrazine	NA	NA	Qualitative	Received from Ken Morrison	GB *
049	7	PAM-D Payload Assist Module-Delta Document Flight, McDonnell Douglas, Jan., 1984 MDC E9967B	Description of the flight safety aspects of the PAM-D operations	Solid Propellants Hydrazine	Premature Motor Ignition & Electrical Malfunction	NI	Qualitative	Received from Ken Morrison	GB *
050	8	Titan 23C History and Failure Analysis, Background for RTG Hazardous Environments - Prelaunch Through Stage 2 Shutdown, MCR-72-260, Oct, 1972 Titan IIIC	To determine the Titan IIIC launch vechile failure modes and probability of occurrence that will produce hazardous environment for the RTG	Radioisotope thermal electric generators (RTG)	Impact Velocity 100 ft drop Tip over at pad Rupture of Tanks	Fire Overpressure Fragmentation	Quantitative	Borrowed from Tom Kerr	GB

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
051	10	Multi-Hundred Watt Radioisotope Thermal Electric Generator (MHW-RTG) Analysis Report for the MJS-77 Mission, General Electric, Doc. No. 77SDS4206, Jan., 1977 Mariner-Jupiter Saturn Spacecraft No Author	Presents the analyses and results used for the evaluation of nuclear safety of MHW-RTG. This document is great for identifying failure scenarios and post accident environments from the vehicle at launch pad to reentry	Nuclear Isotopic fuel Liquid Monopropellant Hydrozine System Solid SRM	Tank Overpressure & Tank Collapse in Flight	Explosion Fireball	Quantitative Qualitative Other	Tom Kerr	GB *
052	9	MJS '77 RTG Safety Study Phase II Range Safety Equipment, Launch Pad Hazards, Launch Vehicle Failure Probabilities,	Presents launch pad hazards, launch vehicle failure probabilities and reentry environment data.	Nuclear RTG Solid SRM	Tank Collapse & Overpressure in Ground	Thermal Fireball Overpressure Fragmentation	Quantitative Other	Tom Kerr	GB
052	Cont	and Reentry Environment, Report No. CASD/LVP 76-004 Titan IIIE/ Centaur E. K. Kaukoner	(Great referen)	Liquid LO_2/LH_2	Operations				
053	7	MJS '77 RTG Safety Study Launch Vehicle Hardware, Launch Complex an Trajectory Data Phase I, Report No. CASD/LVP 75-058 Mariner/Jupiter	Contains description of Titan IIIE/Centaur D-IT launch vehicle system and trajectory data	Nuclear RTG Solid Class II Composite-Binder	NA	NA	NA	Tom Kerr	GB *

NA Not Applicable
NI Not Identified
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053	Cont	Saturn (MJS) General Dynamics Convair Division		SRM Liquid 50/50 Hydrozine & UDMH-N_2O_4 LH_2/LO_2 Hydrogen Peroxide					
054	10	DOD/NASA System Input Analysis (Study 2) Space Shuttle Explosive Equivalency Study Final Report, Volume I, Executive Summary, ATR-74 (7337)-1 Aerospace Corp. Grand Systems & Environ. Dept	Summary-Evaluates the explosive equivalency of the propellants used in Shuttle System and types of credible failure modes were identified	Liquid LO_2/LH_2 Solid Propellant (PBAN)	Liquid Mixing & Explode within Confines of ET & Spill w/	Blast Thermal	Quantitative Qualitative	Borrowed from Joe Mangino	*
055	10	DOD/NASA System Impact Analysis (Study 2.1) Space Shuttle Explosive Equivalency Study Final Report, Volume II, Technical Discussion, ATR-74(7337)-1 Aerospace Corp. Grand System & Environ. Dept	Evaluate the explosive hazards and recommends TNT equivalency criterion applicable to static on pad and early flight operations	Liquid LO_2/LH_2 Solid Propellant (PBAN)	SRM for Ground Operation and In-Flight	Blast Thermal	Quantitative	Borrowed from Joe Mangino	*

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
056	10	DOD/NASA System Impact Analysis (Study 2.1) Space Shuttle Explosive Equivalency Study Final Report, Volume III, Appendices, ATR-74 (7337)-1 Aerospace Corp. Grand System & Environ. Dept	Describes in detail the data and analyses generated during the study	Liquid LO_2/LH_2 Solid Propellant (PBAN)	Same as 054 & 055	Blast Thermal	Quantitative Qualitative	Borrowed from Joe Mangino	*
057	9	Final Report" TNT Equivalency Study for Space Shuttle (EOS) Vol. I, Management Summary Report, ATR-71 (7233)-4 Aerospace Grand System & Environ. Dept	Concise review of TNT equivalency for LO_2/LH_2 at launch pad (static). Summarizes principle conclusion and recommendations.	Liquid LO_2/LH_2	Refer to Literature Search Tank Rupture	Blast Thermal	Quantitative Qualitative	Borrowed from Joe Mangino	GB *
058	9	Final Report" TNT Equivalency Study for Space Shuttle (EOS) Vol. II, Technical Discussion Report, ATR-71 (7233)-4 System Planning Division	Discussion of the available test data and analyses	Liquid LO_2/LH_2	Tank Rupture	Blast Thermal	Quantitative Qualitative	Borrowed from Joe Mangino	GB *
059	9	Final Report" TNT Equivalency Study for Space Shuttle (EOS) Vol. III, Appendices Report, ATR-71(7233)-4 System Planning Division	Contains supporting analyses and back-up material	Liquid LO_2/LH_2	Tank Rupture	Blast Thermal	Quantitative Qualitative	Borrowed from Joe Mangino	GB *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
060	7	Prediction of Pressure Vessel Failure Modes Proposed Test Plan and Cost Estimate, TOR-0059 (6816-13)-3, June, 1971 J. C. Spears	To verify the criteria which predicts the pressure vessel failure mode.	NA	Leakage Burst	NA	Quantitative	Joe Mangino	GB *
061	5	Fracture Mechanics and Pressure Vessel Design, TOR-0059(6816-33)-1, April 1971 N. N. Au	Presents fracture mechanics as a safety design (failure mode)	NA	Flaw induced failure	NA	Quantitative	Joe Mangino	GB *
062	6	Fail-Safe Operation of Aerospace Pressure Vessels, TOR-0059(6816-33)-2, May 1971	Theories to predict pressure failure mode as either leak or burst	NA	Burst	NA	Quantitative	Joe Mangino	GB *
063	8	Weapon System Safety Guidelines Handbook, System Manager's Guide To System Safety Part I, NAVORD-OD-44942	Provides an understanding of System Safety	NA	NA	NA	Qualitative	Joe Mangino	GB *
064	9	Weapon System Safety Guidelines Handbook, System Safety Management Guidelines, Part II, NAVORD-OD-44942	Methodology for doing Safety functions	NA	NA	NA	Qualitative	Joe Mangino	GB *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
065	9	Weapon System Safety Guidelines Handbook, Hazard Control for Explosive Ordnance Production, Part IV, NAVORD-OD-44942	System Safety techniques for the life cycle of explosive ordnance.	NA	NA	NA	Qualitative	Joe Mangino	GB *
066	8	Hazards Analysis of SRM Assembly Operations for KSC-VAB, TRW-11389-1, Thiokol/Wasatch	Identifies failure scenarios that could cause ignition of the SRM.	Solid Propellants	High Velocity Impact	Thermal	Qualitative Quantitative	Joe Mangino	GB *
067	6	USAF Propellant Handbook Vol. I, Hydrazine Fuels, AFRPL-TR-69-149, Mar 1970	Compilation of the physical and chemical properties, T&H, Safety, and the thermal and catalyic decomposition of the hydrazine family of fuels.	Liquid Ammonia Hydrazine Monomethyl-hydrazine 50/50, UDMH/N_2H_4	NI	Blast Toxicity Thermal	NI -3	In-House	GB
068	5	USAF Propellant Handbook Nitric Acid/Nitrogen Tetraoxide Oxidizers, Vol. II, AFRPL-TR-76-76	Provides physical and chemical properties, handling transportation, safety and material compatibility of oxidizers	Liquid Nitric acid Nitric Oxide Nitrogen Tetroxide Nitrogen Dioxide	NI	Blast Toxicity Thermal	Quantitative Qualitative	In-House	GB -3

NA Not Applicable
NI Not Identified
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069	5	NASA/DOD Space Transportation System Payload *Ground Safety Handbook*, KHB1700.7	Ground processing requirements useable *at all STS and cargo* procesing locations.	NA	NA	NA	Qualitative	In-House	GB
070	5	Safety Policy and Requirements for Payloads Using the Space Transportation System, NHB 1700.7A	Establishes the policy and safety requirements applicable to all STS payloads and their GSE.	NA	NA	NA	Qualitative	In-house	GB
071	8	Shuttle/Centaur Phase II Safety Review Data Package, GDC-SSC-83-009, Dec., 1983	Airborne Hazard Reports (Hazard, Cause and Controls)	Cryogenics N_2H_4 LOX/LH_2	Premature Venting & Separation of Upper Stage Leakage of Cryo. Propellant	Blast Thermal Toxicity	Qualitative	Ken [illegible]ison	GB *
072	3	Project Pyro Dynamic Pressure Accuracy Evaluation, June 1969 AFRPL-TR-68-111 C. M. Richey	Accuracy Study was performed on the pressure measurement instrumentation of project pyro.	NA	NA	Blast	NA	Rogers/ Ullian	GB *
073	8	Destruct Tests on Scale Model Saturn I Booster, NASA TMX-53007, Feb., 1964 J. Gayle, & C. Glakewood	Investigation to determine the explosive yields resulting from destruct of the S-I Stage with internal and external destruct systems.	Liquid Lox/RP-1	Internal & External destruct systems	Blast Thermal Fragments	Quantitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
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074	8	Study of Liquid Propellant Blast Hazards, June 1965, AFRPL-TR-65-144 A. B. Willoughby, C. Wilton, T. Coodale, J. Mansfield	This document summerizes the work conducted for Project Pyro up to date, over all design of program and analyze of test data. (120 in. solid motor impact test)	Liquid N_2O_4/50% UDMH - 50% N_2H_4 LO_2/RP-1 LO_2/LH_2	High velocity impact Explosive donor CBM Command Destruct "Cold" Propellant	Blast Thermal	Quantitative Qualitative	Rogers/ Ullian	GB *
075	8	The Detonation Behavior of Ammonium Perchlorate as a Function of Charge Density and Diameter, June 1967, AD-818804 (Limited) A. R. Clairmont	Study of composite propellant models which is of shock sensitivity, detonability and detenation behavior as a function of physical conditions and chemical composition	Solid Ammonium Perchlorate	NI	Blast	Quantitative	DTIC	GB *
076	10	Failure modes and efects analysis, 34D/ transtage, Dec., 1983, MCR-84-002 W. Willard	Probability of occurance for failure scenarios pertaining to the T34D/Transtage	Solid Propellants Liquid Propellants	Ground & In-Flight	NA	Quantitative	In House	GB *

NA Not Applicable
NI Not Identified
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			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
077	10	Failure modes and effects analysis, 34D/IUS, July 1983, MCR-84-075 W. Willard	Probability of occurrance for failure scenarios pertaining to the T34D/IUS	Solid Propellants Liquid Propellants	Ground & In-Flight	NA	Quantitative	In House	GB *
078	10	Prediction of Explosive Yield and Other Characteristics of Liquid Rocket Propellant Explosions, Final Report June 1973, University of Florida NAS10-1255 E. A. Farber, J. Smith, E. Watts	Presents methods by which explosive hazards of liquid propellant can be assessed and explosive yield predicted.	LH_2/LO_2 LO_2/RP-1 LH_2/LF_2	NI	Blast Thermal	Quantitative	Rogers/ Ullian	GB *
079	9	Prediction of Explosive Yield and Other Characteristics of Liquid Propellant Rocket Explosions, Final Report, Oct., 1968 University of Florida NAS 10-1255 E. A. Farber et. al	Presents work done by Dr. Farber's work in arriving at crediable explosive yield for liquid propellant	LOX/RP-1 LOX/LH_2 LF_2	Bulkhead Failure Toppling on the pad Crash landing	Blast Thermal	Quantitative Qualitative	Rogers/ Ullian	GB *
080	9	Summary Report on a Study of the Blast Effects of a Saturn Vehicle, Feb., 1962, ADL Control # 101,907	Describes the results of an experimental and analytical program to predict blast and thermal effects	Liquid RP-1/LO_2 LH_2/LO_2 RP-1/LO_2/ LH_2	Tipping of Tanks (spilled propellants)	Blast Thermal Fragmentation Acoustics	Quantitative Qualitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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080	Cont	Arthur D. Little, Inc.	of a Saturn Vehicle should it fail at launch. Also include fragments and acoustics						
081	7	Evaluation of the Blast Parameters and Fireball Characteristics of Liquid Oxygen/Liquid Hydrogen Propellants, Report #0954-01(01) FP/April 1967, Downey Aerojet-General Corp.	Objective was to study the basic parameters of LOX/LH_2 propellant and compare its blast characteristics with LOX/RP-1 and N_2O_4/A-50.	Liquid LOX/LH_2 LOX/RP-1 N_2O_2/A-50 LH_2	NA	Blast Thermal Fragmentation	Quantitative	Rogers/ Ullian	GB *
082	10	Research on Hazard Classification of New Liquid Rocket Propellants Vol I, Oct., 1961, AF/SSD-TR-61-40 Rocketdyne	Evaluation of hazards associated with the storage and handling of nitrogen tetroxide chlorine triflouride, hydrazine, and pentaborane to establish safety procedures and safe distance.	Liquid Nitrogen Tetroxide Chlorine Triflouride Hydrazine Pentaborane	Propellant Spills	Blast Thermal Toxicity	Other Quantitative Qualitative	Rogers/ Ullian	GB *
083	9	Research on Hazard Classification of New Liquid Rocket Propellants Vol II, Oct., 1961, AF/SSD-TR-61-40 Rocketdyne	Presents results of Titan propellant mishaps and determines hazards.	Liquid Nitrogen Tetroxide/ 50-50 UDMH & Hydrazine	Propellant Spills	Blast Thermal Toxicity	Qualitative Quantitative	Rogers/ Ullian	GB *

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NI Not Identified
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			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
084	10	Thermal Radiation from Saturn Fireballs, Vol. 1, Dec., 1965, TRW 2122-6001-T0000 L. J. Van Nice	Model for predicting the thermal radiation and dynamics of a fireball produced by liquid propellants	Liquid Propellant	NA	Thermal	Qualitative Quantitative	Rogers/ Ullian	GB *
085	10	Thermal Radiation from Saturn Fireballs, Vol. II, Dec., 1965, TRW 2122-6001-T0000 L. J. Van Nice & H. J. Carpenter	Provides results for the Saturn V and Altas/Centaur 5 Thermal Calculations. Includes tempature, duration, liftoff time, Density, rise velocity and heat flux. This reference should be one of the key documents for thermal evaluation.	NI	Explosion	Thermal (Fireball,	Quantitative Heal Flux)	Rogers/ Ullian	GB *
086	8	Determination of Wall Responses to Blast Effects from Explosive Charges Distributed in a Cubicle Type Structure R. Rindner, P. Catinny Arsenal, Dover, NJ Minutes of the 4th Explosive Safety Seminar on High-Energy Solid Propellant, 1962	Deals with the determination of the magnitude blast loads in terms of pressure and impulse on a protective wall as a result of detonation of an explosive charge located close to the wall. (Don't know if methodology is valid).	Explosive Charge	NA	Blast	Quantitative	Explosive Safety Board	GB *

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NI Not Identified
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087	10	Accoustical Effects of Large Motors, L. C. Walther Aerojet-General Corp. Sacramento, CA Minutes of 4th Explosive Safety Seminar on High-Energy Solid Propellant, 1962	This report provides method for predicting acoustical power in watts for solid and liquid propellant motors	TNT Equivalency	NI	Acoustics	Quantitative	Explosive Safety Board	GB *
088	7	Detonation Initiated By High Pressure Gas Loading of a Solid Explosive, Donna Price & F. J. Petrone Minutes from the 5th Explosive Safety Seminar 1963	Provides test results and numerical calculations to determine wheather an accidental detonation of an ordance item can induce detonation in adjacent ordance.	Solid Explosive	Explosive Donor	Blast	Quantitative	Explosive Safety Board	GB /
089	5	Fragmentation Study on Large Solid Rocket Motors, Polaris Model A-3, L. C. Walther Aerojet Corp. Minutes from the 5th Explosive Safety Seminar 1963	The objective of the program was to obtain quantitative data on the size of inert and propellant fragments and thier projected distance from explosively destroyed motors. (Suggest we order documents for results).	Class 2 Solid Propellant	Explosion	Fragments	Qualitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
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			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
090	5	Quantity-Distance Criteria for Liquid Propellants, R. C. Herman Minutes from the 5th Explosive Safety Seminar 1963	Developing quantity distance criteria for liquid propellants.	NI	NI	NI	Qualitative	Explosive Safety Board	GB *
091	9	A Criterion for Predicting Impact Initiation of Explosive Systems, Hyla S. Napadensky & J. E. Kennedy, ITT Research Institute, Chicago, IL, Minutes from the 6th Explosive Safety Seminar, 1964	Method for predicting the critical impact velocity which causes a violent explosion (Critical impact velocity as a function ot explosive length and diameter)	Solid Propellants	Impact Velocity	Blast	Quantitative Qualitative	Explosive Safety Board	GB *
092	8	Titan III Solid Motor Impact Test, F. H. Weals NOTS China Lake, CA, Minutes from the 6th Explosive Safety Seminar 1964	Describes an impact test of a large solid propellant motor simulating characteristics of stage zero of Titan III.	Solid Propellant 92,000 lbs	Fallback	Blast	Quantitative	Explosive Safety Board	GB *
093	9	Detonation Hazards of Large Solid Rocket Motors, B. E. Giesler et. al, USAF, AFRPL Edwards, CA, Minutes from the 7th Explosive	This paper presents the results of the critical diameter experiements and theoretical model consistent with these	RDX-Adulterated propellant Solid Ammonium-	Explosion	Blast	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
093	Cont	Safety Seminar on High-Energy Propellants, 1965	results for determining the detonation characteristics of class II propellants	perchlorate, aluminum, and polyurethane (PBAN)					
094	9	High Explosive Yield Tests of Solid Propellants, Polaris Motors H. M. Richey, NOTS China Lakes, CA, Minutes from the 7th Explosive Safety Seminar on High Energy Propellants, 1965	Blast tests were con ducted to evaluate the explosive hazard characteristics of solid propellant rocket motors (explosive yield, overpressure and fragments)	Solid Propellants Class 7 double-base Class 2- Polyurethane Composite		Blast-(Overpressure Shock waves) Fragmentation	Quantitative (Test Results)	Explosive Safety Board	GB *
095	7	Detonation Characteristics of Large Solid Rocket Motors, V. H. Valor et. al, USAF Air Force Rocket Propulsion Lab., Edwards, CA, Minutes of the 8th Explosive Safety Seminar 1966	Refers to project Sophy - provide experimental results and theoretical model for predicting diameter of a typical class II propellant.	Solid Ammonium Perchlorate (AP) Aluminum PBAN	Detonation	Blast (overpressure)	Quantitative	Explosive Safety Board	GB *
096	5	Blast Attenuation Studies In Dividing Wall Protective Construction, B. R. Sullivan &	Summary of blast attenuation studies, results are used in material selection	NI	Explosion	Blast	Quantitative Qualitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
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096	Cont	A. A. Bombich, OCE Waterways, Exp. Station, Vicksburg, MS, Minutes of the 8th Explosive Safety Seminar, 1966	which reduce peak pressure transmitted from an explosion.						
097	8	Fault Tree Analysis as applied to Missile Systems, R. F. Sellars MMC, Orlando, FL Minutes of the 8th Explosive Safety Seminar 1966	Analysis to (1) Predict the cause of potential hazards or (2) determine the cause of an accident. Fault Tree concept.	NA	NA	NA	Qualitative	Explosive Safety Board	GB *
098	7	TNT Equivalency - Gas Dynamics Comparison for Moderately Pressurized Tanks, R. A. Bourdreaux North Am. Aviation, Inc. Downey, CA AD827743 Minutes of the 9th Explosive Safety Seminar 1967	Comparison between results of gas dynamic and TNT equivalency energy for large, thin walled tanks at moderate pressures.	NI	Detonation	Blast (Shock front)	Quantitative	Explosive Safety Board	GB *
099	6	Ballistic Investigations of Frangible Protective Structures for Space Vehicles, Atlas Missile Saturn IV, D. J. Dunn & A. S. Schlenler, USA	The scope is limited to the ballistic aspects of frangible panels such as fragmentation and shock effects on panels	NA	NA	Fragmentation	Quantitative Qualitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
099	Cont	Ballistic Research Labs AD827752 Minutes of the 9th Explosive Safety Seminar 1967	including drag and distance. Good reference for material selection for attenuation of explosion.						
100	10	Status Report on Hazard Evaluation of Large Solid Rocket Motors, C. Dale, et. al. NOS Indian Head, MD Minutes of the 9th Explosive Safety Seminar 1967	Hazard Analysis - (a) possible incident or malfunction and their contributing stimuli (b) effects of the resulting stimuli on the motor (c) damage to the surroundings resulting from possible reactions	Solid Composite Propellant (solid motors)	NA	Fragmentation	Quantitative Qualitative	Explosive Safety Board	GB *
101	1	Review of the Hydrazine/Oxygen Reaction Kinetics S.B. Dalgaard, M.O. Sanford April 1981 Paper Presented at the International Corrosion Forum	Derivation of the kinetics, rate of reaction, and temperature dependance of the reaction between Hydrazine and Oxygen. Based on empirical data.	Liquid N_2H_4, 02	NA	NA	Qualitative, Quantitative	Doug Banning	DWB 0

NA Not Applicable

NI Not Identified

Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
102	8	In-Silo Launch Abort Environment for LGM25C Missile. E.W. Holtzscheiter, D. Kelleher, D. Tate May 1976 AFWL-TR-75-177	Examines the environment inside a Titan II Silo during and immediately after a large scale hypergolic propellant explosion. determines time-temperature profiles dynamic over pressure (maximum), and time-over pressure profiles. Uses TNT equivalencies to perform analysis.	Liquid Hypergolic Propellants Not Specified (A-50, N204)	Confinement in Closed Silo-Spill Leak of Material	Blast, Thermal and Overpressure	Quantitative	Doug Banning	DWB 0
103	8	Liquid Propellant Rocket Abort Fire Model B. Bader, A. Donaldson, et al, Sandia Laboratories Journal of Astronautics and Aeronautics, December 1971	Model for reaction time, fireball growth rate, critical fireball size, and fireball heat flux (thermal effects) resulting from cryogenic rocket fuel spills of various sizes.	Liquid Cryogenic Rocket Fuels	Launch Abort	Thermal	Quantitative	Doug Banning	DWB *
104	7	Gas Phase Reactions of Hydrazine with NO2, NO, and O2.	Using empirical data the kinetics, rates of reaction, and	Liquid N2H4, NO2, NO, O2	NA	NA	Quantitative Qualitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
104	Cont	R.F. Sawyer, I. Glassman	kinetics of the gas phase reactions of hydrazine with NO2, NO, and O2 are determined.						
105	8	Combustion Characteristics of Condensed-Phase Hydrazine-Type Fuels with Nitrogen Tetroxide. T. Christos, Y. Miron, et al. J. Spacecraft Vol. 4, No. 9, P. 1224	Deterimines empirically the TNT equivalency, shock-wave overpressures, and time-temperature profiles for the reactions of N2H4, MMH, UDMH, and A-50 with N2O4 and air.	Liquid N2H4, MMH, UDMH, A50, N2O4, Air	NA	Thermal, Shock Wave Overpressures	Quantitative	Doug Banning	DWB O
106	2	"The Characterization and Evaluation of Accidential Explosions" R.A. Strehlow, et al. June 1975, NASA CR-134779	This report covers explosion characteristics-wave properties, TNT equivalence, scaling laws, non-ideal behavior, source property effects and atmospheric and ground effects. Also discussed are damage mechanisms and several specific examples of acciden-	Nuclear, TNT, RDX etc., propellants (liquid)	In buildings, ductile vessels, internal explosions vapor cloud	Fragmentation Blast Wave, Overpressures	Qualitative Quantitative	Doug Banning	DWB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
106	Cont		tal explosions. Good discussion on liquid propellant explosions.						
107	2	Thermometric Determination of Oxidant-Fuel Distribution within a Rocket Combustor M.C. Burrows, Lewis Research Center NASA TN D-5626 Jan 1970	Empirical determination of Oxidizer and Fuel distributions within a single element combustion chamber using Thermometric mapping.	Liquid N2H4, N204	NA	Thermal	Empirical, Quantitative	Doug Banning	DWB 0
108	9	Pad-Abort Thermal Flux Model for Liquid Rocket Propellants F. Kite, B. Bader, Sandia Laboratory SC-RR-66-577, Nov. 1966	Determines a model to predict time vs. temperature profiles radiant heat flux, and fireball liftoff time for a pad abort for the Saturn V Launch Vehicle.	Liquid Propellants (Not Specified)	Launch Pad Abort	Thermal	Qualitative	Doug Banning	DWB 0
109	10	Launch Hazards Assessment Program, Report on Atlas/Centaur Abort F. Kite, D. Webb, B. Bader Sandia Laboratory, October 1965	Complete data collection and analysis of the aborted launch of the 1965 Surveyor - Mission Atlas - Centaur. Data includes temp-	Liquid RP-1, LO2, LH2	Launch Abort of Atlas/ Centaur	Overpressure, Thermal, Fragmentation	Quantitative Empirical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
109	Cont	SC-RR-65-333	erature, pressure, particle veloc- ities of all as a function of time and distance from launch pad.						
110	6-7	The Autoxidation of Hydrazine and Alkyl Substituted Hydrazine Vapors J. Bellerby, May 1980 Perme Memo 92	Presents the kinetics and reaction rates for the autoxidation of N2H4, MMH, and UDMH in air. (Useful for determining length of time required for a Hydrazine(s) leak to degrade in the vapor phase.)	Liquid N2H4, MMH, UDMH	NA	Vaporous Hydrazines in Air	Empirical, Quantitative	Doug Banning	DWB 0
111	3	Mixing and Reaction Studies of Hydrazine and Nitrogen Tetroxide using Photographic and Spectral Techniques M.C. Burrows, June 1968 NASA TN D-4467	Examines combustion characteristics of impinging streams of N2H4, N204 in a rocket engine nozzle. Reports temperature profiles UV radiation, total radiation, and photographic spectra of the resultant reaction(s).	Liquid N2H4, N204	NA	Rocket Engine Nozzle	Empirical, Qualitative Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
112	5	Literature Survey (1958-1968) of the N_2H_4, Alkylated Hydrazine - N_2O_4 Bipropellant System. P. Breisacher, July 1969 SAMSO-TR-69-272, TR-0066(5210-10)-2	Discusses the basic kinetics and reaction rates for the decomposition of N_2H_4 and the reactions of N_2H_4, MMH, and UDMH with N_2O_4.	Liquid N_2O_4, N_2H_4, MMH, UDMH, NH_3	NA	NA	Literature Search and Compilation	Doug Banning	DWB 0
113	5	Atmospheric Chemistry of Hydrazine: Gas Phase Kinetics and Mechanistic Studies. J. Pitts, E. Tuazon, et al. August 1980 ESL-TR-80-39 (AFESC)	This study was under taken to experimentally determine the atmospheric fate of N_2H_4, MMH, and UDMH. Reactions with Hydroxyl Racial (OH-) Ozone, and NOx in the atmosphere in the presence of UV light were studied. (Applicable to air pollution problems.)	Liquid N_2H_4, MMH, UDMH, O_3	Spills and other Sources Evaporated into the Atmosphere	Atmosphere	Empirical & Theoretical, Qualitative, Quantitative	Doug Banning	DWB 0
114	5	Atmospheric Reaction Mechanisms of Amine Fuels E. Tuazon, et al. March 1982 ESL-TR-82-17 (AFESC)	Determines the kinetics, rate constants, reaction products, and temperature dependance of the reactions of N_2H_4, MMH, and UDMH with chemicals and conditions present in the atmosphere.	Liquid N_2H_4, MMH, UDMH, A-50, O_3	Non-Explosive Release of Amine Fuels to the Atmosphere	Atmosphere	Empirical, Qualitative, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
115	5	The Hard Start Phenomena in Hypergolic Engines, Vol. IV. The Chemistry of Hydrazine Fuels and Nitrogen Tetroxide Propellant Systems. Pittsburgh Mining and Safety Research Center. March 1974. NASA CR-140360	Kinetics, reaction rates, infrared spectrum, and reaction products for several reactions before, during, and after fuel ignition are discussed here. (For N2H4, MMH, UDMH and N2O4 as the major reactants. Several trace compounds are also included in reactions.)	Liquid HNO3, N2H4, MMH, UDMH, N2O4	NA	NA	Empirical, Qualitative, Quantitative	Doug Banning	DWB 0
116	7	The Hard Start Phenomena in Hypergolic Engines, Vol. II. Combustion Characteristics of Propellants and Propellant Combinations. Pittsburgh Mining and Safety Research Center, March 1974. NASA CR-134314	Combustion characteristics, TNT equivalency, and DSC thermal traces of N2O4, N2H4, UDMH, and combinations of above. Also IR analysis of residue from N2O4/N2H4 reaction.	Liquid N2O4, N2H4 UDMH	NA	Explosive, Thermal	Empirical, Quantitative	Doug Banning	DWB 0 DWB 0
117	3	Mixing and Reaction of Hydrazine and Nitrogen	Examines the automization and re-	Liquid N2O4, N2H4	NA	Rocket Engine Combustion	Empirical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
117	Cont	Tetroxide at Elevated Pressures M.C. Burrows, NASA Lewis Research Center. AIAA Journal, Vol. 5, No. (?) September 1967	action characteristics of impinging streams of high pressure N2H4/N204.			Chamber			
118	3	Theoretical Effect of Non-Equilibrium Combustion of N2H4/N204 Propellant Performance R.F. Sawyer, Jan. 1967 J. Spacecraft Vol. 5, No. 1	Comparison of Equilibrium and Non-Equilibrium Combustion of N2H4/N204.	Liquid N2H4, N204	NA	NA	Theoretical, Quantitative	Doug Banning	DWB 0
119	2	Thermodynamic and Thermophysical Properties of Combustion Products, Vol. II V. Alemasov, A. Dregalin et al., 1972 Translated from Russian properties of several propellant combinations.	Presents chemical composition, enthalpy, density, rate constants of reaction, transport properties, and other thermodynamic	Liquid 02, H2 UDMH, NH3, Kerosene	NA	NA	Theoretical, Empirical, Quantitative	Doug Banning	DWB 0
120	6	The Hard Start Phenomena in Hypergolic Engines Vol. III - Physical and Combustion Characteris-	Physical and combustion characteristics of several potentially explosive	Liquid Hydrazine Nitrate N2H4, H20,	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
120	Cont	tics of Engine Residues March 1974 NASA CR-140361	engine residues individually and in combination with their source fuels.	Hydrazine Dintrate, N2O4, MMH Nitrate, MMH Dinitrate, Tetramethyl-tetrazene					
121	2	Evidence for t[illegible] Fo[illegible]ation of Az[illegible] in th[illegible]4/N2O4 Re[illegible]tion. A. [illegible], G. Koehler, et al. AIAA Journal Vol. 6, No. 11 p. 2186-7	Qualtitative Determination of the presence of Azides (explosive) in the combustion residue of N2H4-N2O4.	Liquid N2H4, N2O4	NA	Rocket Engine	Empirial, Qualitative	Doug Banning	DWB 0
122	3	Popping Phenomena with the Hydrazine Nitrogen-Tetroxide Propellant System J. Houseman, A. Lee, April 1972. J. Spacecraft, Vol. 9, No. 9, September 1972	Investigation of the "Popping" Phenomena in impinging steams of N2H4 and N2O4 using streak photography. Discusses the nature of "pop" propagation, mechanism, and popping rate correlations to various physical parameters.	Liquid N2H4, N2O4	NA	Rocket Engine Combustion Chamber	Empirical, Qualitative Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
123	3	Analysis of Reaction Products of Nitrogen Tetroxide with Hydrazine under Nonignition Conditions M. Saad, M. Detweiler, et al. AIAA Journal, Vol. 10, No. 8, 1972	Low temperature-low pressure reaction products of hypergolic fuels under non-ignition conditions are determined and possible reaction kinetics are postulated.	Liquid N_2O_4, N_2H_4, MMH, UDMH	NA	NA	Empirical, Qualitative	Doug Banning	DWB 0
124	1	Atmospherical Modeling and the Chemical Data Problem D. Garvin, R. Hampson AIAA Paper No. 73-500, June 1973	Modeling chemical reactions that may take place in the stratosphere involving O, O_3, NO_2, OH, and HNO_3. (Applicable to high elevation pollution problems.)	Liquid O, O_3, NO_2, HO, HNO_3	NA	Stratosphere	Empirical, Qualitative	Doug Banning	DWB 0
125	5	Preignition Products from Propellants at Simulated High-Altitude Conditions S. Mayer, D. Taylor, et al., May 1969 Combust. Science and Technology Vol. 1, 1969	Identification of preignition reaction products for hypergolic fuels used at high altitude using IR and mass spectroscopy. Also data on solubility and volitility of residues.	Liquid MMH, UDMH, A-50, N_2H_4, N_2O_4	NA	High Altitude Rocket Motor Combustion Chambers	Empirical, Qualitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
126	8	Size and Duration of Fireballs from Propellant Explosions J. Gayle, J. Bransford August 1965 NASA TM X-53314	Determination of fireball size and duration for several propellant mixtures and explosives as a function of reactant weight(s).	Liquid RP-1, LO2, LH2, N2O4, UDMH, N2H4, High Explosives	Large Scale Explosions of Fuels or Explosives	Fireball Size and Duration	Empirical, Quantitative	Doug Banning	DWB 0
127	2	NO, NO2 and HNO3 below 35km in the Atmosphere M. Ackerman, Dec. 1974 Journal of the Atmos. Sciences, Vol. 32, Sept. 1975	Investigation into the abundance and location(s) of NO, NO2, and HNO3 in the upper atmosphere (pollution study). Surface maps, height profiles, and number densitys are presented.	Liquid NO, NO2, HNO3	NA	Upper Atmosphere	Empirical & Theoretical, Qualitative, Quantitative	Doug Banning	DWB 0
128	1	Investigation of the Rate Coefficient for O(3P) + NO2 02 + NO* T. Slanger, B. Wood, et al. 1973 Int. Journal of Chem. Kinetics, Vol. 5, p 615-620, 1973	Determination of the rate of constant for the reaction O(3P) + NO2 02 + NO*	Liquid O, NO2, 02, NO*	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0
129	5	The Rate Constant for the Reaction 03 + NO2 02 + NO3 Over the	Determination of the rate constant and temperature depend-	Liquid 03, NO2, 02, NO3	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
129	Cont	Temperature Range 259-362 K R. Huie, J. Herron, May 1974, Chemical Physics Letters Vol. 27, No. 3, August 1974	ance of the reaction 03 + NO2 02 + NO3.						
130	7	The Structure of the Reaction Zones of Ammonia/Oxygen and Hydrazine-Decomposition Flames D.I. Maclean, H. Wagner 1967 11th Int. Symposium on Combustion 1967, p. 871-878	Determines concentration and temperature gradients in NH3/02 and N2H4 flames as a function of position in the flame.	Liquid NH3, 02, N2H4	NA	Thermal, Concentration in Flames	Empirical, Quantitative	Doug Banning	DWB 0
131	5	Oxidation of 1,1-Dimethylhydrazine by Oxygen M. Miller, H. Sisler March 1980 Inorg. Chem. 1981, 20, 429-433	Determination of the mechanism and rate constant of the oxidation of UDMH by oxygen.	Liquid 02, UDMH	NA	NA	Empirical, Qualitative	Doug Banning	DWB 0
132	5	Reactions of Hydrazine with Ozone under Simulated Atmospheric Conditions E. Tuazon, W. Carter,	Determination of the reaction products of hydrazines with ozone in the atmosphere usng FT-IT	Liquid N2H4, MMH UDMH, 03	NA	Pollutants in the Atmosphere	Empirical, Quantitative Qualitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
132	Cont	et al. 1981 Eviron. Science Tech. Vol. 15, No. 7, July 1981	instrumentation.						
133	7	Direct Mass Spectrometric Measurements in a Highly Expanded Rocket Exhaust Plume T. McCay, H. Powell, et al. J. Spacecraft Vol. 15, No. 3, 1978	Determination of combustion products from the ignition of MMH/N204 at various O/F ratios.	Liquid MMH, N204	NA	Pollutants in the Upper Atmosphere	Empirical, Qualitative, Semi-Quantitative	Doug Banning	DWB 0
134	3	A Basic Study of the Nitrogen Tetroxide-Hydrazine Reaction. H.G. Weiss, July 1965 Dynamic Science Corp. N65-30838, 1965	IR Spectra, thermochemistry and miscibility of N204, N2H4 and mixtures of N204 and N2H4. Combustion products were also determined	Liquid N204, N2H4	NA	NA	Empirical, Quantitative Qualitative	Doug Banning	DWB 0
135	6	The Autoxidation of Hydrazine Vapor D.A. Stone, Jan. 1978 CEEDO-TR-78-17	Determines the "fate" of hydrazine in air at various pressures and surface to volume ratios. (Good analysis of hydrazine as a pollutant.)	Liquid N2H4, 02	NA	Atmospheric Pollution	Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
136	5	Identification of Important Chemical Reactions in Liquid Propellant Rocket Engine S. Cherry, L. VaNice TRW 1968 Western States Section Combust. Inst. WSS/C1-68-7, 1968	Determination of reaction rate constants for several specific fuel mixtures and rocket geometries.	Liquid 02, H2, F2, N204, A-50, CTF, N2H4, OF2, B2H6, RP-1, L02, FLOX	NA	NA	Theoretical, Quantitative	Doug Banning	DWB 0
137	5	Experimental Investigation of the Enthalpy of the Chemically Reacting System N204, 2NO2, N20 + 02 A.E. Sheindlin, N.I. Gorbunoiva, et al. Soviet Physics-Doklady Vol. 14, No. 6 December 1969	Determines the enthalpy of the chemically reacting system N204, 2NO2 2NO + 02 over a wide range of isobars.	Liquid N204	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0
138	6	A Semi-Empirical Analysis of the Hypergolicity of Gas-Gas and Gas-Liquid Reactions of N2H4-N204 Type Propellants P. Choudhury, P. Wilber International Symposium A68-34787	Discusses kinetic Theory, spontaneous ignition temperature and activation energy of the hypergolic reaction betwqeen H2H4 and N204	Liquid N2H4, N204, MMH	NA	NA	Semi-Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
139	5	The Vapor Phase Autoxidation of Unsymetrical Dimethylhydrazine and 50-Percent Unsymetrical Dimethylhydrazine-50-Percent Hydrazine Mixtures D.A. Stone, Environmental Sciences Lab ESL-TR-80-21, April 1980	Determines the rate, kinetics, and IR spectra of the aut-xidation of N2H4, UDMH, A-50, and reaction products.	Liquid UDMH, N2H4 A-50	NA	NA	Empirical, Qualitative	Doug Banning	DWB 0
140	6	A Basic Study of the Ignition of Hypergolic Liquid Propellants. L.B. Zung, B.P. Breen February 1970 NASA N70-27380	Quantitative analysis of vapor phase reaction products of UDMH and MMH with N2H4, ignition studies of OF2 and B2H6, and theoretical analysis of ignition mechanisms. (Includes kinetics and boundry conditions.)	Liquid N2H4, N2O4 UDMH, MMH, OF2, B2H6	NA	NA	Empirical, Theoretical, Quantitative	Doug Banning	DWB 0
141	1	Nitrogen Dioxide Photolytic, Radiometric, and Meteorological Field Data. J. Sickles, L.A. Ripperton, et al.	Compilation of data for the photolysis of NO2 in the atmosphere. Rate constant data is presented as a	Liquid NO2	NA	NA	Empirial, Quantitative	Doug Banning	DWB 0

NA Not Applicable

NI Not Identified

Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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141	Cont	EPA-600/7-78-053	function of solar altitude, cloud cover, and geographic location. (Air pollution study.)						
142	2	Combustion Process of Impinging Hypergolic Propellants L.B. Zung, J.R. White May 1971 NAS CR-1704	Photographic examination of combustor pressure poppings, stream mixing, and stream separation for impinging stream of N2H4 and N204 for various configurations.	Liquid N2H4, N204	NA	Rocket Engine Combustor	Empirical, Qualitative	Doug Banning	DWB 0
143	10	Blast and Fireball Comparison of Cryogenic and Hypergolic Propellants R.E. Pesante, M. Nishibayashi 1964 NASA-CR-69088	Shockwave velocities fireball size and duration, total radiation yield, calculated fireball temperatures, temperature records, and blast yields for several configurations of LO2/RP-1 and N204/A-50 Propellant spills. (Excellent reference for SPHAM manual-explosion section.)	Liquid N204, A-50, LO2, RP-1	Sudden fuel spill with resulting explosion and fireball initiated by a drop or impact by a fragement.	Blast, fireball, thermal, shockwave	Qualitative, Quantitative Empirical Theoretical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
144	2	Combustion of Liquid Hydrazine with Removal of the Gaseous Products Directly from the Flame. A.P. Aleksecv, G.B. Manelis Ccmbustion, Explosion, and Shockwaves, Vol. 16, No. 4, Jan. 1981	Effect of pressure and rate of heat removal on the rate of combustion of pure N2H4 (pressures ranged from 1-27.5 atm, combustion initiated in 8 and 10.8 mm diam. quartz tubes. Combustion gasses were directly removed from the flame into a vacuum and analyzed).	Liquid N2H4	Ignition of Liquid N2H4	NA	Empirical, Quantitative	Doug Banning	DWB O
145	10	Atmospheric Dispersion of Hypergolic Rocket Fuel Phase II S. Prince, W.R. Haas, et al., March 1984 Contract F42600-81-D-1379	Evaluation of fire-balls resulting from a large-scal spill and ignition/explo-sion of A-50/N204. Included are chem-ical reactions of fuels and combustion products with air, condensation of fuels out of the fireball, and a users manual for the computer program HARM (Theoretical	Liquid N204, A-50	Catastro-phic Spill in Silo	Thermal, Fireball	Theoretical, Quantitative	Doug Banning	DWB O

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
145	Cont		model for fireball size and dispersion using spill size and atmospheric conditions as inputs) developed during this study.						
146	2	Compatability of Materials with Rocket Propellants and Oxidizer W.K. Boyd, W.E. Berry, E.L. White DITC Memorandum 201, January 29, 1965	Classification of materials of construction as to their compatability and rate of corrosion when exposed to the various propellants and oxidizers as a function of temperature. Most material are metals, but some non-metals were tested.	Liquid NH3, B5H4, F2, FLOX, OF2, O3F2, CTF, BTF, BPF, IPF, PF, N2H4, MMH, UDMH, A-50, H2, H2O2, CH2, CH2Cl2, RFNA, N2O4, O2, O3, Solid Prop's	Contact of Propellant with Materials	NA	Qualitative, Empirical, Literature Search	Doug Banning	DWB 0
147	2	Heat Release Rate for the Liquid N2O4/N2H4 Reaction by Somogyi and Feiler's Method SN-114 B.R. Lawver, E.A. Tkachenko, February 1969 NASA-CR-73608	Determines the heat release rate for impinging streams of N2H4 and N2O4 as a function of injection velocity.	Liquid N2O4, N2H4	NA	Thermal	Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
148	7	Prevention and Extinguishment of Fires Involving Hypergolic Propellants D.S. Burgess, T.A. Christos, et al. October 1968 NASA-CR-101226	Prevention of hypergolic ignition, evaporation rates, vapor pressures and concentrations above a pool, fire extinguishment and/or prevention by the application of H2O, CO2, CO2/Halon 1301, corrosive characteristics and combustion products for the hydrazine-type hypergolic propellants and N2O4.	Liquid N2O4, A-50, MMH, N2H4, UDMH	Hypergol Spill	Thermal, Toxic	Empirical, Quantitative	Doug Banning	DWB 0
149	4	Gas Phase Ignition of Hydrazine with Nitrogen Dioxide H. Miyajima, H. Sakamoto Combustion Science and Technology Vol. 8, 1978 p. 199-200	Measurement of ignition delay time for impinging stream of NO2 and N2H4 as a function of temperature and pressure	Liquid N2H4, NO2	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0
150	2	Study of the Reaction Mechanism and Thermal Decomposition of Certain Storable Propellants Rocketdyne Research, December 1980	Rate of decomposition of hydrazine as a function of temperature and vessel material, (kinetics and arrhenius rela-	Liquid N2H4, B5H9	NA	NA	Empirical, Qualitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID. 00C1W AD. 0643G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
150	Cont	AFFTC TR-61-30	tionship), and analysis of the Pentaborane-Hydrazine reaction.						
151	3	Apollo 16 Mission Anomaly Report No. 1, Oxidizer Deservicing Tank Failure Mission Evaluation Team, June 1972 NASA-TM-X-68604	Analysis of accident involving ground support equipment designed to remove and neutralize liquid and gaseous N204/NO2 from the Apollo command module RCS.	Liquid N204	High rate generation of gases due to an insufficient quantity of neutralizer in scrubber causing scrubber tank failure.	Explosive	Analysis of Actual Accident	Doug Banning	DWB 0
152	10	Atmospheric Dispersion of Hypergolic Rocket Fuels Phase I: Source Characterization Final Report S. Prince, Sept. 1982 Contract F42600-81-D-1379	Describes the chemistry of the Hydrazine/N204 reactions, adiabatic flame temperatures in various configurations, calculation of fireball size and heat flux, and vaporization of	Liquid N204, N2H4, UDMH, A-50	Large Scale Accident in Titan II Silo	Toxic, Thermal, Overpressure	Theoretical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
152	Cont		excess propellants for an accident involving A-50 and nitrogen tetroxide.						
153	10	Propellant Spill Analysis D. Banning, July 1983 No ID #	Chemical reaction stiochiometry, rate of mixing for MMH/ N2O4 spill, thermo-chemical calcul-ations, maximum static pressures, and shock wave over-pressures where app-licable for MMH/N2O4 and N2H4 mono spills. Also inves-tigates the effects of a water deluge system and air venting systems on the severity of all possible accidents including N2O4 only and MMH only spills. Includes computer program to determine thermal and over-presure data for these type accidents	Liquid N2O4, N2H4, MMH	Large Propellant Spill inside Payload Prepara-tion Room, VAFB	Toxic, Thermal, Blastwave, Overpressure	Theoretical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
154	6	Radiation from Rocket-Exhaust Plumes R. Zirkind, Dept. of Aerospace Engineering and Applied Mechanics, Polytechnic Institute of Brooklyn No ID #	Spectral radiance, isoradiance, plume structure, boundry-layer temperature profiles, far-field temperature profile, and fluid mechanics of several propellants exhausted from several different nozzle configurations.	Liquid LO2, RP-1, Kerosene, A-50, N2O4, Metalized Solid Propellant	NA	Thermal	Empirical, Quantitative	Doug Banning	DWB 0
155	6	Reactions and Expansion of Hypergolic Propellant in a Vacuum J. Simmons, R. Gift, et al. AIAA Journal, May 1968 p. 887-893	Determine the rate and type of reaction between A-50 and N2O4 at absolute pressures of 1 X 10^{-5} to 1 atmosphere.	Liquid A-50, N2O4	NA	Pressure, Temperature	Empirical, Qualitative, Quantitative	Doug Banning	DWB 0
156	5	The Autoxidation of Monomethylhydrazine Vapor. D.A. Stone, April 1979 ESL-TR-79-10	Determines the kinetics and reaction rate for the autoxidation of MMH in air.	Liquid MMH	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0
157	5	Nitrogen Dioxide Absorption in Evaporating and Condensing Water Droplet	Kinetics and rates for the absorption of NO2(g) into	Liquid NO2	NA	Air Pollution from N2O4 & Burning	Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
157	Cont	J. Herrmann, M. Matteson November 1977 Recent Advances in Air Resourse Management Symposium ORO 5592-1	condensing and evaporating water "fogs".			Hydrocarbons			
158	5	Review of the Hydrazine/ Oxygen Reaction Kinetics S. Dalgaard, M. Sanford International Corrosion Forum, April 1981 Paper No. 15	Kinetics and reaction rates for the reaction of hydrazine with oxygen. Also temperature dependence of the rate constant	Liquid N2H4, 02	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0
159	10	Heat Transfer Hazards of Liquid Rocket Propellant Explosions J.A. Mansfield AFRPL-TR-69-89, February 1969 (Project PYRO)	Characteristics and heat transfer within a fireball, gas temperature, heat flux density, and radiant flux density for several fuel/ oxidizer combination and configurations. (Good source for SPHAM.)	Liquid LO2, LH2, RP-1, N2O4, A-50	Large-Scale Propellant Spills	Blast, Shockwave, Explosion, Thermal	Empirical, Quantitative	Doug Banning	DWB 0
160	3	Thermodynamic and Thermophysical Properties of Combustion Products, Vol. IV:	Enthaiples, heats of formation, heats of mixing, density, transport properties	Liquid N2O4, UDMH, MMh, N2H4, A-50, B5H9	NA	NA	Empirical, Quantitative	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
160	Cont	Nitrogen Tetroxide-Based Propellants. V. Alemasov, A. Dregalin et al. TT-76-50007-Vol-4	and rate constants are presented for several hypergolic propellants individually and in combination with one another.						
161	5	Titan III Destruct System, N. C. Spencer Pan American, World Airways, Inc. Cape Kennedy Air Force Station, FL, Minutes of the 10th Explosive Safety Seminar, 1968	Information only.	NA	NA	NA	NA	Explosive Safety Board	GM *
162	10	Hazardous Explosive Problems at the Kennedy Space Center, F. X. Hartman, AASA NASA, JFK Space Center Cocoa Beach, FL Minutes of the 10th Explosive Safety Seminar 1968	Propellant weights, locations (Stages) and TNT equivalencies and quantity distance are given for the Apollo/Saturn V Space Vehicle	Liquid & Class 2 solid propellant N_2O_4 MMH Aerozine 50 LO_2 LH_2 RP-1	NA	NA	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
163	9	Far Field Overpressure from Closely Spaced Sequential Detonations, T. A. Zaher, IIT Res. Institute, Minutes of the 11th Explosive Safety Seminar, 1969	This paper describes a program of numerical analysis of sequential explosions designed to determine quantitatively as a function of distance from the explosion site, time delay between detonations, and charge weights of the explosions.	NI	Explosion	Blast (pressure, velocity, & distance)	Quantitative Qualitative	Explosive Safety Board	GB *
164	9	Watch Your Equivalent Weight, Naval Ordnance Laboratory J. Petes, NOL Naval Ordnance Lab, Silver spring, MD Minutes of the 12th Explosive Safety Seminar 1970	How to determine the airblast equivalent weight of any particular explosion or explosive	Solid Propellant	Explosion	Blast (airblast)	Quantitative Qualitative	Explosive Safety Board	GB *
165	5	Applying Systems Analysis to Ordnance Production Systems, R. J. Firenze, NOSC Safety School, Crane, IN Minutes of the 12th Explosive Safety Seminar 1970	Objective is to acquaint the hazard control specialiest with the methods and techniques of systems analysis and describes how to apply these tools. Paper is incomplete.	NA	NA	NA	Qualitative	Explosive Safety Board	GB/BL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843C

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION						
			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
166	7	Fragmentation Hazards to Unprotected Personnel D. I Feinstein, et. al ITT Research Institute Minutes of the 13th Explosive Safety Seminar 1971	Utilizes current data and techniques to establish safe distance for personnel from fragments.	Munition	Explosion	Fragmentation	Quantitative Qualitative	Explosive Safety Board	GB *
167	9	Sequential Explosion Studies, IIT Research Institute, J. J. Swatosh, Jr. IIT Research Institute Minutes of the 13th Explosive Safety Seminar 1971	To determine blast pressures, peak impulse and catchup times of shock waves generated by sequentially detonating two explosive charges.	Explosive charges	Explosion	Blast (peak pressure, peak impulse)	Quantitative Qualitative	Explosive Safety Board	GB *
168	5	Explosive Yield Criteria Studies at the Ballistic Research Laboratories, G. D. Teel, Minutes of 13th Explosive Safety Seminar, 1971	Didn't offer any useful information, methodology or test results.	NA	NA	NA	NA	Explosive Safety Board	GB *
169	7	Trajectory Calculations in Fragment Hazard Analysis, Armed Services Explosive Safety Board, T. A. Zaner, Minutes of the 13th Explosive Safety Seminar, 1971	Provides different equations for ballistic trajectories. Equation were compared to test results for accuracy.	Ammunition	Explosion	Fragmentation	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
170	10	Blast Criteria for Personnel in Relation to Quantity - Distance, D. R. Richmond & E. R. Fletcher, Lovelace Foundation for Medical Education & Research, Minutes of the 13th Explosive Safety Seminar, 1971	Provides Blast Criteria for Personnel from overpressure effects, blast displacement of the individual and fragments.	NI	Explosion	Blast (Overpressure shock wave) Fragmentation	Quantitative	Explosive Safety Board	GB *
171	8	A Note on Fragment Injury Criteria, Ballistic Research Laboratories, W. Kohinakis, Ballistic Res. Labs., Minutes of the 13th Expolsive Safety Seminar, 1971	Criterion for predicting the injury to people from munition fragments.	Munition	Detonation	Fragmentation	Quantitative	Explosive Safety Board	GB *
172	9	Fragment Hazard Criteria D. I. Feinstein, ITT Research Institute Minutes of the 13th Explosive Safety Seminar 1971	Discussion relates to a review of data pertaining to the effects of fragments upon the human body and the rational development of a casualty criteria developed from this data.	NA	NI	Fragmentation	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
173	8	Critical Mass (Hypothesis and verification) of Liquid Rocket Propellants, E. A. Farber Solar Energy & Energy Conversion Lab. Univ. of FL, Minutes of the 13 Explosive Safety Seminar 1971	This paper presents the concept of critical mass, which is the quantity of fuel and oxidizer when mixed together that generate an explosion Experimental studies presented verify theory.	Liquid $LO_2/RP-1$ LO_2/LH_4	NI	Blast	Quantitative	Explosive Safety Board	GB *
174	9	Safety of Prepackaged Liquid Propellant Rocket AFRPL, F. S. Forbes et.al Air Force RPL, Minutes of the 13th Explosive Safety Seminar, 1971	This paper describes the design features and the extensive testing and evalua- tion procedures that are used to assure safety for liquid rocket missiles. A review of Bullpup, Lance, Condor safety criteria, classification tests, and experience is presented	Liquid MMH UDMH Nitric Acid Nitrogen Tetroxide Chlorine Triflouride	NA	Blast Thermal	Qualitative	Explosive Safety Board	GB *
175	7	Initiation of Explosives by Fragment Impact, J. J. Paszek, V.M.Boyle ABL,MO, Minutes of the 15th Explosive Safety Seminar, 1973	The objective was to develop sufficient quantitative data to generate predictive model for initiation of explosives.	NI	NI	Fragmentation	Quantitative Test Results	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
176	9	Explosive Yield Limiting Self-Ignition Phenomena in LO_2/LH_2 and LO_2/RP-1 Mixture E. A. Farber, Univ. of Florida, Gainsville, FL Minutes of the 15th Explosive Safety Seminar 1973	Summary of the work Dr. Farber has done in arriving at credible explosive yield values for liquid rocket propellants. The results are based upon logical methods which have been well worked out theoretically and verified through experimental procedures. The three independant methods developed for this purpose are: 1. The Mathematical Method 2. The Seven Chart Approach 3. The Critical Mass Method	Liquid LO_2/LH_2 LO_2/RP-1	Explosion	Blast (Explosive Yield)	Quantitative	Explosive Safety Board	GB *
177	10	Blast Hazards of CO/N_2O mixtures, AFRPL, Laser, D.R. Mastromonico and F. S. Forbes, Edwards AFB, CA Minutes of the 15th	The purpose of this program is to make blast hazard mixing determinations on CO and N_2O in their various physical sta-	CO/N_2O Liquid Gas	NA	Blast (Peak overpressure, and explosive yield) Thermal	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD. 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
177	Cont	Explosive Safety Seminar 1973	tes under several mixing conditions. This information will provide the basis for establishing structural and quantity distance safety criteria for an AF test facility. Consideration is also given to CO and air mixtures.			Fragmentation			
178	3	Air-Blast Pressure Measurement Systems and Techniques, L. Giglioros et. al, Ballistic Res. Labs. Aberdeen PG, MD Minutes of the 15th Explosive Safety Seminar 1973	This paper gives details for specifying and implementing instruction for the gathering of airblast overpressure data.	NI	NI	Blast (Overpressure)	NA	Explosive Safety Board	GB *
179	7	Initation Mechanisms of Solid Rocket Propellant Detonations, IIT Research Institute, H. S. Napadensky, et. al ITT Res. Inst. Chicago Illinois, Minutes of the 15th Explosive Safety Seminar, 1973	An experimental and analytical investigation was conducted to determine the mechanisms which may lead to the initiation of detonation in solid rocket propellants under accid-	Solid Propellants (PBAN and ANP had aliminum as fuel and ammonium perchlorate	Impact Velocity	Thermal	Quantitative (Test Results) Qualitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
179	Cont		entally applied low-amplitude stimuli. In particular, the effects of moderate speed impact are studied.	as oxidizer)					
180	9	Measurement of Pressure and Impulses of Gaseous Explosions at High Initial Pressures, A. B. Wenzel, Minutes of the 16th Explosive Safety Seminar, 1974	This paper presents the results of laboratory scale experiments based on a model analysis using Hopkinson's and/or Sach's scaling laws of gaseous explosions in an oil orgas well bore as a function of initial pressures.	Oxygen Methane	NI	Blast (Impulse, Pressure)	Quantitative	Explosive Safety Board	GB *
181	5	The Roles of Deflagration and Explosiveness in Hazard Assessment, P. J. Hubbard & P.R. Lee Royal Arsenal, Woolwich, U.K., Minutes of the 17th Explosive Safety Seminar, 1976	Data on deflagration versus denotation of explosives when in confinement.	Solid RDX/TNT TNT TORPEX 4D RDX/PU RTX/WAX/AL	Accidental initiation of explosives by stimuli which provide less energy than required for detonation	Blast Fragmentation	Experimentation	Explosive Safety Board	RL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
182	5	Fragment Penetration Into Single and Spaced Target, Ballistic Research Laboratory, T. Riechiazzi, J. Barb ABL, MD, Minutes of the 17th Explosive Safety Seminar, 1976	Primarily this paper presents results from a experimental study of light and heavy fragment terminal ballistic performance against single and double spaced targets of equivalent thickness.	NA	NI	Fragmentation	Quantitative	Explosive Safety Board	GB *
183	5	Non-Ideal Explosions The Blast Wave from Low Energy Density Explosion Sources, Dept. of Aeronautical and Astronautical Engineering, R. A. Strehlow Univ. of IL, Minutes of the 17th Explosive Safety Seminar, 1976	Presents theoretical work on far field equivalency, energy flows during wave propagation and development of scaling laws.	NI	Explosion	Blast	Quantitative (Test Results)	Explosive Safety Board	GB *
184	10	The History of the Quantity Distance Tables for Explisive Safety Army Ballistic Research Laboratories, Ona R. Lyman Minutes of the 18th Explosive Safety Seminar 1978	Presents Quantity distance regulations as defined in AMCR-385-100 and TM 9-1300-206	Munition	NA	NA	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
185	7	Blast Measurements at Close Standoff Distance for Various Explosive Geometries, J.J. Kulesz et. al, Minutes of the 18th Explosive Safety Seminar, 1978	Objective - to expand blast technology of pressure and impulse versus scalled distance into the region of small scalled distances of 0.12 to 1.2 $m/kg^{1/3}$ (0.3 to 3.0 $ft/lb_m 1/3$)	Explosive Charges (Pentelite and composition C-4)	NA	Blast (Peak overpressure and impulse)	Quantitative (Test Results)	Explosive Safety Board	GB *
186	5	Reflect Blast Measurements Around Multiple Detonation Charges, J. C. Hokanson & A. B. Wenzel Minutes of the 18th Explosive Safety Seminar 1978	This paper presents the results of an experimental program in which blast parameters were measured at close standoff distance from three simutaneously detonated charges.	Explosive Charges	NA	Blast (Peak overpressure and impulse)	Quantitative (Test Results)	Explosive Safety Board	GB *
187	8	Airblast Enhancement from Multiple Detonation J. Keefer, Minutes of the 18th Explosive Safety Seminar, 1978	This paper will review multiple detonations from two to twenty-four charges and both simulaneous and non simultaneous detonations.	TNT	NA	Blast (Peak and dynamic pressures) Thermal	Quantitative (Test Results)	Explosive Safety Board	GB *
188	7	Fragment Hazard Investigation Program, NSWC,	Provides methodology for the determination	TNT loaded projectiles	NA	Fragmentation	Quantitative	Explosive Safety	GB *

NA Not Applicable
MI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
188	Cont	R. T. Ramsey, et. al Minutes of the 18th Explosive Safety Seminar 1978	of quantity-distance standards for mass-detonating hazard materials (Class 1, Division 1)					Board	
189	10	Drag-type Fragment Motion in Two Dimensions, Minutes of the 19th Explosive Safety Seminar 1980	Approximate analytical solution for determining the motion (trajectory) of a projectile or fragment.	NA	NA	Fragmentation	Quantitative	Explosive Safety Board	GB *
190	8	Fragment Hazard Investigation Program (Large Scale Detonation Tests), NSWC, J.G. Powell & W. D. Smith III Minutes of the 19th Explosive Safety Seminar 1980	Methodology for predicting far field fragment density to determine standards for Class 1, Division 1 materials	Solid Class 1, Division 1	NI	Blast Fragmentation	Quantitative	Explosive Safety Board	GB *
191	7	Metric Quantity-Distance Based on Blast Impulse, NWC, G. F. Kinney, R. G. S. Sewell K. J. Graham, Minutes of the 19th Explosive Safety Seminar, 1980	Safe distance from blast based on blast impulse	TNT Equivalency	Explosion	Blast	Quantitative	Explosive Safety Board	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
192	5	Comparison of Predictive Methods for Structural Response to HE Blast Loads, Army Corps of Engineering, W. T. Char Minutes of the 19th Explosive Safety Seminar 1980	Comparison of Predictive Methods for Structural Response to HE Blast Loads.	Gas HE (High Explosive)	Internal and External HE Tank Burst	NI	Qualitative	Explosive Safety Board	GB *
193	10	Yield and Blast Analyses with a unified Theory if Explosions, NSWC, F. B. Porzel, Minutes of the 20th Explosive Safety Seminar, 1982	Provides methodology to relate all explosions (blast, thermal and fragments) non-ideal, any media, over all ranges in air, underwater, underground, confined spaces etc.	NA	Explosions	Blast Thermal Fragmentation	Quantitative	Explosive Safety Board	GB *
194	5	Safety Analysis Report for the Vehicle Assembly Building Concerning Inadvertent Ignition of a Solid Rocket Booster or Segment, April 1981, NASA	Talks about the probability of an inadvertent ignition of a SRM in the VAB, also provide a reference to a follow on study assessing the hazards after ignition.	SRM Solid	Inadvertent Ignition	NI	Qualitative	In-house	GB *

NA Not Applicable
NI Not Identified
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ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
195	2	Fracture Control Criteria for Space Shuttle Pressure Vessels and Pressurized Structures TOR-0076 (6451-03)-2, Dec. 1975 N. N. Au	Requirements on structural design, factors of safety, materials, test, and documentation to insure that structural failures due to crack initiated fractures will not occur.	NA	Crack-initiated fracture	Fragments	Qualitative	Mangino	EK *
196	1	Traffic and Safety Impact of STS Consumables Delivery at Vandenberg AFB, SAI 80-005-ATO, Sept. 1979	Concerns safe transportation, handling, and storage of large quantities of propellants and other liquid/gaseous supplies over public highways	NA	NA	NA	Qualitative	Mangino	EK *
197	6	SPIF Emergency Procedure Document, SPOC-SF-001, Basic, McDonnell Douglas H. J. Hutchinson	Document provides emergency instruction for payload processing teams in the SPIF in the event of emergency resulting from several failure scenarios.	Nitrogen Helium Hypergolics Cryogens Ordnance Solid Propellant Battery Electrolyte Hydrogen	Toxicants. Contact cryogens. Release Hydrogen. Premature ignition of solid propellants.	Blast Fragmentation Fire Toxicity	NA	Joe Mangino	RL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
198	6	Preliminary Estimates of Environment Exposure for fuel and exhaust products. K. D. Hage N. E. Bowne, 1965, Vol 1 The Travelers Research Center, Inc. NAS 8-11450 MSFC/NASA 10015	The methods and Preliminary estimates for rocket propulsion engines were mathematically modelled. Experimental verification was done to measure horizontal & vertical diffusion coefficients for the gaseous cloud formation	Hydrogen Fluorine	Liquid spills	Blast	Quantitative	R. F Fletcher	KKM *
199	8	Overall Safety Manual, Vol. 1 Summary, NUS Corp., July, 1981 NUS Corp. & Research Place Rockville, MD 20850	Provides information on nuclear safety analysis for space applications, with general description of risk analysis and data on launch activities. Contains some safety design requirements and mission exposure limits.	Radioisotope	NI	NI	Qualitative Quantitative	NUS Bart Bartram	RL *
200	8	Overall Safety, Vol. II Technical Models, June 1975 NUS Corp. & Research	Probabilistic source term models, meteorological and demographic models atmos-	Radioisotope	NI	NI	Qualitative	NUS Bart Bartram	RL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
201	Cont	Place, Rockville, MD 20850	pheric and aquatic dispersion models, inhalation ingestion and exposure doses.						
201	8	Overall Safety, Manual, Vol III, Reference Data, Jan, 1975 NUS Corp. & Research Place, Rockville, MD 20850	Provides additional relevant data to worldwide and launch site environments including population, meteorology, soils, topography, ocean depths and currents, and land use.	Radioisotope	NI	NI	Quantitative	NUS Bart Bartram	RL *
202	8	Overall Safety, Manual, Vol IV, Instruction Kit, July, 1981 NUS Corp. & Research Place, Rockville, MD 20850	Supplemental information for OSM updates Contains longterm population risk model TDOS users manual, plutonium environmental behavior bibliography and worldwide demographic update.	Radioisotope	NI	NI	Quantitative	NUS Bart Bartram	RL *
203	4	Accident Prevention Plan Air Force Eastern Test Range, April 1973	General accident prevention manual delineating the roles and responsibilities of	NI	NI	NA	NA	Joe Mangino	RL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
203	Cont		personnel at the AFETR to eliminate accidents. Includes industrial, flight and missile system Safety action plans.						
204	3	Safety Engineering Bulletin No. 3, System Safety Analysis Techniques, Dec. 1982 G. B. Mumma	Purpose of bulletin is to provide detailed information on safety analysis techniques. This draft copy has TBD sections and is not useful.	NI	NI	NA	General Techniques per MIL-STD-882 and MIL-STD-1574.	Joe Mangino	RL *
205	3	Vehicle Handling and Servicing, Martin Marietta, Aug. 1965 D. P. Wood	Training manual for handling, erecting and servicing the Titan III X launch vehicle at the WSMC. Significantly out of date.	NI	NI	NI	NI	Joe Mangino	RL *
206	5	Viking '75 Project, Integrated Spacecraft Handling Plan, PL-3720357, Dec. 1974 NASA, Langley	Provided information on systems design and handling data for the Viking Lander and Spacecraft operations. Data is not	Liquid Hydrazine Mono-Methyl Hydrazine Nitrogen	NI	NI	NI	In House	RL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
206	Cont		provided in the booster	Tetroxide Radiosotope Thermoelectric Generator Ordnance Nitrogen					
207	3	Viking '75 Project, Lander Hazardous Systems and Operations, Presentation, April, 1973 NAS1-9000 R. Bryant	Document designed for safety presentation for KSC/ETC on Lander Hazardous Systems and Operations. In view graph format. Much data was presented verbally and printed material is lacking in continuity and information. Of little value to SPHAM.	Liquid Hydrazine Mono-methyl Hydrazine Nitrogen Tetroxide RTG Ordnance Nitrogen	NI	NI	NI	In House	RL *
208	9	Giant Patriot - Incremental Velocity Curves, TRW, Memo 71-4342.2-21 March, 1971	Memo provides analytical models for predicting incremental velocities of Minute Man fragments following destruct action.	Solid Propellants	Destruct action	Fragmentation	Quantitative	Lou Ullian Richard Rogers	RL *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
209	7	Giant Patriot - Incremental Velocity of Destruct, TRW Memo 71-4342.2-18, March, 1971	Memo summarizes the incremental velocity data of debris generated by destruct action. It will be presented to AFWTR for use in Range Hazard Analyses.	NI	Destruct Action	Fragmentation	Quantitative	Lou Ullian Richard Rogers	RL
210	9	Fragment Initial Velocities and Trajectories, TRW Memo 66-3324.5-63, June 1966 and Memo 66-3324.5-82, August 1966	Memos' discuss methodology for determining fragment velocity based on the blast flow field effects	NI	NI	Blast Fragmentation	Quantitative	Lou Ullian Richard Rogers	RL *
211	10	Blast and Fragment Hazards From Bursting High Pressure Tanks, NOLTR-72-102, May 1972 J. F. P. Human	Report provides test results of pressurizing five (5) various diameter pressure vessels to the burst point. Data provided on test set up, overpressure, fragment dispersion, size and velocity and effects on anthropomorphic dummies.	Pressurant Nitrogen	Overpressurization to Burst	Blast Fragmentation	Quantitative	Lou Ullian Richard Rogers	RL *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
212	9	Data Correlation and Velocity Calculations for Giant Patriot Debris TRW, April 1971 8522-5-71-6 R. A. Lawrence	Provides debris list (velocity, impact energy, weight, size) that was compiled as as result of a destruct test using Giant Patriot.	NI	NI	Fragmentation	Quantitative (Test Results)	Lou Ullian Richard Rogers	GB *
213	10	An Improved Fragment Model for Internal Explosions of LO_2/LH_2 Propellant Tanks, Teledyne Energy Systems, Jan., 1981. Saturn IV CBM Missile, Shuttle TES-16018-05 D. C. Anderson & W. Owing	A model is proposed for predicting fragment sizes and speeds generated by an inertial explosion of Liquid Oxygen/Liquid Hydrogen propellant tanks.	Liquid LOX/LH_2	LOX Tank Overpressurization	Fragmentation	Quantitative	Riehl	GB *
214	10	Fragment Environments for Postulated Accident Sequences of the STS Vehicle, Shuttle TES-16018-02 D. Anderson, T. Olsen	Describes liquid and solid propellants models used for predicting fragmentation environments resulting from explosion scenarios.	Liquid Propellant LO_2/LH_2 LO_2/LH_2 RP-1	Refer to Document for all failure scenarios	Fragmentation (Speed, size, Flux . . .)		Lou Ullian Richard Rogers	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
215	9	Fragment Environment Modeling for STS Launch Accidents, AFWL-TR-79-167	Proposed scenarios which lead to accidental explosions of fuel. These are the basis for postulation of blast and fragment environments in this analysis.	Liquid LO_2 LH_2 Hydrazine Solid Propellants TNT Equivalency	Ref this document page 14	Blast Fragmentation	Quantitative Qualitative	Lou Ullian Richard Rogers	EK *
216	9	Break Up of the Space Shuttle Cluster via the Range Safety Command Destruct System, NSWC, Aug., 1977 J. Peters, et. al NASA H-13047B Amend 6 Task No. WR14ZAN01	Identifies the size distribution and weight of fragments resulting from STS breakup, also identifies the failure scenarios.	Liquid LOX/LH_2 Solid Rocket Booster	Refer to Document for complete list	Blast Fragmentation	Quantitative Qualitative	Lou Ullian Richard Rogers	GB *
217	9	Model Development for Explosive Debris Dispersion Studies, Acta Inc., Sept., 1983 Jon D. Collins	Study objective to identify a statistical method for determination of critical range for debris from walls of buildings which have had interior explosions.	NI	Building Explosion	Blast Fragmentation	Quantitative	Lou Ullian Richard Rogers	EK

NA Not Applicable
NI Not Identified
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218	1	Fragmentation Analysis LOX/LH_2 Explosion Aft of ET/Orbiter Shuttle No ID #	Table of contents and schedule. (4 pages) Order Analysis.	Liquid LOX/LH_2	NI	NI	NI	Lou Ullian Richard Rogers	GB *
219	7	Appendix I Fragment Velocity Distribution No ID #	Reports pooled data from several tests pertaining to fragment velocity distribution.	Liquid LO_2/LH_2 LO_2/RP-1	Confined by Missile Confined by Ground Surface	Fragmentation	NI	Lou Ullian Richard Rogers	GB *
220	9	Fragment Environment, Battelle	Prediction of fragment environment, mean fragment velocity, mean size and distribution, and fragment flux resulting from a launch vehicle explosion.	Liquid LO_2/LH_2 LO_2/RP-1	Explosion confined by structure between the propellant tanks and the payload	Fragmentation	Quantitative Qualitative	Lou Ullian Richard Rogers	EK
221	6	Safe Distance for Firing of Payload Liquid Propellant Engines NASA ET4/8302-8	To assess the probability of orbiter damage from an upper stage explosion	Liquid Bipropellant Monopropellant	Explosion	Fragmentation	Quantitative	Lou Ullian Richard Rogers	GB *
222	7	Handout information from Shrapnel Meeting on March 25, 1980 EH31/W. A. [illegible]ehl	Has fragment model, Appendix A	NI	NI	Fragmentation	Quantitative	Lou Ullian Richard Rogers	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
223	3	Launch Area Overpressure Probabilities No ID #	Not enough information (3 pages).	NI	NI	Blast Fragmentation	Quantitative	Lou Ullian Richard Rogers	GB *
224	7	History of Solid Rocket Motors Fragments Resulting from Destruct Action Memo	Provides results for fragments (velocity number and weight) from actual missile destruct analysis. Minuteman, Trident and Space Shuttle	Solid PBAA CXDB SRB	NI	Fragmentation	NA	DTIC	GB *
225	5	Scaling Law for Estimating Liquid Propellant Explosive Yield J. Spacecraft V. 15 N. 2 L. C. Sutherland	Refers to Project Pyro, which has already been evaluated. Paper is only 2 pages	NA	NA	NA	NA	Lou Ullian Richard Rogers	GB *
226	6	Initial Tabulation of Liquid Propellant Explosions involving 25,000 lbs and over, Jan., 1973 S&E-ASTN-M J. Petes	Summation of test data where propellant mixing initiated explosion.	Liquid LH_2/LOX RP-1/LOX RP-1/LOX/LH_2	NI	Blast	Quantitative	Lou Ullian Richard Rogers	GB *
227	3	Some Comments on ET(CBM) Yield No ID #	Not enough information present.	Liquid LOX/LH_2 LO_2/RP-1	CBM	Blast (Peak overpressure)	Quantitative Qualitative	Lou Ullian Richard Rogers	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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228	7	Report of Fragmentation Test Report, #0179-64F, Aerojet, May 1963	The objective of the test program was to obtain information and statistics relative to fragmentation effects that are produced when a solid-propellant motor is subjected to a high-energy internal explosion and impingement.	Solid Propellant	NA	Fragmentation	Quantitative (Test Results)	Lou Ullian Richard Rogers	GB
229	10	Workbook for Estimating Effects of Accidental Explosions in Propellant Ground Handling and Transport Systems W. E. Baker, et. al	Provides rapid methods for predicting damage and hazards from explosion of liquid propellants and compressed gas vessels used in ground storage, transportation and handling.	Liquid Ammonia Propane	Vessel Failure Fuel Leakage	Blast (explosive yield, pressure waves) Fragmentation (Distribution Velocity)	Quantitative	Lou Ullian Richard Rogers	GB
230	2	Thermal Environments: Stress and Strain Measurements in Strain Evaluation Cylinders, NWC TP 6466, Feb. 1984 R. F. Vetter	Evaluation of strain on cylinders (tests) and evaluation of service life of instrumentation.	NA	NI	NI	Quantitative (Test Results)	Lou Ullian Richard Rogers	GB

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
231	2	Operational Base Launch (OBL) Fire Probability Study, SMTC, VAFB, March 1974 J. L. Jantz, Vandenberg No ID #		NI	NI	Thermal	Quantitative Qualitative	Lou Ullian Richard Rogers	GB
232	5	Memo - Expected Casualties for Saturn 1B Impact on Miami No ID #	Presents calculations for determing casualties froma Saturn 1B impact.	Liquid LOX/LH_2 LOX/RP-1	Impact	Blast (over pressure)	Qualitative	Lou Ullian Richard Rogers	GB 0
234	6	Test Plan, 624A Solid Motor Sled Tset, April 1964, U.S. Naval Ordnance Test Station No ID #	Provides information on the hazards and damage potential of a 624A solid motor when subjected to dynamic impact.	Solid Propellant	Impact	Blast	Quantitative (Test Results)	Lou Ullian Richard Rogers	GB
235	6	1/8 Scale Model Missile Tests Phase 1, Hot Rocket Motor Frangible Silo June 1958 MT 59-8425	Determine explosive yield as a function of geometries and initiating mechanisms not a good copy.	Liquid LOX	Seam Rip	Blast		Lou Ullian Richard Rogers	GB *
236	4	Explosive Potential of "Atlas" Propellants, Final Summary Report Broadview Research Corporation, BRD-57-8A1 June 1957	Report presents results of scale testing of ATLAS models in 6 defined failure scenarios at ground level. TNT equiva-	Liquid Oxygen JP4	Seam failure Tank overpressurization	Blast	Quantitative	Lou Ullian Richard Rogers	RL

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
236	Cont		lency reported as high as 65%.		Fall back bulkhead rupture Missfire Pushover				
237	5	A Study of Liquid Propellant Autoignition, Contract NAS10-8591, May 1975 D. H. Lester, et. al	Report presents technical evaluation for causes of autoignition of liquid oxygen liquid hydrogen and liquid oxygen/kerosene fuel mixtures. Limited application to SPHAM.	Liquid Oxygen Liquid Hydrogen Kerosene Fuels	NI	NI	Quantitative	Lou Ullian Richard Rogers	RL
238	2	Fluid Flow Calculations for External Tank Destruct, D. Lehto, Surface Weapons Center, White Oak Lab, No date No ID # D. Lehto	Viewgraph presentation of effects of destruct event of the Space Shuttle External Tank.	Liquid Oxygen Liquid Hydrogen	Destruct Action	Blast Fragmentation	Quantitative	Lou Ullian Richard Rogers	RL

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
239	10	Statistical Analysis of project Pyro Liquid Propellant Explosion Data, TM-69-1033-3, July 1969 P. Gunther and G. R. Andersen	Provides data for assessing the hazards from liquid propellent rocket explosion and data for estimating explosive yield.	Liquid LO_2/RP-1 LO_2/LH_2	Controlled Ignition Confinement by missile Confinement by ground surface	Blast Thermal Fragmentation	Quantitative Qualitative	Lou Ullian Richard Rogers	EK *
240	6	Large Space Systems/Low-Thrust Propulsion Technology NASA Conference Publication 2144, May 1980 R. F. Carlisle	Inter disciplinary exchange of information addresses the potentially critical interactions between propulsion, structures and materials, and controls for large spacecraft; and net effects on large space systems	Electrical Propulsion Chemical Propulsion	NA	NA	Quantitative Qualitative	Joe Mangino Riley	EK *
241	2	Missile System Ground Safety Approval for DSV-3P-11E/F Delta Launch Vehicles, MDC Report G9455, May 1982 J. H. Harada et. al	Data for launch operations in conjunction with the DSV-3P-11F/F delta launch vehicles Pertains to NASA 391X and 392X vehicle	Liquid RP-1 LOX Oronite additive				Joe Mangino Riley	GB

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
241	Cont		launched from WSMC. Intended to satisfy requirements for a missile system ground safety approval package.	TEA, TEB Aerozine 50 Nitrogen Tetroxide Solid Propellant, Class 1.3 Ordnance Devices					
242	4	NASA/DOD Scout Ground Safety Report, Rev. M, July 1976 R. E. Kincade	Document provides detailed description of Scout vehicle and receipt, transportation preparation and launch activities. Provides no data on hazards or hazard analyses.	Hydrogen Peroxide Nitrogen Pyrotechnics Solid Propellants - Polyurethane - Polybutadiene - - DDP-80 Axcite 362M	NI	NI	NI	Mangino/ Riley	RL)

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
243	6	Defense Meteorological Satellite Program Block 5D July 1975 No ID #	Document provides detailed descriptions of defense meteorological satellite (DMSP) subsystems, ground systems and operations. No data provided on hazards or hazard analyses. Also provifdes limited information on Thor Booster Vehicle.	Liquid LO_2 RP-1 Nitrogen Hydrazine	NI	NI	NI	Mangino/ Riley	RL *
244	9	(T.O. 21M-HGM16F-1SS-4) Safety Supplement Operation Manual, USAF Series HGM-16F Missile, July 1964 No Author	Document provides description of missile (ATLAS) configuration facilities and AGE. Included are procedures for operations, emergency and trouble analysis, and malfunction. No hazard analyses are provided	Liquid RP-1 LO_2	NI	NI	NI	Mangino/ Riley	RL *
245	9	An Investigation of an Final Report Hazards Associated with the Storage and Handling of Liquid Hydrogen, Mar., 1960, AD324194 Arthur D. Little, Inc.	Provides experimental information on the results of hydrogen/air ignitions. Provides data on vapor cloud effects following LH_2 spills	Liquid Hydrogen Gaseous Hydrogen	Experimental spills	Blast Fire	Qualitative Quantitative	Joe Mangino Defense Documentation Center	RL

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
245	Cont		recommends design criteria for LH_2 storage and vapor vent systsms.						
246	8	Characteristics of Liquid Rocket Propellant Explosion Phenomena, Erich A. Farber, University of Florida, NASA Project NAS 10-1255, Parts VI and VII, July 1968 Saturn V	Paper consists of two papers; "Explosive Yield Estimates for Liquid Propellant Rockets Based Upon A Mathematical Model" and "Interpretation of Explosive Yield Values Obtained From Liquid Rocket Propellant Explosives."	Liquid oxygen RP Fuel Liquid Hydrogen	Rupture of barrier between oxidizer and fuel tanks	Blast	Quantitative	Rogers/ Ullian	GB
247	4	Hydrogen Technological Survey-Thermophysical Properties, 1975, NASA-SP-3089 R. D. McCarty	Thermo physical properties of hydrogen and voluminous references to source data are provided. No practical application to SPHAM.	Hydrogen - Gaseous - Liquid - Solid	NI	NI	Quantitative	Joe Mangino	RL *
248	8	Characteristics of Liquid Rocket Propellant Explosion Phenomena, E. A. Farber, University of Florida, Technical Paper No. 396, November 1967	This paper presents the results of a thermocouple grid analysis of two 25,000 pound LOX/RP liquid propellant explosion experiment	Liquid oxygen RP Fuel	Rupture of barrier between oxidizer and fuel tanks	Blast	Quantitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
249	9	Study of Detonation Induction In Solid Propellants By Liquid Propellant Explosions, Report # 0797-01(01)FP/April 1965, Downey Plant D. R. Irwin et. al	Because some launch vehicles employ both solid and liquid stages tests were performed to determine the following: 1) determine whether conditions of vigorous mixing, the hypergolic mixture of N_2O_4 Aerozine -50 can be initiated to detonation. 2) measure the detonation velocity and detonation pressure as a function of charge diameter for LOX/RP-1 and LOX/LH_2 3) demonstrate if the detonation of liquid propellants can initiate detonation of adjacent solid propellant.	Liquid N_2O_4/ Aerozine LOX/RP-1 LOX/LH_2 Solid Propellant		Blast Thermal	Quantitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION						REV BY
			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	
250	10	Investigation of S-IV All Systems Vehicle Explosion, April 1964, NASA TMX-53039 J. B. Gayle	Presents qualitative and quantitative results of S-IV explosion pertaining to fireball (size duration), fragments (size and distance) and explosive yield.	Liquid LOX/LH_2	Overpressure of LOX container	Blast Thermal Fragmentation	Quantitative Qualitative	Rogers/ Ullian	GB *
251	10	Space Distances and Shielding for Prevention of Propagation of Detonation By Fragment Impact Dec., 1960, AD321962 R. M. Rindner N. S. Wachtell	Relationships are presented which permit the calculation of safe distance for prevention of propagation of detonation due to fragement impact between adjacent potentially mass-detonating systems for any assumed degree of risk and degree of acceptor shielding.	TNT (Can use scaling laws)	NI	Fragmentation	Quantitative	DTIC	GB *
252	2	Register of Hydrogen Technology Experts, Oct. 1975, N76-10323 R. D. Ludtke	Names of individuals who are experts in various fields of technology related to hydrogen, and their principal area of expertise.	Hydrogen	NA	NA	Qualitative	Joe Mangino	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
253	2	On the Transition of Burning of Explosives to Detonation, AD634307 K. K. Andreyev	Characteristics of the acceleration of the burning of explosives to detonation	Powered Explosives Liquid Explosives	Detonation	Blast Thermal	Qualitative Quantitative	DTIC	EK *
254	2	Attenuation of Air Shock Waves In Tunnels, June 1960, AD241878 (Lim.) R. O. Clark, G. A. Coulter	Discussion of the attenuation of peaked shock waves as a function of travel distance along smooth walled tunnels.	NI	NI	NI	Quantitative	DTIC	EK *
255	4	The Biodynamics of Airblast, July 1971, AD734208 C. S. White, et. al	Basically identifys possible injuries from Airblast.	NI	NI	Blast Thermal	Quantitative Qualitative	DTIC	GB *
256	4	An Experimental Program to Determine The Sensitivity of Explosive Material to Impact By Regular Fragments, Dec. 1965, AD477875 D. G. McLean, et. al	Establishes sensitivity of explosives to the impact of regular steel fragments, to predict bounary velocity for safety purpose.	Cased and uncased charges of pentolite and cyclotol explosives	Impact Velocity	Blast	Quantitative Qualitative	DTIC	EK *
257	3	Shock Induced Sympathetic Detonation In Solid Explosive Charges, June 1961, AD637251	Investifation of the transition from initiation to high order detonation, with dir-	Pentolite charge explosive	Detonation	Blast	Qualitative Quantitative	DTIC	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
257	Cont	M. Sultanoff	ect observation of the core reaction in the receptor.						
258	5	Explosion Hazards of Mixed Hydrazines Fuel, AD617531 H. P. Moran	Examination of MHF type fuels for their tendancy to explode when heater or burned Foreign materials were also introduced.	Liquid Mixed proportions of hydrazine, monomethyl hydrozine and hydrazine nitrate	Explosion Buring	Blast Thermal	Qualitative Quantitative	DTIC	EK *
259	10	Fragment Weight Distributions from Naturally Fragmenting Cylinders Loaded with Various Explosives, Oct., 1973, AD-772-480 H. M. Sternberg	Presents a fragment formula for the rapid calculation of fragment directions, velocities, and weight distributions from shells, bombs, and warheads.	Ammonium Nitrate-TNT (Pressed) and 25 other materials not identified in SOW	Detonation	Fragmentation	Quantitative	DTIC	GB *
260	9	Maximum TNT Equivalence of Naval Propellants, Feb., 1983, ADB075867 M. Swisdak Jr.	Determines the TNT equivalency of propellant by measurement of airblast	Solid Ammonium Perchlorate/alumium/Rubbey Binder Nitrocellulose/Nitcoglycerin/Binder/	NA	Blast (Pressure and impulse)	Quantitative (Results) Qualitative	DTIC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
261	7	The Development of Damage Indexes to Structures Due to Liquid Propellant Explosions, Phase I - Feasibility Study, Apr., 1966, AD481497 (Limited) M. J. Rosenfield	Phase I of Development of damage indexes to fire explosion in the vicinity of the launch pad. As developed by project pyro Phase II is #413	Liquid Propellant	Prelaunch & Launch Lift-off	Fire Blast Wave	Quantitative	DTIC	KKM *
262	10	The Development of Damage Indexes to Structures Due to Liquid Propellant Explosions, Phase II - Damage Index, Nov., 1968, AD845513	Goals of this project is to develop damage indexes for structure near the explosion of a liquid-fueled rocket poised on the launch pad. Consideration in determining these indexes is given to the size of the rocket, the size and type of structure and the distance of this structure from the rocket.	Liquid LO_2/RP-1 LO_2/LH_2 N_2O_4/ N_2H_4 + UDMH	Explosion at launch pad or at lift off	Blast (Pressure Impulse)	Quantitative	DTIC	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
263	9	Calculations of Fragment Velocities From Naturally Fragmenting Munitions, Memo. Report # 2509, July 1975, ADB007-377 R. R. Karpp & W. W. Predebon	Calculation techniques for predicting warhead performance based upon the fragment velocity distribution are presented and compared with experimental data.	Munitions	Explosion	Fragmentation	Quantitative	DTIC	GB *
264	7	The Unified Theory of Explosions an assessment Aug., 1976, AD-A032406 E. Stromsoe	Offer "simple analytic descriptions and methods for analyses of blast, absolute hydro-dynamic yield and the damage potential of explosions in general"	NI	NI	Blast	Quantitative	DTIC	GB *
265	9	A Preliminary Investigation of Analytic Debris Estimation Procedure Final Report, Feb., 1970 AD874061 J. F. Melichar	Basically provides methcdology for predicting damage to structures induced by Airblast.	NI	Explosion	Blast Thermal	Quantitative Qualitative	DTIC	GB *
266	9	Burning to Detonation Transition in Porous Beds of A High Energy Propellant, AD-AD80-315 R. R. Bernecher, et. al	A study of the deflagration to detonation transition behavior od porous charges of a high-	Solid Shredded Porous Propellant Ignitor	Explosion (High Confinement Steel Tube)	Blast Thermal	Quantitative Qualitative	DTIC	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0643G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
266	Cont		energy propellant	(High-Energy)	Explosion (Low Confinement Plastic Tube)				
267	10	Debris Hazards, A Fundamental Study, Oct., 1963, AD342723 E. B. Ahlers	Collection and analysis of data on various aspects of debris formation and dispersion, eased on several hundred high energy incidents.	Nuclear or High-Energy Explosives	Blast Induced Flight of Structural Fragments. Pickup of Material by blast winds and crater throwout. Hazards to personnel, utilities and eqiupment.	Blast Fragmentation	Qualitative Quantitative	DTIC	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
268	9	Damage To Structures By Fragments and Blast, Aug., 1971, AD728868 J. G. Greenspon	This report presents methodology for computing damage from fragment-blast weapons. Calculations are based on an assumed mode of failure after fragment damage takes place.	NI	Buckling of wings and fuslkages	Blast Fragmentation	Quantitative Qualitative	DTIC	GB *
269	7	The Blast Environment: Methodology and Instrumentation Techniques with Applications to New Facilities, Aug., 1965, AD622980 E. A. Zeitlin	Report presents blast effects on structures and provides quantitative methods for measurement.	NI	Structures Failure	Blast	Quantitative	DTIC	RL *
270	8	Explosion Phenomena Intermediate Between Deflagration and Detonation, 1967, AD662778 J. A. Brown & M. Collins	The program was aimed at providing an under standing of Intermediate Explosions, also present some example of actual Intermediate Explosions.	NI	NI	Blast (Detonation Deflagration)	Quantitative	DTIC	GB *
271	9	Offshore Oil Hazards for STS Assuming a 14% TNT Yield, C-84-160, Wiggins CO., Feb. 1984 J. B. Baeker	To eva'uate the launch hazards to potential offshore oil facilities in order to provide WSMC with	NI	Impact	Blast Fragmentation	Qualitative Quantitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
271	Cont		information required in establishing their position on evacuation, sheltering or exclusion. Vehicles addressed STS, Titan 34D, Titan IIIB Attas H and MX.						
272	8	Technical Support Package on Equations for Composite Propellant Burning, JPL #30-4834/ NPO-15324, 1982 L. C. Stehald and N. S. Cohn	Objective is to predict the buring rate characteristics of composite propellants at high pressure.	Solid Ammonium Perchlorate binder and aluminum	NA	Thermal	Quantitative	Rogers/ Ullian	GB *
273	5	Inter-Relationship of Explosive Characteristics 111, NAVORD Report 4510. April 1957 Donna Price	To extend current knowledge of explosive behavior and thus improve the ability to predict behavior (detonation velocity, shaped charge and fragmentation.)	High Explosives	NA	Blast Fragmentation	Qualitative Quantitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Fertinence of data (Preliminary Evaluation)

ID: 0001W AD: 0843G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
274	1	Shock Sensitivity of Damaged Propellants, Los Alamos, RL. NO 123173, August 1981 R. C. Rabie, B. C. Craig	Workshop report suggesting models of crack propagation, shock-to-detonation transition, and porous bed compaction for experimentation.	Solid Propellants	NA	NA	Qualitative	In-House	EK *
275	10	Report on Giant Patriot Missile Destruct Test, AD880714, Feb., 1971 AVCO Government Product Group	Report of a ground test on a full-scale Minuteman II missile to confirm that the destruct system would render all stages non propulsive. Data was generated on detonation rate, missile debris dispersion, and TNT equivalency.	Shaped Charge Destruct Ordnance Solid Propellants	Explosion Detonation Debris Dispersion Overpressures	Blast Fragmentation Thermal	Qualitative Quantitative	DTIC	EK *
276	8	Explosive Phenomena Charts for Missile Safery Determination, AS808806, Feb., 1967 (Limited) C. R. Eshelman	A graphical presentation of AFM 127-100C quantity distance criteria, to assist in evaluation of hazards and safety requirements in the event of missile propellant explosion.	Various liquid and solid propellant combinations	NA	Blast Fragmentation Fire	Quantitative	Rogers/ Ullian	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
277	10	Quick Look Reports, for Aerojet - General Corporation Minuteman, June 1960 Larry Letrow	Reports include: Cold destruct tests Detonation sensitivity tests Destruct system tests Detonation interaction test	Ordnance destruct device RDX explosive Solid Propellants	Detonation Stage engine propulsive after destruct Fire, fragmentation and air blast over pressure hazards	Blast Fragmentation Thermal	Qualitative Quantitative	Rogers/ Ullian	EK *
278	7	Seismic Disturbances Generated By Titan III 624A Solid Motor Sled Test, October, 1964 W. V. Mickey T. R. Shugart U. S. Dept. of Commerce Washington DC 20230	Study damage potential of a propulsive solid motor under dynamic impact. Record induced seismic energy, earth motion, and propellant-TNT equivalence.	Solid propellant	Stage Engine propulsive after dynamic impact	Blast Fragmentation Thermal Acoustics	Qualitative Quantitative	Rogers/ Ullian	EK *
279	1	DDT Test Sensitivity Improvement, AFRPL-TR-82-043, July 1981	A program to improve the sensitivity of the dynamic shear/ deflagration to detonation transition testing, in order to	Solid Propellant Class 1.1 Solid Propellant Class	NA	NA	Qualitative	Rogers/ Ullian	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
279	Cont		screen candidate propellants for their DDT susceptibility	1.3					
280	10	Effects of Blast Overpressure on Humans and Buildings, Technical Report No. 83-1453, Wiggins, CO. Nov. 1983 J. Baeker, J. Haber, L. Philipson	This report presents the effects of blast induced overpressure on humans and structures (windows & buildings, also identifies candidate methodologies and empirical data for the evaluation of blast overpressure hazards.	NI	Explosion	Blast Fragmentation Acoustics	Quantitative Qualitative	Mr. Riehl	GB *
281	10	Space Shuttle Range Safety Hazards Analysis, Technical Report No. 81-1329, July 1981, Wiggins, CO. J. B. Baeker	Potential failure Scenaries and their probability of occurence were evaluated from solid Rocket Motor Ignition command to main engine cut-off. Casualties and damage were also est. for impacting frag.	Liquid LOX/LH_2 SRB Solid	Tipover at pad # SRB Case or nozzle failure. Loss of main Eng. Refer to Document for all failures	Blast Fragmentation	Quantitative Qualitative	Mr. Riehl	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
282	10	Evaluation of Fragment Models and Analysis of Solid Propellant Response to STS Launch Accidents, AFWL-TR-81-150, Jan. 1982 (Limited)	This report evaluates methodologies which define the fragment environment caused by an explosion of the space shuttle's external tank (ET) at the Multihundred Watt Radioisotopic Thermoelectric Generator (MHW-RTG).	Solid Propellant LOX/LH_2	Impact	Blast (yield overpressure) Fragmentation (Distribution size and velocity)	Quantitative Qualitative	Mr. Riehl	GB *
283	10	Response of the GPHS-RTG To Explosive Environments, AFWL-TR-82-140, May 1983 F. Schumann, W.D. Owings D. C. Anderson	This report describes modeling and analyses performed to evaluate the response of a space nuclear power system to accidents involving an explosion of the space transportation system's (STS's) external tank. The power generator is the radiosotope thermoelectric generator (RTG) designed for use in Galilec and International Solar Polar (ISP) missions.	Solid Propellants LH_2 LO_2	Buckling of the cylindrical shell	Blast (overpressure, thermal fireball) Fragmentation	Quantitative Qualitative	Mr. Riehl	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795C

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
284	8	Summary of Interagency Nuclear Safety Review Panel Meeting to Review the General Purpose Heat Source Preliminary Safety Analysis Report - Galileo and International Solar Polar Missions, Nov. 1980 INSRP-GPHS-01, May 1981 INSRP	Some of the viewgraphs have good information on thermal environment for launch pad abort and overpressure results for impact.	Liquid LOX/LH_2 RTG	Impact Launch abort Tip over Tank failure	Blast Thermal	Quantitative Qualitative	Mr. Riehl	GB *
285	2	Detonation Propagation, AD323055, Thiokol, March 1961	A study of detonation propagation in monopropellants to permit safe usage in rocket engines.	Liquid Propellants (nitromethane-acetonitrile monopropellants)	Detonation	Blast	Qualitative Quantitative	DTIC	EK *
286	4	A Study of Energy Release in Rocket Propellants By a Projectile Impact Method, NASA CR-69926, Dec. 1969 A. Macek	A study of the rate of energy release in solid propellants by means of shock waves of known durations and known constant amplitudes, to determine explosion thresholds.	Solid Propellants (Based on ammonium perchlorate)	Shock Impact	Blast	Quantitative Qualitative	NASA	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV
287	5	Initiation of Solid Explosive By Impact, N71-22534, 1968 G. T. Afanas'ev & V. K. Bobolev	Quantitative theory of the sensitivity of solid explosives to mechanical action, with results of investigations.	Solid Propellants (TNT, RDX, PETN, Tetrazene, lead azide, mercury fulminate)	Shock Impact	Blast	Quantitative	NASA	EK *
288	10	Identified Shuttle/ Centaur Airborne Hazards for the Radioisotope Thermoelectric Generator Safety Study, # GDC-SP-83-005 March 1983 B. U. Batand	Describe the airborne hazards associated with the Shuttle/ Centaur system, and reveal the methods used to eliminate or control. Good reference for failure scenarios.	Liquid LO_2 LH_2 GH_2	Documents lists, hazards relating to cryogenics electrical hydraulics	NI	Qualitative	Mr. Riehl	GB 0
289	5	Fragment Environments for Postulated Accidents of the Orbiting Vehicle, TES-15018-04, Oct 1980, Teledyne D. C. Anderson, et. al	1) Develops a model for fragment environments from exploding liquid propellent tanks and applies it to potential STS accident scenarios.	Liquid LOX/LH_2 SRM MMH Helium	LOX tank overpressure ET collapse from SRB Failure	Fragmentation	Quantitative	Mr. Riehl	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
289	Cont		2) Considers potential accidents for the Orbiter vehicle and includes a model for fragments from pressure vessel over-pressurization.		ET residual auto-ignition from RDD destruct				
			2) Considers potential accidents for the Orbiter vehicle and includes a model for fragments from pressure vessel over-pressurization.		Intertank collapse on TPS failure In-flight				
					SSME Push-over (No SRB ignition				
					STS curve-over flight and 8 - high speed impact; with and without destruct				

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
290	8	Summary of Susan Explosive Tests Conducted at the Navel Weapons Laboratory Between August 1961 and March 1970, NWL Technical Report TR-2837, Sept. 1972	Presents technique for assessing the sensitivity of explosives to shock and crushing impact. Documented in this report are the results of NWL tests of 129 explosive formulations evaluated for various sponsors.	Explosives - aluminum - ammonium chlorate - RDX	Shock Impact Velocity	Blast (over pressure)	Quantitative Qualitative	DTIC	GB
291	5	Hazards of Chemical Rockets and Propellants Handbook Vol. 1 General Safety Engineering, CPIA 1984, June 1983, ITT Research Institute	Gives methodology for predicting blast, thermal and fragmentation environments resulting from explosions.	Solid Propellant Liquid MMH UDNH LOX/LH_2	Explosion in air ground and water	Blast Thermal Fragmentation	Quantitative Qualitative	Rogers/ Ullian	GB *
292	8	Safety Tests for Nuclear Power Sources Used in The Space Shuttle, January 1984 M-4:GR-84-1 (Letter) J. D. Jacobson	Assessment of the blast effects that could result from an accidental explosion of the Shuttle propellant pertinent experiments (results) were compared to a computer simulation.	Liquid LOX LH_2 RTG	Tank Rupture	Blast Fragmentation	Quantitative Qualitative	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
293	5	Review of Shuttle/Centaur Failure Probability Estimates for Space Nuclear Mission Applications, AFWL-TR-83-61, Dec. 1983	This report reviews the NASA contractor prepared reports that estimate the probability of vehicle failure for a combined Space Shuttle, Centaur Mission. This review considers methodology utilized, effectiveness of implementation and credibility and usefulness of results	RTG SRM	Refer to Document for complete listing	NI	Quantitative Qualitative	Rogers/ Ullian	GB *
294	10	MX Stage Destruct Tests, AFRPL-TR-82-068, July 1982 S. F. Bridges	Summarizes the result of the MX flight termination ordnance system stage destruct tests, which were conducted on the three solid rocket booster stages.	Class 1.3 Propellant (HTPE) Class 1.1 Propellant (NEPE) RDX Explosive	Detonation	Blast Fragmentation Thermal	Quantitative Qualitative	Rogers/ Ullian	EK *
295	5	Large Solid Propellant Boosters Explosive Hazards Study Program, Jan. 1964 (Proposal for Sophy) Aerojet DO-63444	This as the proposal for Project Sophy.	NA	NA	NA	NA	Rogers/ Ullian	EK *

NA Not Applicable
NI *Not Identified*
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
296	5	Detonability of Solid Propellants II Sensitivity of Some Double Base and Composite Propellants, NAVORD Report 6222, Dec. 1958 A. B. Amster	Determine if temperature and/or porosity of double base and composite propellants can initiate detonation. (Not enough information)	Solid Ammonium Perchlorate	NI	NI	Quantitative	Rogers/ Ullian	GB *
297	7	The Susceptibility of Solid-Composite Propellants to Explosion or Detonation, Report #025301, Aerojet, Jan. 1960 W. H. Andersen	This report summarizes the work conducted in the areas of (1) detonation of solid composite propellants and (2) deflagration of solid composite propellants relating to thermal explosion.	Solid Composite Propellants o Ammonium Perchlorate	NI	Blast Thermal	Quantitative Qualitative	Rogers/ Ullian	GB *
298	3	Test Report for MX System Test T-18 Stage I Destruct Test, TRW-23945 July 1982 S. E. Morris	The purpose of the test was to demonstrate that the Flight Termination Ordnance System (FTOS) was capable of rendering the Stage I rocket motor nonpropulsive without causing detonation.	NI	NA	NI	NA	Rogers/ Ullian	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
299	10	Air Shock Parameters and Design Criteria for Rocket Explosions, N65-39951, NASA F. V. Bracco	This report contains design charts and recommendations for the calculations of rocket explosion air blast parameters of interest in structural load calculations. Parameters, based on far-field-TNT-equivalencies are defined for the far-, medium, and close-field of rocket explosions.	Liquid LH_2/LO_2 $RP-1/LO_2$ LO_2/NH_3 N_2O_4/UDMH $+N_2H_4$	NI	Blast (shock, Peak Overpressure)	Quantitative	NTIS	GB *
300	7	Preliminary Investigation of Blast Hazards of RP-1/LOX and LH_2/LOX Propellant Combinations, N67-23667, NASA TMX-53240 J. B. Gayle, et. al	This report discusses the current status of information regarding the blast hazards of liquid propellants and presents results obtained from one part of a comprehensive analytical and experimental investigation of this problem.	Liquid RP-1/LOX LH_2/LOX	Spills Tank Rupture	Blast (Over pressure, Impulse, time of arrival)	Quantitative	NTIS	GB *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
301	8	Titan II Weapon System, Review Group Report, Dec. 1980 Ogden Air Logistics Ctr.	Reviews the current state of the Titan II Weapon System, as regards to safety and supportability. Focuses on policies related to preventing and responding to accidents.	Liquid Aerozine 50 Nitrogen Tetroxide	Propellant Spills Explosion Propellant Vapor Propagation	Blast Thermal Toxicity	Quantitative Qualitative	In-House	EK *
302	8	Space Shuttle Range Safety Command Destruct System Analysis and Verification, Phase I, NSWC TR-80-417, March 1981 W. M. Hinckley, et. al	Investigation of destructive mechanism options for SRB breakup and LOX tank destruct. Includes explosion effects and stress analysis of space shuttle during destruct.	Solid Linear-Shaped Charge	Explosion	Blast Thermal Fragmentation	Quantitative Qualitative	NSWC/M. Swisdake	EK *
303	8	Space Shuttle Range Safety Command Destruct System Analysis and Verification, Phase II, NSWC TR-80-417, March 1981 W. M. Hinckley	Analyzes ordnance options for a destruct system to overcome the Phase 1 short-comings, and assure catastrophic breakup of the external liquid propellant tanks.	Solid Linear-shaped charge	Explosion	Blast Fragmentation Thermal	Quantitative Qualitative	NSWC/M. Swisdake	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
304	10	Space Shuttle Range Safety Command Destruct System Analysis and Verification, Phase III, NSWC TR-80-417, March 1981 W. M. Hinckley	Analyzes breakup of cluster components via triplex range safety command destruct system.	Solid Linear-shaped charge	Explosion	Blast Fragmentation Thermal	Quantitative Qualitative	NSWC/M. Swisdake	EK *
305	8	Space Shuttle Data for Nuclear Safety Analyses, Nov. 1983, JSC-16087-R2	Data on shuttle, Centaur G; Galileo and Solar Polar Spacecraft to provide backup for nuclear safety analysis. Several sections have not been completed, but vehicle configuration data is complete. Obtain final Report.	Liquid LO_2 LH_2 MMH N_2 Solid Propellants	All recognized failure modes are listed in tables. Refer to Document	Blast Fragmentation Fire Toxicity Acoustics	Qualitative Quantitative	John Marshall	RL *
306	5	Approved Flight Safety Hazards Reports for TDRS-A	Hazard reports identify potential failure scenarios associated with the TDRSS.	Liquid Hydrazine GN_2 Hydrogen Oxygen	Refer to Document for complete list	NA	Qualitative	In-House	GB

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
307	9	Accident Risk Assessment Report, Inertial Upper Stage (IUS)/Tracking and Data Relay (TDRS), D290-10017-5, 1982	The purpose of this report is to provide a comprehensive assessment of the risk concerning the Tracking and Data Relay satellite (TDRS)/ Inertial Upper Stage (IUS) interface.	NA	Refer to Document for complete list	NA	Qualitative	In-House	GB *
308	10	Weapon System Safety Guidelines Handbook, System Safety Engineering Guidelines, Part III, NAVORD OD 44942	Detailed description of the responsibilities, activities and analytical methods of system safety engineers to detect and eliminate or control hazards in support of a weapon system development program.	Tends toward explosive devices	NI	Blast Fragmentation Fire Toxicity Acoustics	MIL-STD-882	Joe Mangino	RL *
309	2	Close-In Pressure-Time Histories, AD801628, Oct., 1966 (Limited) P. Lieberman	Measurements of pressure-time histories at stations close-in to a 20-ton sphere of TNT.	TNT	Detonation	Blast	Qualitative Quantitative	DTIC	EK
310	6	Sensitivity of Solid Propellants to Impact, AD816625, April 1967 (Limited)	Determines potential hazards of propellant subjected to a long duration but low-pre-	Solid Propellant (composite PBAN)	Explosion from accidental impact of	Blast	Qualitative Quantitative	DTIC	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
310	Cont	H. S. Napadensky	ssure loading, and minimum impact required for ignition.		a rocket motor				
311	5	The Explosive Properties of Ammonium Perchlorate, AD849090, March 1969 (Limited) M. A. Cook & L. C. Udey	Tests of the hazards of AP so as to formulate recommendations for safe storage and handling of large quantities.	Solid Propellant (ammonium perchlorate)	Impact Detonation Shock	Blast Fragmentation	Qualitative Quantitative	DTIC	EK *
312	10	DSP Payload Annex to Integrated Accident Risk Assessment Report for T34D/TS, MCR-82-071, Dec., 1983 T. Cooper	Assessment of the accident risk associated with the DSP Payload being integrated to the T34D/TS launch vehicle at LC-40 (CCAFS).	Liquid Hydrazine	N_2H_4 leakage & spillage	Blast Thermal Toxicity	Quantitative Qualitative	In-House	EK *
313	5	Space Shuttle Solid Rocket Motor Destruct Calculations, Jerry M. Ward, Naval Surface Weapons Center, White Oak Lab, No Date	Viewgraph presentation of SRB destruct effects.	Solid Propellant	Destruct action	Fragmentation	Quantitative	DTIC	RL
314	5	Space Shuttle Breakup Analysis, William Hinckley, November 1979	Viewgraph presentation depicting fragment pattern and blast wave resulting	Liquid Oxygen Liquid Hyd-	Destruct action	Fragmentation Blast	Quantitative	DTIC	RL

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
314	Cont		from SRB destruct and or ET destruct.	rogen Solid Propellant					
315	7	Development of Hazards Classification Data on Propellants and Explosives, ARLCD-CF-78035, Nov., 1978 H. S. Napadensky, et. al	A study to develop a procedure for the hazards classification of in-process materials.	Single Base Propellant Double Base Propellant Triple Base Propellant Explosive (RDX)	Friction Impact Electrostatic discharge Thermal heating Adiabatic Compression	Blast Thermal	Quantitative Qualitative	K. Morrison	EK *
316	6	Recommended Hazard Classification Procedure for In-Process Propellants and Explosive Material, ADLCD-CR-80025, Sept, 1980 H. S. Napadensky, et. al	Describes a procedure for characterizing hazards imposed by chemical mixtures which exist in propellant and explosive manufacturing operations. Classifications include sensitivity evaluation and effects evaluation.	NI	Impact Friction Electrostatic discharge Thermal heating	Blast Thermal	Quantitative Qualitative	K. Morrison	EK *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
317	1	Manual of Analytical Methods - Ammonia, 79-141, Oct., 1979 No Author	Process for collecting particulate ammonium salts from sulfuric acid-treated silica gel.	Ammonia	NA	NA	Quantitative	K. Morrison	EK *
318	10	Updated Safety Analysis Report for the Galileo mission and the International Solar-Polar mission GE Doc # GESP-7186 April, 1984 General Electric Adv. Energy Program Dept.	Analysis and results of the interim evaluation of the nuclear safety potential of the GPHS-RTG as employed in the Galileo and Solar-Polar missions.	RTG Liquid Propellants LO_2/LH_2 Solid Propellants	Prelaunch Launch Early ascent Final ascent Orbit Injection Loss of SSME thrust	Explosion Fragmentation	Quantitative	John Marshall	KKM *
319	10	Fundamentals of Fire and Explosion Hazards Evaluation AICHE Today Series 1983 C. Grelecki	Collection of description and methodology of various fuels and their sensitivity to explosion.	Liquid Propellants H_2O_2 & Nitroglyse-	Detonation in condensed phase velocity impact	Blast Thermal Dust Explosion Shock Wave	Quantitative Qualitative	W. A. Riehl	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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319	Cont	Hazard Res. Corp. Danville, NJ		rin Solid Propellants AP, RDX	Gas phase explosion Deflagration				
320	8	A Computer Program for the Reduction of Fragmentation Data AD843035 October 1968 R. D. Webster	FORTRAN computer program simulating the fragmentation environment.	NA	NA	Fragmentation	Quantitative	DTIC	KKM *
321	7-8	1981 JANNAF Safety and Environmental Protection Subcommittee Meeting KSC, Nov. 1981 Research Library RL No. 123174 (No. 1) John A. E. Hannum Naval Plant Rep. Naval Plant Rep. Office Laurel, MD 20707	Progress reports on safety problems including solid rocket exhaust particulate, toxic vapor detector development, Titan II accident and environmental impact of the space Shuttle. 1. Solid propellant and explosive safety. 2. Liquid propellant safety. 3. Hazardous meterial management. 4. Safety & hazards of STS.	Liquid N_2O_4 RJ-4, RJ-5, RJ-10 MCH, JP-4 HZ, MMH, UDMH	In-flight Propellant leak explosion in silo Environmental effects	Air Blast Radiotoxic (Nuclear Blast) Chemical Toxicity (ingestion U^{235}) HE Detonation Thermal effects Acoustic	Quantitative Qualitative	In-House	KKM *

NA Not Applicable
NI Not Identified
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322	5-6	1982 Propulsion Systems Hazards Subcommittee Meeting; Vol. 1 April 1982 Naval Weapons Center China Lakes, CA CPIA Publication 356	Hazard test analysis for Navy high performance/special purpose rocket propellants (liquid). Liquid propellant hazard classification.	AP/PBAA Solid Propellants Two phase solid & gas propellants	NA (Mathematical Model developed) Explosion due to weak shock	Pressure loading Convection combustion Fragmentation	Quantitative Qualitative (Kinetic correlations)	In-House	KKM *
323	2-3	AIAA 81-14118 The Mark-II Propulsion Model Journal of Spacecraft and Rockets Vol. 19 #5 J. F. Haley, Jr.	Mark-II description and operations.	Hydrazine Liquid	NA	NA	Quantitative	In-House	KKM *
324	7-8	E220-2106 Explosibility of Explosive and Propellant Dust ARLCD-TR-79045 July 1980 W. T. Moore, et. al	Dust-conc. sampling was to assess the explosion hazard during loading operation	M1 Propellant RDX, HMX, and Ball Powder, Freon	Loading Operation	Blast Detonation Deflagration	Quantitative	K. Morrison	KKM *
325	4-5	Hazards Classification Testing of Ammonium Perchlorate (AP) No ID # J. P. Caltagirone	Experimental analysis on AP to det. explosive behavior. No explosive hazard was exhibited.	AP	None noted with nominal particle size 200 mg.	Thermal	Quantitative	NSWC	KKM *

NA Not Applicable
NI Not Identified
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326	6	The Development of Damage Indexes to Structures Due to Liquid Propellant Explosions, Phase I - Feasibility Study, Apr., 1966, AD481497 (Limited) W. J. Rosenfield	The report describes the collection and evaluation of existing data regarding the overpressure and impulse characteristics of shock or blast waves from liquid propellant explosions and the development of damage factors due to these waves.	Liquid Propellants & High Explosive Cryogenic fuels	Before launch In-flight	Blast Overpressure Shock	Quantitative	DTIC	KKM *
327	3-4	Safety Hazard Criteria for STS Vehicle Launches Report # TOR-0059(6770-04)-11, May 1971 O. A. Refling & R. C. Vaga	Individual and total casualty expectation for STS launch scenarios. In comparison to common casualities. p. 20	NA	In-flight and Re-entry space debris	NA	Quantitative	Joe Mangino	KKM *
328	8	Methods of explosive sensitivity technology for very insensitive explosives and propellants IRECO proposal # 62-110A April 15, 1962 Intermountain Research and Engineering	Determined the crit. diameters for explosives providing a range of detonation pressures.	Solid AP Ammonium nitrate (AN)	Hi velocity impacts	Blast Deflagration Detonation	Quantitative Qualitative (modeling)	Rogers/ Ullian	KKM *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
329	5-6	User's Manual for the Prevention Control, Removal Detoxification and Disposition of Hazardous Material Spills, Vol. 1, April 1975 Martin Marietta J. L. Simpson, F. J. Conn III	Technical proposal for the 1975 Manual with the same title.	Liquid LO_2/LH_2 N_2H_4 A-50	None specified	NA	NA	In-House	KKM *
330	3	Facilities Handbook for Explosive Safe Area 60 June 28, 1979 K-STSM-14.1.6 D. C. Zimmerman	Summary of PPF data for KSC launch sites safety equipment discussed.	GN_2	NA	NA	NA	NA	KKM *
331	6	A Mathematical Model for defining explosive yield and mixing probabilities of liquid propellants (Engineering progress at U of Florida) No. 346 March, 1966 V. 20 N.3 E. A. Farber	Describes how a mathematical model is used to predict the explosive yield from liquid propellants.	Liquid Propellants	No failure scenario described but probability of failure given	NA	Qualitative Mathematical model.	Rogers and Ullian	KKM *
332	6	A Systematic Approach for the Analytical Analysis and Prediction of the Yield From Liquid Propellant Explosions No. 347 March, 1966 V. 20 N.3 E. A. Farber & J. H. Deese	Systematic approach by which the expected yield from liquid propellants can be predicted.	Liquid Propellants LO_2/LH_2 RP-1	"Fireball" detonation	NA	Quantitative		KKM *

NA Not Applicable
NI Not Identified
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333	8-9	Liquid Propellant Safety Handbook KSC GP359 August, 1968	Handbook on safety requirements, chemical/physical properties and hazard analysis on fire and explosion for liquid propellants and cryogen.	Liquid Propellants Cryogens	NA	Fire Blast Toxicity	Quantitative	Rogers/ Ullian	KKM *
334	8	Activities of the Launch Abort Subpanel for the Gailieo Mission Nov. 1982 L. J. Ullian	Galileo and Solar-Polar mission, Centaur destruction and failure analysis. Hazards addressed with STS SRM & ET. Hazardous assessment using Centaur stage in lieu of IUS Stage on Galileo and Solar-Polar missions.	Liquid Propellants Explosion LH_2/LO_2 HE Hydrazine	Tip-over on the pad In-flight Inner tank failure mode SRB Separation failure	Blast Thermal Fragmentation	Quantitative testing	John Marshall	KKM
335	8	RTG Safety Study Galileo International Solar Polar Missions. STS/ Centaur G-Prime Primary and Secondary Failure Response Modes. January 1984, GDC-SSC-83-013	Effects of 10 and 20 Centaur vehicle explosive failure modes WRT the RTG environment in terms of peak overpressures and resultant overpres-	Liquid LH_2 LO_2	LH_2 tank and LO_2 tank leakage, rupture or collapse	Fragmentation Thermal Blast Resultant shock wave overpressure	Quantitative Testing		KKM *

NA Not Applicable
NI Not Identified
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335	Cont	INSRP Meeting 17 Jan. 84	sure probabilities.		I/M Bulkhead reversal failure ET explosion tip-over high velocity ground impact SSME: failure SRB: failure Burn-through				
336	9	The Development of an Explosion Velocity Model Using Debris, #74-3D29 -1, J.H. Wiggins Co., Nov., 1974 J. D. Collins, et. al.	Model for determining the velocity of debris as a result of an explosion.	NA	Explosion	Fragmentation	Quantitative	Lou Ullian Richard Rogers	GB *
337	9	AIAA Launch Operations Meeting No. 70-246, Range Safety of the	This paper discusses missile flight safety (not prelaunch safe-	NI	Impact Flight	Blast Thermal Fragmentation	Qualitative Quantitative	Lou Ullian Richard Rogers	EK

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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337	Cont	Eastern Test Range Feb. 2-4, 1970	ty). It reviews statistical data on missile and space vehicle performance, on destruct system failures and instrumentation reliability.		Termination Failure Explosion				
338	10	624A Solid Propellant Motor Impact Test, Technical Progress Report 381 NOTS TP 3674, Oct., 1964	A test to evaluate hazard of a solid-propellant motor when used as the booster of a Titan III C missile.	Solid Propellant TNT Equivalency	Impact shock	Blast Thermal Fragmentation	Quantitative	Lou Ullian Richard Rogers	EK
339	7	Launch Risk Analysis Program (LARA) Presentation No Author	Analysis of Launch Vehicle In-Flight Risks for most space vehicle and missile systems.	NI	Impact from Malfunction turn Propulsion Casualties	Blast Fragmentation Thermal	Quantitative	Lou Ullian Richard Rogers	EK
340	9	Range Safety Staff Study Missile Vulnerability Study - Program 624A, June 1962, Mathematical Branch, Range Safety Division	Study evaluates the hazards to which a large space vehicle is subjected as a result of launching, or attempting to	Liquid Propellant Solid Propellant	Pad Explosions Explosions Near Pad	Blast Fragmentation Thermal	Quantitative	Lou Ullian Richard Rogers	GB *

NA Not Applicable
NI Not Identified
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340	Cont		launch. Also examines probability that a malfunctioning missile will cause damage to another missile on an adjacent pad.		Missile Fall Back Missile Impact along intended flight line Missile impact away from intended flight line				
341	10	Discussion of Results of 120" Solid Propellant Motor Impact Test By Gerald Couch No ID #	Impact test to determine the hazard potential should such a motor impact the earth at maximum terminal velocity.	Composite Solid Propellant (PBAN-AP-Al)	Impact Velocity	Blast Fragmentation	Quantitative	Lou Ullian Richard Rogers	GB *
342	9	Safety and Design Considerations for Static Test and Launch of Large Space Vehicles, Control No. MT61-11634, June 1961	Defines hazards and degree of hazard, associated with launching of large space vehicles. Develops design data for sit-	Liquid Propellants Solid Propellants	Ref Part II of this document for failure scenarios	Blast Acoustics Fire Fragmentation Radiation Toxicity	Qualitative Quantitative	Lou Ullian Richard Rogers	EK *

NA Not Applicable
NI Not Identified
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			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
342	Cont		ing and safety for launch facilities and personnel.						
343	10	A Manual for the Prediction of Blast and Fragment Loading on Structures, DOE/TIC-11268, Nov, 1980	A manual to provide architect-engineer firms guidance for the prediction of air blast, ground shock, and fragment loadings of structures as a result of accidental explosions in or near these structures.	Liquid Propellants Solid Propellants Ref Appendix A of this document for properties of explosive	Ref chapter 3 of this document for explosives and damage mechanisms	Blast (Air) Cratering and ground shock Fragmentation	Qualitative Quantitative	Lou Ullian Richard Rogers	EK *
344	8	Summary of Bellcomm's Statistical Analysis of the Project Pyro Data, Safety Seminar, June 1969 DE-FSO J. H. Deese, JFK-NASA	A statistical analysis estimating yield and departures from TNT blast behavior, and dependance of yield upon weight and other variables.	LO_2/LH_2 LO_2/RP-1 Liquid	Self Ignition Combined by missile failure Combined by ground surface failure	Blast	Qualitative Quantitative	Lou Ullian Richard Rogers	EK *

NA Not Applicable
NI Not Identified
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345	7	Fire and Explosion Hazards of Flight Vehicle Combustibles, Mar., 1965 AFAPL-TR-65-28 E. Litchfield, H. Perlee	Investigates suppression of cryogenic propellant explosions and flammability limits.	Liquid Hydrogen-Solid Oxygen Liquid Oxygen-Solid Hydrocarbon	Impact Shock Explosion Ignition Detonation	Blast Thermal	Qualitative Quantitative	Rogers/ Ullian	EK *
346	8	Explosive Equivalence of Liquid Oxygen -RP Mixtures to TNT, May 1957, Rocketdyne R-1476 T. S. H. Yee	Determination of TNT equivalency ratio to liquid oxygen-RP mixtures.	LOX-RP mixtures Liquid	Detonation	Blast	Quantitative	Rogers/ Ullian	EK *
347	2	Technical Memo on Explosive Effects Produced By Failure of Launch Vehicles, Feb., 1967 Arthur D. Little, Inc.	Recommends a program to provide additional information on modes of structural failure and the mechanisms that will cause ignition of escaping propellants during launch vehicle accidents	NI	Explosion Structural Failure Propellant Ignition	Blast Thermal Fragmentation	Qualitative	Rogers/ Ullian	EK *
348	5	Large Rocket Motor Hazards, LS81-29, CPIA, Dec., 1981 J. A. E. Hannum	A bibliography involving a literature search for large rocket motor hazards data.	Solid Propellants	Ref this document page iii for principal search terms	NA	Qualitative	Rogers/ Ullian	EK *

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NI Not Identified
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349	3	High Density Solid Rocket Propellant, LS82-13, CPIA, June 1982 T. M. Gilliland	A Bibliography involving a literature search for high density solid rocket propellant data.	Solid Propellants	Ref this document page iii for principal search terms	NA	Qualitative	Rogers/ Ullian	EK *
350	5	Ammonium Nitrate Solid Propellants, LS82-10, CPIA, May 1982 T. M. Gilliland	A bibliography involving a search for ammonium nitrate solid propellant data.	Solid Propellants Ammonium Nitrate	NA	NA	Qualitative	Rogers/ Ullian	EK *
351	9	Determination of Transient Temperatures during the expansion of fireballs. V. W. Klein et. al. Final Report NAS9-4448 MRI, 425 Volker Blvd, Kansas City Missouri 64110 Project No. 2885-E 1965	The development of a transient temperature sensing system to measure the thermal response of a fire ball. Data obtained from large scale experimentation but the report states the data is incomplete for the transient phase of the fireball growth	Aerozine -50/N_2O_4	NA	Fire	Quantitative	R. F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
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			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
352	5	Solid Propellant Critical Impact Velocity and Friability Testing, LS83-17, CPIA, June 1983 J. A. E. Hannum	A bibliography involving a literature search for solid propellant critical impact velocity and friability test data.	Solid Propellants	Ref this document page iii for principal search terms	NA	Qualitative	Rogers/ Ullian	EK *
353	6	Bullet Fragment and Thermal Hazards to Tactical Solid Rocket Motors LS81-31, CPIA, Dec. 1981 J. A. E. Hannum	A bibliography involving a literature search for bullet, fragment, and thermal hazards to tactical solid rocket motors.	Solid Propellants	Ref this document page iii	NA	Qualitative	Rogers/ Ullian	EK *
355	5	Impact Effects on Solid Propellants, AFRPL-TR-76-90, July 1976 G. A. Beale, et. al.	Report presents results of impact testing various solid propellant mixtures by means of .30 CAL, .50 CAL, 20 MM, Soviet 23MM HEI-T Ammunition and a 40-foot drop test.	ANP3146-2 (II) TPH-1123(II) TPH-8226(II) UTP-15908 (VII) FZO (VII) N4 (II) NPCL-11(VII)	Impact Test	Blast Fragmentation	Quantitative	In House RL118197	RL *
356	5	1980 JANNAF Propulsion Systems Hazards Subcommittee Meeting, Vol. 1, RL20717 (Safety of Liquid Propulsion Systems for STS Payloads). A. J. Tanlis, T. A. Erickson						In House	*

NA Not Applicable
NI Not Identified
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356	9 Cont	1 Fluorine/Hydrazine Propulsion System, Safety Considerations for STS operations, D.L. Bond, G.W. Kruger L.C. Montgomery.	Provide design and safety constraints data for payload propulsion system on the STS orbiter	Liquid Nitrogen Liquid Fluorine Hydrazine Helium	Cooling loss line rupture insulation loss explosion tank rupture	Blast Fragmentation Fire Toxicity	Qualitative		RL *
	7	2 Safety of Liquid Propulsion Systems for STS Payloads, J.C. Lewis, R.G. Gilroy Shuttle	Provides information on earth-storable liquid propellants and outlines design methods to comply with STS safety design requirements	Liquid Hydrazine, Nitrogen Tetroxide Mercury	NI	NI	Qualitative		RL *
	8	3 Liquid and Solid Propulsion System Safety design criteria for space transportation system payloads, Lt. Col., J. S. Smith Naval Post Graduate, Monterey, CA CPIA Publication 330 Johns Hopkins Univ.-Applied Physics Lab-Laurel, MD	STS Design Criteria interpreted and applied to example subsystems	NI	NI	NI	Qualitative		RL *

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NI Not Identified
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357	9	Titan-III Solid Rocket Motor Explosion Hazard Analysis, AD8213516, Jan., 1966 L. Barker, S. Stanick	Large solid rocket launch evaluated for explosive quantity distance based on propellant composition vehicle type, & environment	Solid Propellant Liquid Propellant	Launch Pad Fire Velocity Impact Prelaunch	Fire Explosion pressure rupture w/o Blast	Quantitative	DTIC	KKM *
358	9	Reaction of Titan-III Solid Rocket Motors in the Event of Liquid Propellant Fire on the Launch Pad, AD9078266, Nov., 1967, G. H. Hasley	Titan-III with SRM evaluation of liquid/solid propellant causing a fireball accident	Liquid Propellant Solid Propellant mixed	On-pad abort missions	Fireball propagation thermal	Analytical Study	DTIC	KKM *
359	8	Liquid and Solid Chem. Rocket Propellants, AD-B0712146, Oct., 1982 Y. M. Paushkin	Presents the use of chemical propellants in rocket engines & discusses the problem of propellant ignition and self-ignition. No failure scenario evaluated	Liquid Propellant (LH_2) Solid Propellant (metal & Boron) propellant for Hybrid Rocket engines Hydrocarbon fuels	NI	NI	Empirical	DTIC	KKM *
360	7	Fundamentals of Liquid Propellant Sensitivity AD818649, Sept., 1966	Describes the explosive sensitivity of liquid propellant	Liquid Propellant Pressurants Argon	NI	Pressure Shock Explosive Decomposition	Quantitative	DTIC	KKM *

NA Not Applicable
NI Not Identified
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361	7	Shuttle Shrapnel Subpanel Marshall Space Flight Center October 1980 No ID # W. A. Riehl	The results from the MSFC Subpanel Study on Shuttle Accidents involving explosion	LH_2/LO_2 Liquid	External Explosion Internal Explosion (over presure)	Fireball Shock Wave Shrapnel Fragmentation	Qualitative (No data given, just results)	W.A. Riehl	KKM *
362	8	Real time debris patterns for Ballistic Missile Launches, Collins, J.D., Jameson M. and Jantz, JL J. of Spacecraft and Rockets Vol. 13 N.5 May 1976, pp. 310-315	A methodology to predict probabilistic boundaries of the dispersion of debris	NI	In-Flight Prelaunch	Debris from Explosion	Quantitative		KKM *
363	8	Blast parameters and other characteristics of N5 Propellant Shepherd, Levmore ARLCD-TR-77023 December, 1977	Determines TNT equivalency and other blast parameter (peak overpressure & positive impluse) for N5 propellant. Also determines N5 propellant for sympathetic deflagration or detonation under several conditions.	N5 Solid propellant	Catastrophic Accident-explosion	Blast, thermal	Empirical	Doug Banning	DWB O

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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364	9	Determination of the TNT Equivalency of a typical Class 1.1 Solid Rocket Propellant (Blast Hazards). T.L. Kinsel AFRPL-TR-80-24 April 1980	Determines the TNT equivalency, overpressure, and impulse for a 6,772 Class 1.1 propellant grain.	Class 1.1 Solid Propellant (MX)	Catastrophic explosion/	Blast	Empirical, Theoretical	Doug Banning	DWB O
365	7	Airblast Performance of an absolete high-energy propellant, VOY-4 P.J. Peckham February, 1978 NSWC/WOL TR 77-52	Airblast parameters (pressure, shock wave impulse), nozzle effects, and growth rate for detonated VOY-4 under various conditions/configurations.	VOY-4 solid propellant, pentolite, H-6, and HMX	Detonation in air	NA	Empirical	Doug Banning	DWB O
366	9	Blast and Fireball Comparision of cryogenic and hypergolic propellants with simulated tankage. NAS-9-2055 Aerojet General Downey, California Report No. 0822-01(02)FP 1964	Eight tests comparing explosive and fireball characteristics of hypergolics and cryogenic propellants	LOX/RP-1 A-50/N_2O_4	Fall-back Rupture	Fire Blast	Quantitative	R. F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
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367 447	6	STS Launch Analysis Lt B. Haswell Space Shuttle No ID #	Describes a computer program that analyzes pushover and tip-over type launch aborts for the Space Shuttle.	NI	Launch Abort	NA	Theoretical	Doug Banning	DWB 0
368	10	Assembly and Analysis of Fragmentation Data for Liquid Propellant Vessels. W.E. Baker, et al NASA-CR-134538	Determination of fragment velocities masses, shapes, and blast effects as a function of type and quantity of fuel, type of accident, and blast yield for several possible accident scenarios. Excellent reference for SPHAM.	Liquid N2H4, A-50, LO2, RP-1	Confined by Missile, Confined by ground	NA	Theoretical using existing empirical data	Doug Banning	DWB 0
369	8	Explosive Hazards of Composite Solid Propellants Billings, Brown IDA Research Paper P-362 March 1968	Paper only partially legible.May have some scavengable information. Need new copy.	Solid PBAN, Polyurethane Double Base TNT, Comp. R	Explosion of solid propellant	NA	Theoretical literature search	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
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ID: 0002W AD: 0795C

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
370	1	Explosion Working Group Minutes of meeting DOE: Germantown, MD January 24, 1984	No useful information	NA	NA	NA	NA	Doug Banning	DWB 0
371	10	Trip report - visits to AFWL, S cubed, and DRI relative to Shuttle Accident Scenarios and direct course. R. W. Englehart NUS Corporation February 1984 NUS 84-006-RWE	Reports blast parameters and accident scenarios for several potential Space Shuttle Launch aborts. Includes TNT equivalency, blast wave overpressures, and % yields.	Liquid Solid LOX/LH2	Catastrophic Accident	Blast, thermal, fragmentation	Empirical/ Theoretical	Doug Banning	DWB 0
372	8	Minutes of the STS Explosion Working Group (EWG) Meeting, Kennedy Space Center U. S. Govt. Memo Feb. 22-23, 1984 DOE F1325-8	Small amount of data on a few Space Shuttle Launch aborts and the resulting explosions.	Liquid LOX/LH2 Solid	Catastrophic Failure	NA	Theoretical	Doug Banning	DWB 0
373	8	Memorandum No. 667 - ON Burst Failures in Rocket Motors RPE - Memo 667 R. C. Parkinson November, 1975	Dynamic burst pressure, burst overpressures, and effects of propellant strength on low temperature motor failures.	Solid Rocket Propellants	Detonation of rocket motor during initiated burn.	Blast	Theoretical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
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374	1	Solid Rocket Booster Mass Properties Status Report No. 21 G.C. Marshall Space Flight Center July, 1975 Shuttle	Defines design parameters for the shuttle SRB/SRM system. Includes mass, structural, and weight properties of the system. Not much useful data for SPHAM.	Solid SRB & SRM	NA	NA	Theoretical	Doug Banning	DWB 0
375	10	High Explosive Equivalent Tests of Rocket Motors F. H. Weals, C.H. Wilson TR-413 NOTS-TP-3910 1965	TNT equivalency, overpressures & impulses for Class 2 and Class 7 solid propellant rocket motors in various configurations, using various initiators.	Solid Propellant Motors, Class 2 & Class 7	Catastrophic Accident	Blast, overpressure	Empirical	Doug Banning	DWB 0
376	8	Further studies of the critical nature of cracks in Solid Propellant Grains. (No cover sheet) FO-4611-70-C-0006	Crack growth, fracture initiation overpressures, and local damage in a solid propellant system.	Crack propagation during solid propellants.	Minuteman III, general solid propellants.	NA	Theoretical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
377	7	Vulnerability of solid propellant motor case W.D. Smith, L.J. Hulsey NSWC-TR-80-280 August, 1980	Determines conditions necessary to cause an operating end-burning solid propellant rocket motor case to catastrophically rupture when hit in the gaseous section with a high speed projectile. Includes assumptions, thermal analysis, and impact dynamics.Not much application to SPHAM.	Solid Rocket Motor	Solid Rocket Motor hit by projectile	Blast	Theoretical/ Empirical	Doug Banning	DWB 0
378	8	Design Aspects of Explosive Mixtures in a Vehicle Interstage D.S. Allen, et al NASA CR-59440 July, 1964	Explosions in a vehicle interstage, effects of explosions on interstage structures, explosion hazards during flight, explosion prevention, and explosion characterization are considered for general vehicle interstages using LH2/LO2 propellants.	Liquid LH2,LO2	Slow leak of LH2/ LO2 into a vehicle	Blast, thermal	Theoretical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
379	2	Initiation mechanisms of solid rocket propellant detonation A.N. Takata, A.H. Wiedermann AFOSR-TR-76-1068 August, 1976	Development of a computer model to describe propellant detonation in a solid rocket motor. Gas dynamics, mechanical response of cracks, and a dynamic burn model are presented.	HMX based solid propellants	NA	NA	Theoretical empirical	Doug Banning	DWB 0
380	2	Propellant/rocket motor hazards - A perspective and review A. Adicoff NWC-TM-4992 June, 1983	Outlines the testing requirements for qualification of rocket motors for fleet use. Includes drop, bullet impact, slow cook-off, fast cook-off, and propagation tests. No real data reported, excellent source of references.	Solid Propellant Motors	NA	NA	Literature Search	Doug Banning	DWB 0
381	8	Memo:Airblast Results from EX-104 Dual Thrust Rocket Motor Hazard Test	Reports overpressures and impulses as a function of distance	EX-104 (700 lb solid propellant)	Bullet Impact	NA	NA	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
381	Cont	P.J. Peckham, Naval Surface Weapons Center June, 1982 NSWC Memo R15	from blast for two tests involving bullet impacts on EX-104 rocket motors.						0
382	3	Full scale, Stage I Minuteman Development Motor Hot Destruct Test T.W. Gould TW-327-9-62 November, 1962	Test of the linear shaped charge distruct subsystem of Minuteman Stage I Motor. Includes detonation velocity of RDX explosvie, and fragmentation distribution/-velocity data.	Minuteman Stage I, RDX Solid	In-flight voluntary abort.	Blast, fragmentation	Empirical	Doug Banning	DWB 0
383	2	Effect of Ammonium Perchlorate on the Burning Rate of Composite Propellants Under Atmospheric Pressure. I. Takeshi et al Journal of Ind. Explosives Soc. Vol.36, No. 6 1975	Nothing Applicable to the SPHAM.	AP-CTPB Composite propellants Solid	NA	NA	NA	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
384	8	Fragment velocities from S-IVB Explosion W.A. Riehl R-Aero-AD-69-13 April, 1979	TNT equivalencies for several diff-erent accident scenarios involving the Space Shuttle. Also describes particle velocity and distribution as a result of an S-IVB explosion.	TNT EQ Solid	Launch Abort	Fragmentation blast	Empirical, Theoretical	Doug Banning	DWB 0
385	7	Failure Diameter of Composite Propellants Part I M.L. Pandow, T.H. Pratt Rohm & Haas Company Special Report S-67	Failure diameters and detonation velocities in active-binder propellants. Derivation of extrapolation method. Determin-ation of detonation velocity vs diameter for solid propellants.	Active-binder solid propellants	Detonation of solid propellant motors.	NA	Empirical/ Theoretical	Doug Banning	DWB 0
386	10	Propellant Hazards Evaluation Final Report R.B. Leining, J.H. Thacker NWC-TP-6309	Manufacture, tensile properties, thermal character-istics, shotgun impact resistance and combustion	Solid propellants QBC, GCV, VTP-15908A SAO-109 KAA-114,	NA	NA	Empirical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
386	Cont	April 1982	quickness (to determine critical impact velocity), and critical diameter (for QBC only) were determined for several solid propellants.	GBP-1					
387	4	"Beauregard" Critical Diameter Tests of Second stage Minuteman Wing VI Motor Propellant L.S. Banas AFRPL-TR-65-5 January, 1965	Determines the explosive classification and critical diameter of M.M. wing VI solid propellant. 1.3 Class	Minuteman Wing VI solid rocket propellant.	Catastrophic accident involving minuteman missile.	NA	Emirical, scaled tests.	Doug Banning	DWB 0
388	6	The Detonability of Solid Composite Propellants:Part I R.F. Chaiken, W.H. Andersen From contracts NOrd-17012 and NOrd-18487	Develops a theoretical model of propagation of detonation in solid propellants. Oxidizer-binder ratio, particle size, confinement and other parameters are considered.	Solid propellants	Detonation, abnormal deflagration of solid propellants	NA	Theoretical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: G002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
389	7	Explosion Working Group: Charter Assignments Space Shuttle No ID #	Impact ignition behavior, shock Hugoniots, Impact properties, and critical impact velocity for single, double, and triple base propellants and several explosives. Also covers an Impact ignition model, vulnerability, and analysis of ignition mechanism.	Solid	Launch Abort	Blast	Theoretical	Doug Banning	DWB 0
390	9	Minutes of Meeting - Space Shuttle Range Safety Command Destruct System - December, 1975 No ID #	Investigates the effects and results of an SRB failure. Includes explosion phenomena, flight characteristics after inadvertent separation of SRB or orbiter, and structural loads imposed on rest of shuttle due to aerodynamic and explosion loads.	Solid	SRB Failure/ explosion	Blast, fragmentation	Theoretical	Doug Banning	DWB 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
391	10	DOE Memo - Formation and organizational meeting of space transportation system (STS) Explosicn Working Group, January, 1984 DOE Memo F1325.8	Investigates push-cver, tipover, and LOX tank over-pressure accidents for the STS.De-scribes scenario, TNT equivalency, and overpressure for each accident. (Good SPHAM re-ference)	Solid Liquid LOX/LH2 TNT EQ	Push-over, tip-over, LOX tank overpress-ure	Blast	Theoretical	Doug Banning	DWB O
392	5	Memo: The TNT equiv-alent weight of the solid propellant used in the SOPHY Tests fcr 60" and 72" Diameter Rocket Motors. N. Holland, J. Peters Memo 241	Determine the TNT equivalency of 60" and 72" diameter solid propellent motors.Derives a TNT cylindrical change-on-the-surface p-d curve.	Solid Propellant TNT-EQ	Detonation	Blast	Theoretical, Empirical	Doug Banning	DWB O
393	9	Detonation Propagation Report RMD 2052-Q1 (Confidential Document)	A quarterly report on detonation prop-erties of monopro-pellants. Includes also a literature search and review of monopropellant accidents	Liquid Monopropel-lants	Detonation	o Blast o Thermal	Qualitative Quantitative	DTIC	EK

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
394	9	Detonation Propagation, Report RMD 2052-Q2 (Confidential Document)	A quarterly report on detonation phenomena in monopropellants. Studies events that propagate detonations, and systems that will quench a detonation	Liquid Monopropellants	Detonation	Blast Thermal	Qualitative Quantitative	DTIC	EK
395	10	Summary of Explosive sensitivity tests at [illegible]L, TR-28 [illegible] Partly classified Confidential Document)	A technical report assessing relative sensitivity of explosives to shock and crushing impact, using the Susan test Technique	Solid propellants Liquid Propellants	Shock Impact	Blast Thermal	Qualitative Quantitative	DTIC	EK
396	7	Supplementary Scaling Studies for Project Pyro. Nov. 1965 Air Force Rocket Pro-Pulsion Laboratory URS Corp.	Proposal Outlining the Scaling studies for predicting the damage or explosive effect	Liquid Propellants $LO_2/RP-1$ LO_2/LH_2 $N_2O_4/50-N_2H_4$ 50-UDMH	During Launch In-Flight Impacts	NI	Empircal	In-House	KKM *
397	9	Heat Transfer Hazards of Liquid Rocket Propellant Explosions February 1969 AFRPL TR-69-89 J. A. Mansfield	Summary of the thermal or heat transfer measurements from the pyro project	Liquid Propellants $LO_2/RP-1$ LO_2/LH_2 $N_2O_4/50-N_2H_3$ 50-UDMH	Launch Pre-Launch	Fireball & Thermal Explosion	Quantitative Summaries		KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
398	6	1/18 Scale Model Missile Test Phase II Hot Rocket Motor Test with Flame Deflector, Channel & Barricades 1959 MT59-91772 Broadview Res Corp No Author	1/18 Model Missile Failure Test Simulating a typical full scale launching site with a flame deflector. Not a good copy	Liquid LOX	Seam Rip	Blast	Quantitative		KKM *
399	7	1/18 Scale Model Missile Test Phase III. Modified Al Launcher Simulator 1959 MT59-91771 Broadview Research Corp.	Preliminary results from the modified Al Launcher Testing gross shielding or channeling effects and predict average explosive yields. Not a good copy	Liquid LOX	Seam Rip Explosion Fall Back	Blast Fireball	Quantitative	W.A. Riehl	KKM *
400	10	Hazards of Chemical Rocket & Propellant Vol. 1 Safety, Health & The Environment CPIA Pub. 394 N00024-83-5301 Johns Hopkins Univ. App. Physics Lab, Laurel, MD ITT Research Inst.	Air Force Manual Evaluating Data of various failure scenarios based on the type of propellant used.	Liquid and Solid Propellants	Pre Launch Launch Post Launch (Air Blast)	Blasts Fireball (Thermal) Overpressure	Empirical	John Marshall	KKM *
401	10	Project SOPHY Solid Propellant Hazards Program Elwell, Irwin & Vail Jr. 1967	Theoretical and Experimental investigation of the hazards associated with large	Solid Propellant	Prelaunch: Shock, Impact, External	Airblast & Fireball Fragmentation Radiation	Quantitative	W.A. Riehl	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
401	Cont	R. Elwell, O. Irwin, R. Vail, Jr. AF04 (611)-10919	solid propellant systems		Fire Post Launch Flight Malfunction				
402	9	Launch Hazards AssessmentProgram, Report on Atlas/Centaur Abort Oct. 1965 SC-RR-65-333 F. D. Kite, et. al.	Information and Analysis concerning the aborted launch of the Surveyor-Mission Atlas-Centaur, March 2, 1965	RP/LO_2 LO_2/LH_2	PAD Abort	Thermal Shock Wave Overpressure Explosion	Quantitative Empirical	John Marshall	KKM *
403	10	STS (Shuttle) Fragmentation Subcommittee MSFC/Riehl AFETR/Ullian Feb. 1980 Fragment Environment Eric Rice	Definition of fragment environment, mean fragment vel., mean fragment size and distribution and fragment flux	Liquid Propellant LO_2/LH_2	Bulkhead rupture On-pad Fall Back	Fragmentation Blast	Quantitative Emperical	Riehl/ Ullian	KKM *
404	10	Blast Phenomena from the breakup of the SRB D. Lehto/J. Fetes Data for Oct. 75 progress report for NASA	Information concerning loading of the ET and evaluation of the detonation process of the SRB.	Liquid LH_2 in ET	Impact on the ET buckled by the lateral thrust of the jet of propellant	Air Blast Fragmentation	Quantitative Emperical	Riehl/ Ullian	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
405	8	Explosion Containment Progress Scaling laws and Material Properties J. White, III Minutes of the 20th Explosive Safety Seminar, 1982	Information on Containment of explosion in portable spherical chambers		TNT Detonation of the MK 634 Mod 0 explosive device container	Blast Some Fragments noise and toxic materials	Quantitative (Impulse Approx)	Explosive Safety Board	KKM *
406	8	Department of Defense: Australian Scientific Service Materials Research Lab. Report 628 Scaling effects in the Natural Fragmentation of AISI 9260 Steel Cylinders. 1975 (Limited) B.E. Walsh/A.J. Bedford	Investigation on the effects of increasing the size of steel exlinder on their fragmentation behavior to evaluate the expansion velocity	TNT equivalent	Detonation of increasing diameter cylinders	Fragmentation	Quantitative	DTIC	KKM *
407	8	Department of Defense: Australian Defense Scientific Service Materials Research Labs. Report 624 Scaling of Fragmentation 1975 A. J. Bedford	Information concerning fragmentation scaling relationships the Mott and Payman Fragment mass distribution analyses are used	TNT	Detonation of an internal hi explosive charge	Fragmentation	Quantitative Mott and Payman plots	DTIC	KKM *
408	6	Solid Rocket Plant Report on Fragmentation Test Program Aerojet	Test Program that evaluated fragmentation effects on a	AEREX S-1 High Explosive	Detonation of an Internal Hi	Fragmentation Air-Blast	Quantitative Testing	W.A. Riehl	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
408	Cont	General Tire Sept. 1972 Report No. 0179-64F Walther, Pate, Eldredge L. C. Walther	solid-propellant motor which was subjected to high-energy internal explosion and unpingement		explosive charge				
409	9-10	Vehicle Malfunction Breakup and Velocities and "Explosive Yields" James Baeker C-81-1831329 1981	Space Shuttle Hardware failure modes were presented and the total failure rates computed	Liquid Propellants LOX/LH_2 Solid Propellant TNT Equivalent	Breakup/ Impact of the vehicle Launch abort RTLS Tip over Inadver. separation at the SRB/ET ET Puncture Loss of ME propulsion See p. 3.10-16	Blast Explosion Fire	Quantitative	W.A. Riehl	KKM *
410	8	Research on Hazard Classification of New Liquid Rocket Propellants. Final Report Vol. I Armed Service	Test results to develop safety and design criteria for storage of N_2O_4, N_2H_4, chlorine	Liquid Propellants	Spills causing fire damage Rupture	Blast Fire	Quantitative	DTIC	KKM *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
410	Cont	Technical Information Agency 1961 AD272025	trifluoride, and plutaboraue. Safe-distance values were established						
411	7-8	Research on Hazard Classification of New Liquid Rocket Propellants Final Report Vol. II 1961 AD272-026	Tests to determine hazards of Titan II Propellant handling Mishaps. (Poor reproduction)	Liquid N_2O_4 & UDMH - Hydrazine	Model Missile & Large Scale Spills	Fire Blast	Quantitative	DTIC	KKM *
412	7	View Graphs from Marshall Space Flight Center RE: Shuttle Accidents 1979 No ID # W. A. Riehl	General overhead viewgraphs giving general results on Shuttle accidents involving explosions	Liquid LO_2/LH_2 LO_2/RP-1	Impact Velocity	Blast Fragmentation	Quantitative	W.A. Riehl	KKM *
413	9	The Development of Damage Indexes to Structures Due to Liquid Propellant Explosion Phase II-Damage Indexes Report No. 4-75 M. J. Rosenfield Dept. of the Army Ohio River Division Laboratories 1968	Develops and presents damage indexes to open-flame structures in the vicinity of the launch pad resulting in an explosion (Project Pyro Included)	Liquid Propellants TNT Equivalence	PreLaunch & During Lift-Off	Fire Explosion Blast Blastwave	Empirical Quantitative	L. Ullian	

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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414	8	Impact of Confined Composite Propellant and Performance of Fragments Explosive Safety Board D. H. Jones U.S. Naval Weapons Lab. (No Date) No ID #	Short paper on fragmentation data involving velocities and drag coefficient curves.	NI	During Flight	Fragmentation	Quantitative Empirical	L. Ullian	KKM *
415	9	Shock Development from Compression Waves Due to Confined Burning in Porous Solid Propellants and Explosive. 1983 D.W. Coyne, P.B. Butler and H. Krier (U of I) 83-04-28-061	Porous Solid Propellants are more sensitive to shock-to-detonation transition	Solid Propellants	NI	Blast	Empirical Quantitative	DTIC	KKM *
416	5	Range Safety Data for Titan 34D/IUS (U) 1981 Space Launch Systems Annex A to Vol. I MCR-79-060 DSAP 12/13	Information contained breakup definition as required for range safety analysis.	Solids	NI	Blast	Empirical Quantitative	In-House	KKM *
417	5	Range Safety Data for Titan 34D/IUS 1981 Space Launch Systems Annex B to Vol. I MCR-79-060 T34D Systems Performance	This Annex contains changes or addition to Vol. I Associated with Flight of a PSCS II/III Payload	Solids	In Flight	Blast	Empirical Quantitative	In-House	KKM *

NA Not Applicable
NI Not Identified
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418	8	AIAA Launch Operations Meeting No. 70-246 Range Safety of the Eastern Test Range 1970 R. M. Montgomery	Statistical Probabilities are discussed on Vehicle distraction and failure rates.	NI	Missile Flight	Blast	Empirical	L. Ullian	KKM *
419	8	Appendix I Fragment Velocity Distribution Data Supplied by NSWC to MMC (No Date) No Author	Statistical data given that will describe fragmentation behavior	Liquid Propellant	NI	Fragmentation	Quantitative	NSWC	KKM *
420	8	Naval Surface Weapons Center: White Oaks Labs Memo: From D. Lehto & Data on Shock Wave Pressure on the ET 1975	Preliminary Estimation of the Shock Wave Pressures on the ET due to explosion of on SRB.	NI	NI	Blast Overpressure	Quantitative	L. Ullian	KKM *
421	8	Marshall Space Flight Center Initial Tabulation of Liquid Propellant Explosions Involving 25,000 lb. plus Bill Riehl Critique and Joe Peters rebuttal 1973 No ID #	Data and critiques presented on liquid propellant systems that may be involved blast explosion	Liquid LO_2/LH_2 LO_2/RP-1 High Pressure Systems	Prelaunch Mixing Problems Premature Ignition	Fire Blast	Quantitative	L. Ullian	KKM *
422	7	Space Shuttle Solid Rocket Motor Linear Shaped Charge/Propellant Interaction Jerry M. Ward 1980	General Description of Minuteman Stage 1 Propellant detonation	SRM Propellant Liquid Propellant	LH_2 Premature Mixing with LO_2 Detonation	Blast	Qualitative	Ward	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
423	7	Giant Patriot: Incremental Velocities of Destruct Debrie Dey & Lawrence TRW Systems Group 71-4342.2-18 1971	Summarization of terminal velocities for use in range hazard analyses of the LRL destruct system for Giant Patriot Flight Altitude Scaling was also evaluated.	NI	NI	NI	Qualitative	L. Ullian	KKM *
424	7	Accidents JSC-16087:RI (Pages 3-4 missing) No Author	Shuttle accidents leading to catastrophic failure and explosion.	Liquid RTG LO_2/LH_2	Rupture of LOX and LH_2 tanks Pushover on launch pad. Tipover on pad.	Blast Overpressure	Qualitative	L. Ullian	KKM *
425	8	Manual of Analytical Method - Ammonia GIDEP E170-1694 1979	Ammonia was absorbed on H_2SO_4 treated silica gel and desorbed with 0.1N H_2SO_4 calibration and standards were presented	Ammonia	NI	NI	Quantitative	L. Ullian	KKM *
426	10	Detonability of Large Solid Rocket Motors L. J. Ullian ESMC 1984 (STS) No ID #	Viewgraphs and data evaluating the probability failure rate of solid propellant detonation	Solid Propellant TNT EQ	Failure to detonation Impact Velocity	Explosive	Empirical	L. Ullian	KKM *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
427	8	LWRHU SAR Status Report E. W. Johnson Monsanto Research Corp. (INSRP Meeting, KSC July 10-12 1984 Lightweight Radio-isotope Heater Unit No ID #	Probability results on release of 232 P_uO_2 from the LWRHU	RTG	Overpres-sure Im-pact vel. Prelaunch and Launch Phase	Fragments	Quantitative	AFWL	KKM *
428	3	Acute Toxicity of In-haled 238 P_uO_2 in Beagle Dogs Oct. 1979 Air Force Weapons Lab. AFWL-TR-69-75 J. F. Park DVM E. B. Howard DVM W. J. Bair Ph.D	Evaluation of hazard and knowledge of tox-icity of inhaled 238 P_u in beagle dogs & compares the data with 239 P_uO_2 studies.	Radiosotope 238 P_uO_2	Inhalation of 238 P_uO_2 and 238 P_u	Toxicity	Quantitative	AFWL	KKM *
429	7	Proceedings of the Third International System Safety Conference, Oct. 17-21, 1977, Stouffer's International Center Inn, Washington, D.C. The System Safety Soc.	Presentation of Sys-tem Safety Technical Papers (Some Aero-space oriented)	NA	NA	NA	Qualitative Quantitative	Joe Mangino	EK *
430	8	Fault Tree Handbook, U.S. Nuclear Regulatory Commission, NUREG-0492, November 1978	A handbook intended to provide NRC and contractor personnel with a set of refer-ence material on	NA	NA	NA	Qualitative Quantitative	In-House R. Lomax	EK O

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
430	Cont		fault tree analysis. Includes probability and statistical concepts						
431	9	Accident Risk Assessment Report for the Shuttle Payload Integration Facility (SPIF), MDTSCO, MDS Y0601S, Nov. 1982 McDonnell Douglas	The report evaluates the accident risk assumed during operation and maintenance of SPIF. It includes a facility integrated OHA & Safety Requirements	Liquid & Gaseous Fuel Liquid & Gaseous Oxidizer GN2 Flammable Liquids Halon 1301	Critical Activities During Processing of Payload Hardware High Pressure Systems Propellant Transfer	NA	Qualitative	In-House	EK *
432	8	Liquid-Propellant Explosions R.F. Fletcher NASA Manned Spacecraft Center 1968 Reprint from Journal of Spacecraft and Rockets Vol. 5 No. 10	Evaluation of hazard limits for liquid propellant explosions using TNT equivalent values for shock wave propagation, flame distance limits for fireball propagation & shrapnel ranges.	Liquid Propellant TNT	Detonation Deflagration	Fire Shockwave	Quantitative Qualitative	R.F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
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REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
433	10	MSC Technical Symposium NASA Manned Spacecraft Center 1965 Gemini (Titan II) Atlas C203630 C1	This report describes the fireball and over pressure blast parameter of Gemini Launch Vehicle. The Atlas fireball was determined to be more hazardous than the Gemini	Liquid 50-50 UDMH-N2H4 N2O4 LOX/RP-1	Pad fall Back	Blast Fireball Formation	Quantitative	R.F. Fletcher	KKM *
434	8	Prevention of and Protection against accidentak explosion of Munitions, Fuels and other hazardous mixtures. Anal of the New York Academy of Science Vol. 152, Art. 1 Edward Cohen 1968 (Fletcher: Liquid Propellant Explosion	This is an extention of the above report #433. Comparison of Liquid Propellants & TNT Shock Waves. Shock strength of liquid propellant explosions can be determined from the early fireball expansion. Vehicles: Atlas/ Centaur 5 and Atlas 96	TNT Liquid LOX/RP-1	Detonation & Deflagration Pad-full back	Overpressure Blast Fireball	Quantitative	R. F. Fletcher	KKM *
435	9	DOD/NASA System Impact Analysis Space Shuttle Explosive Equivalency Study Final Report.	This is the background material to the JSC Data Book. These supporting data	Liquid LO2/LH2 Solid SRM	NI	Explosion Blast	Quantitative	R. F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
435	Cont	Vol. III-Appendixes 1974 Ground Systems and Environment Dept. ATR-74 (7337)-1. Vol. 1	are based on the Space Shuttle, the heat of explosion given and various explosion barriers evaluated.						
436	8	Explosion of Propellants R. F. Fletcher NASA-Manned Spacecraft Center, Houston, TX 1968	This paper gives an empirical approach to determine the upper bound of overpressure on surfaces near the explosion	Liquid LOX/RP-1 N204/A-50	NI	Explosion Blast	Empirical	R. F. Fletcher	KKM *
437	9	Reaction and Expansion of Hypergolic Propellants in a Vacuum. Simmons, Gift, Spurlock, and Fletcher 1968	This paper characterized the expansion of hypergolic propellant into a vacuum and determine the occurance of any chemical reaction. The application of these results is to evaluate the explosion hazard in storage and handling these propellants in space	Liquid Hypergolics N204/A-50	Spills	Blast Thermal (Fire)	Quantitative	R. F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
438	5	Vaporization of radic- isotope fuels in launch vehicle abort fires. D. C. Williams, 1971 SC-RR-71 0118 Saturn V Titan III SC-RR-71-0118	Models were developed for estimating the extent to which radio isotopic fuel forms will vaporize if re- leased to the fires resulting from the catastrophic abort of rocket booster Vehicles.	Liquid Solid Pro- pellant 238 PuO2 RTG	Pad Aborts Lift-Off Aborts	Fire	Experimental Quantitative	In-House	KKM *
439	9	System Safety Handbook for the acquisition manager, SD-GR-10, prepared by the Aero- space Corp. for SD/ AFSC California, 10 DEC 1982 R. E. Olson	Informs the program office safety manager about system safety and mishap. Scope in- cludes planning; acq- uisition; program implementation; fund- amentals of hazard analysis/ nuclear, explosive, and bio- medical safety re- quirements; and as- cent and reentry safety. Also inclu- des reference policy documernts and check- lists.	NA	NA	NA	Qualitative	In-House R. Lomax	EK 0

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION: DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
440	9	AFSC Design Handbook, DH 1-6, System Safety HDQ, Andrews AFB, December 1978.	Provides system designers with system safety design principles, criteria, and guidance in technical areas of Air Force Systems. Subjects include facilities, ground equipment, and aerospace vehicle safety design, and life support.	NA	NA	NA	Qualitative	In-House R. Lomax	EK 0
441	10	Chemical Rocket/Propellant Hazard Vol. 1 General Safety Engineering Design Criteria (Jannaf Hazards Working Group Oct. 1971) Saturn S-IV and Titan I AD-889763L	This volume represents a set of guidelines for use in the chemical propulsion industry. The basic philosophy of medical and environmental factors explained from the point of view of the health officer or professional toxicologist. This volume includes liquid propellant explosive equivalents, pyro-SOPHY fireball data and noise criteria	Liquids and Solids	Fall back/ Prelaunch	Blast Overpressure Fire Acoustics	Quantitative	DTIC	KKM

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
442	9	Study of Phenomena Associated with Mixing of Cryogenic Fluids Final Summary Report 0808-10(16)FP Downey Plant Research Div Aerojet-General Corporation Feb. 1965 H. Perkins, MSFC NAS8-11063	A study of mixing cryogenic fluid was completed to determine the factors controlling the assessment of explosive hazards. Analysis of high-speed pictures showed relative densities of the material upon the mixing characteristics and the order of mixing.	LO2/RP-1 LO2/LH2 LN2	NA LH2 spills in LO2 & LO2 spills in LH2 systems	NA Mixing Temperature profile	Qualitative	R. F. Fletcher	KKM *
443	10	Report of Fragmentation Test Program 0179-64F May 1963 Aerojet-Gen. L. C. Walther R. W. Pate E. H. Eldredge	This report discusses the Armed Services Explosives Safety Board (ASESB) work group that studied the potential fragmentation hazard from large missiles and weapon systems. The objective was to obtain quantitative data on the size of inert and propellant fragments and their prjected distance from explosively destroyed motors.	Solid Class 1.3 Aerex S-1 Explosive	NA Experimental	Blast Fragmentation	Quantitative	R. F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
444	9	Air-Blast Parameters from TNT Spherical Air Burst and Hemispherical Surface Burst. AFBRL-TR-0255 April 1984 C. N. Kingery G. Bulmash	This report compiled, analyzed, and presented various air blast parameters associated with TNT spherical free-air burst and hemispherical surface bursts. The Hopkinson and Sach scaling laws were applied. (Some pages are missing).	TNT and Pentolite Explosives	NA	Blast	Quantitative	R. F. Fletcher	KKM *
445	9	Chemical Explosions in Space, Final Report Feb. 1965 Houston Research Institute R. F. Fletcher, NASA NAS-2640	This report developed a mathematical model to describe the characteristics resulting from ch~~ical explosions in space. The major c ideration was explosions at reduced pressure without shock waves.	Explosive	NA	Blast	Quantitative	R. F. Fletcher	KKM *
446	8	Numerical Solution of Spherical Blast Waves Harold L. Brode 1955 J. of Applied Physics Vol. 26, N-6	A strong-shock, point source solution and spherical isothermal distribution were the parameters to the numerical analysis of blast waves.	NI	NI	Blast	Quantitative	Wyle Labs (R. F. Fletcher)	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795C

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
447 367	7	STS Launch Analysis Lt. Brett Haswell AFWL/NTSN No ID #	Viewgraphs of Galileo Centaur missions of various launch accidents plus computer simulation of a launch abort.	NI	Inter Tank Fail in Tipover Pushover SRM Blowout RLTS or Once-around	NI	Qualitative Quantitative		KKM *
448	9	N75-32191, The Charac. and Evaluation of Acc. Explosion 1975 R. A. Strehlow, et. al University of IL US Dept. of Commerce National Tech. Info. Service NASA CR134779	Types of Explosion Characterized. Good Reference for TNT Equivalents	Explosives TNT	Rupture of Frangible Vessels	Blast	Qualitative Quantitative	In-House	KKM *
449	6	Impact Cratering Mechanics and Structure D. E. Gault, et. al. Space Sciences Division Ames Research Center NASA, Moffett Field California 94035 .965	Experiments conducted to measure the Impact craters characteristics based on hyper velocity	NA	NA	Ground Blast	Quantitative	R. F. Fletcher	KKM *
450	9	Prevention of and Protection against acciden-	Hazards associated with liquid propel-	LOX/RP-1 LH_2/LOX	NA	Fire Blast	Empirical & Analytical	R. F. Fletcher	KKM *

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION					REFERENCE LOCATION	REV BY
			DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY		
450	Cont	tal explosion of munitions, Fuels and other hazardous mixtures Ed Cohen et. al. Annals of the New York Academy of Sciences Vol. 152, Art. 1 p. 1-913 1968 "The Saturn Fireball" R. W. High	lants explosion which concentrated on the Thermal Environment Vehicle-Atlas/Centaur Saturn V	Aerozine-50/ N_2O_4			Data		
451	5	Craters Produced by Missile Impact H. J. Moore N67-19393	The study of Craters produced by the impact of missile Warheads	Explosives	NA	Ground Blast	Experimental	R. F. Fletcher	KKM *
452	5	The handling and storage of Nitrogen Tetroxide 1961 U. S. Army Chemical Research and Development Lab. Army Chemical Center MD 8011 AF-MIPR (33-616) 60-20	A comprehensive source of the basic information for the use in the design, fabrication and operation of Nitrogen Tetroxide Handling Equipment	N_2O_4	NA	NA	NA	R. F. Fletcher	KKM *
453	9	MIL-STD-882A (USAF) System Safety Program Requirements, 28 June 1980	Systems, Associated Subsystems, and Equipment requirements for Space and Missile Programs	NA	NA	NA	Qualitatiave	In-House	EK

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

ANNOTATED BIBLIOGRAPHY

REF NO.	RANK	REFERENCE IDENTIFICATION	REFERENCE APPLICATION DESCRIPTION	MATERIAL	FAILURE SCENARIO	ENVIRONMENT	METHODOLOGY	REFERENCE LOCATION	REV BY
454	9	Integrated System Safety Program for the MX Weapon System, SAMSO STD 79-1, AFSC Space and Missile Org., El Segundo, CA 25 September 1979	A tailored application of MIL-STD-1574A for implementing program requirements for the MX Weapon System. Identifies significant accident risks and methods to cope with those risks.	NA	NA	NA	Qualitative	In-House R. Lomax	EK
455	9	Software Sneak Circuit Analysis, DTIC, AF WL-TR-75-254; Boeing Co., J. P. Rankin, G. J. Engels, S. G Godoy Houston, TX April 1976	Describes the Proprietary Boeing Co. technique for identifying and evaluating problems in the design and operation of control systems for large scale programs.	NA	NA	NA	Qualitative Quantitative	In-House E. Hillman	EK
456	8	Engineering Reliability Policy and Procedures Manual, M63-3, Martin Marietta, 30 April 1963	Describes MMC Methodology for Reliability of Space and Missile Programs.	NA	NA	NA	Qualitative	In-House W. Willard	EK
457	4	Reliability and Maintainability Allocations Assessments and Analysis Report-IUS System, Boeing, D290-10404-1, 10 January 1979	Describes Statistical data required for the T34D/IUS Launch Vehicle Program.	NA	NA	NA	Quantitative	In-House W. Willard	EK

NA Not Applicable
NI Not Identified
Ranked 1-10 according to Pertinence of data (Preliminary Evaluation)

ID: 0002W AD: 0795G

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Mg	RTC	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
7	1	TITAN II								X	X		X													X		X	X	
6	2																													
6	3																								X	X	X	X		
5	4	WEAPONS	ARTILLERY MORTAR																											
8	5		LIQUIDS																						X	X	X	X		
9	6																									X	X			
9	7							X 94/6 AP																	X	X				
10	8					X	X						RP-1											X		X	X	X		
10	9	PYRO				X	X			X			RP-1											X		X	X	X		
10	10	PYRO				X	X						RP-1											X		X	X	X		
8	11																								X	X				X
1	12																													
2	13																													
7	14	SHUTTLE/ IUS/ GALILEO				X	X																X			X	X			
10	15	T34D/IUS	SOLID ROCKET MOTORS							X			X											X		X		X	X	
9	16	T34D/IUS																								X		X	X	
2	17	THIOKOL 2134A																												
2	18	K:																												
2	19																													
2	20	R/ THIOKOL																												
8	21																													
8	22																													
8	23	SKYNET 4																								X		X	X	
8	24	TEAL RUBY (P80-1)			GN2					X										X	LNEON	X				X		X	X	
4	25					X	X			X	X	X	X											X		X		X	X	
1	26																													
10	27	SOPHY							X																	X				
7	28							X	X																	X	X	X		
7	29							X	X																	X		X		
7	30																									X	X	X	X	
6	31	NUCLEAR WEAPONS																									X	X		
3	32									X		X	X						X											

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
10	33	LIQUID PROPELLANTS																										X		
6	34																								X	X				
10	35					X	X	NH3					X											X			X			
8	36																													
8	37																								X	X	X			
8	38																								X	X	X			
9	39		X			X	X					X	X													X		X		
7	40								X																	X				
7	41					X	X																			X		X		
7	42					X																				X	X	X		
9	43	STS		GN2			X	X	SOLID		X												X							
10	44																													
	45	PAM-D						SOLID/LIQ		X																X		X	X	X
7	46			X																										
10	47							X	X																	X	X			
7	48							SOLID		X																				
7	49	PAM-D						SOLID		X																				
8	50	TITAN 23C																					X				X	X		
10	51	SATURN/																												
		MARINER/																												
		JUPITER								X													X			X		X		
9	52	TITAN IIIE/																												
		CENTAUR				X	X	SOLID	X															X		X	X	X		
7	53	TITAN IIIE/																												
		CENTAUR				X	X		X	X			X										X	X						
10	54	SHUTTLE				X	X	SOLID (PBAN)																		X		X		
10	55	SHUTTLE				X	X	SOLID (PBAN)																		X		X		
10	56	SHUTTLE				X	X	SOLID (PBAN)																		X		X		
9	57	SHUTTLE				X	X																			X		X		
9	58	SHUTTLE				X	X																			X		X		
9	59	SHUTTLE				X	X																			X		X		
7	60																									BURST				

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	F2	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT	EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
5	61																														
6	62																										BURST				
8	63	WEAPONS																													
9	64	WEAPONS																													
9	65	WEAPONS																													
8	66																												X		
6	67									X	X	X												X			X		X	X	
5	68												X														X	X	-	X	
5	69																														
5	70																														
8	71																										X		X	X	
3	72					X	X			X																	X		X	X	
8	73						RP-1																								
8	74					X	RP-1			X			X											X			X	X	X		
8	75							SOLID																			X				
10	76	T34D						SOLID																							
10	77	T34D/IUS						SOLID/LIQ																							
10	78																										X		X		
9	79				X	X		LIQUID																			X		X		
9	80					X	RP-1																				X		X		
7	81					X	RP-1						X											X			X	X	X		
10	82									X			X														X		X	X	
9	83												X											X			X		X	X	
10	84																												X		
10	85	ATLAS/ CENTAUR 5/ SATURN																											X		
8	86																										X				
10	87																								X						
7	88							SOLID																			X				
5	89								X																			X			
5	90																														
9	91																								X		X				
8	92	TITAN III						SOLID																			X				

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT	EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
9	93																										X				
9	94							X	X																		X	X			
7	95	SOPHY						SOLID(PBAN) AND AP																	X						
3	96																										X				
8	97																										X				
7	98																								X						
6	99	ATLAS/																													
		SATURN																										X			
10	100							SOLID																				X			
1	101									X																					
8	102	TITAN II											X											X			X		X		
8	103																												X		
7	104									X	X	X	X											X					X		
8	105																														
2	106																								X						
2	107									X			X																X		
9	108	SATURN 5 LIQUIDS																											X		
10	109	ATLAS/																													
		CENTAUR				X	X	RP-1																				X	X		
7	110									X	X	X	X																		
3	111																														
5	112									X	X		X	X																	
5	113								03	X	X	X																			
5	114								02	X	X	X	X											X							
5	115									X	X	X	X																		
7	116									X	X		X																X		
3	117									X			X																		
3	118									X			X																		
2	119									X			X																		
6	120									X		X	X																		
2	121									X			X																		
3	122									X.			X																		
3	123									X	X	X	X																		

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LN2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
1	124																													
5	125									X	X	X	X											X						
8	126					X	X			X	X	X	X															X		
2	127																													
1	128																													
5	129																													
7	130									X				X														X		
5	131						X				X																			
5	132									X	X	X																		
7	133											X	X																	
3	134									X			X																	
6	135									X																				
5	136						X	RP-1		X														X						
5	137												X																	
6	138									X			X																	
5	139									X	X													X						
6	140									X	X	X	X																	
1	141							NO2																						
2	142									X																				
10	143						X	RP-1																X						
2	144									X																				
10	145												X											X				X		
2	146									X	X			X		CH3		X						X						
2	147									X			X															X		
7	148									X	X	X	X															X	X	
4	149									X																				
2	150									X																				
3	151	APOLLO 16											X													X				
10	152									X			X													X		X	X	
10	153																													
6	154						X																							
6	155												X											X				X		
5	156											X																		
5	157									NO2		X																		

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
5	158									X																				
10	159					X	X	RP-1					X											X		X		X		
3	160	SEMINAR								X	X		X		B5H9									X						
5	161	TITAN III																												
10	162	APOLLO/																												
		SATURN V							X				X																	
1	164	SEMINAR																								X				
9	163	SEMINAR						SOLID																		X				
5	165	SEMINAR																												
7	166	SEMINAR																									X			
9	167	SEMINAR																								X				
5	168	SEMINAR																												
5	169	SEMINAR																												
10	170	SEMINAR																								X	X			
8	171	SEMINAR																									X			
9	172	SEMINAR																									X			
8	173	SEMINAR				X	X	RP-1																						
9	174	SEMINAR									X	X	X													X		X		
7	175	SEMINAR												CF3													X			
9	176	SEMINAR				X	X	RP-1																			X			
10	177	SEMINAR														X	N20									X	X	X		
3	178	SEMINAR																								X				
7	179	SEMINAR					SOLID: PBAN, AMP, AP																					X		
9	180	SEMINAR																		X						X				
5	181	SEMINAR					SOLID: RDX/TNT/TORPEX4D/PU																		X	X				
5	182	SEMINAR																									X			
5	183	SEMINAR																								X				
10	184	SEMINAR																												
7	185	SEMINAR					EXPLOSIVE																			X				
5	186	SEMINAR					EXPLOSIVE																			X				
8	187	SEMINAR																							X	X		X		
7	188	SEMINAR																							X	X		X		
10	189	SEMINAR																									X			

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
8	190	SEMINAR						X																		X	X			
7	191	SEMINAR																							X	X				
5	192	SEMINAR																												
10	193	SEMINAR																								X	X	X		
5	194	SEMINAR					SOLID																							
2	195	SEMINAR																									X			
1	196	SEMINAR																												
6	197	SEMINAR	X	X			SOLID																			X	X	X	X	
8	199						RADIOISOTOPE																							
8	200						RADIOISOTOPE																							
8	201						RADIOISOTOPE																							
8	202																													
4	203																													
3	204																													
3	205	TITAN III																												
3	206	VIKING '75		X						X		X	X										X							
3	207	VIKING '75		X						X		X	X										X							
9	208	GIANT PATRIOT							SOLID																		X			
7	209	GIANT PATRIOT																									X			
9	210																									X	X			
10	211			X																						X	X			
9	212	GIANT PATRIOT																									X			
10	213	SHUTTLE/ ISCBM/ SATURN V				X	X																				X			
10	214	SHUTTLE				X	X																				X			
9	215	SHUTTLE				X	X																			X	X			
9	216					X	X																			X	X			
9	217							BUILDING EXPLOSION																	X	X				
1	218	SHUTTLE/ ET/ORBIT				X	X																							
7	219					X	X	RP-1																			X			

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
9	220					X	X	RP-1																			X			
6	221				BI PROPELLANT/MONO PROPELLANT																						X			
7	222																										X MODEL			
3	223	INCOMPLETE																												
7	224							SOLID: PBAA/CXDB																			X			
5	225																													
6	226					X	X	RP-1																		X				
3	227	INCOMPLETE																												
7	228							SOLID																			X			
10	229													X	PROPANE											X				
2	230																													
2	231																											X		
5	232	SATURN 1B				X	X	RP-1																		X				
6	234							SOLID																						
6	235						X																			X				
4	236	ATLAS					X																			X				
5	237					X	X																							
2	238					X	X																			X	X			
10	239	PYRO				X	X	RP-1																		X	X	X		
6	240						ELEC CHEM PROPULSION (GENERAL)																							
2	241	DSV-3P-11E/F DELTA							X				X											X						
4	242							SOLID PROPELLANTS										X												
6	243						X	RP-1		X																				
9	244	ATLAS					X	RP-1																						
9	245					X																				X				
8	246					X	X	RP-1																		X				
4	247																													
8	248						X	RP-1																		X				
9	249					X	X	RP-1	SOLID				X											X		X		X		
10	250	S-IV				X	X																			X	X	X		
10	251																								X		X			

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTC	A50	TNT	EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
2	252																														
2	253					EXPLOSIVES																					X		X		
2	254																														
4	255																														
4	256					CASED/UNCASED CHARGES																			X						
3	257																										X				
5	258									X		X															X		X		
10	259																											X			
9	260																								X		X				
7	261	PROJECT PYRO																									X		X		
10	262					X	X	RP-1		X	X		X														X				
9	263					MUNITIONS																						X			
7	264																										X				
9	265																										X		X		
9	266																										X		X		
10	267					NUCLEAR & EXPLOSIVE																					X	X			
9	268																										X	X			
7	269																										X				
8	270																										X				
9	271																										X	X			
8	272							SOLID-AP																					X		
5	273					EXPLOSIVE																					X	X			
1	274					SOLIDS																									
10	275	MINUTEMAN II																									X	X	X		
8	276																										X	X	X		
10	277	MINUTEMAN							RDX																		X	X	X		
7	278	TITAN III							SOLID																		X	X	X		X
10	279							X	X																						
10	280																										X	X			X
10	281	SHUTTLE				X	X	SOLID																			X	X			
10	282	STS LAUNCH				X	X	SOLID																			X		X		
10	283					X	X	SOLID																			X		X		

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
8	284	GALILEO/																												
		INT SOLAR																												
		POLAR				X	X																X			X		X		
2	285																									X				
4	286							SOLID (AP)																		X				
5	287							SOLID-TNT RDV																						
10	288	SHUTTLE/																												
		CENTAUR				X	X																X							
5	289	ORBITING																												
		VEHICLE				X	X																				X			
8	290								RDX SOLID																X					
5	291					X	X	SOLID			X	X																		
8	292	SHUTTLE				X	X																X			X	X			
5	293	SHUTTLE/																												
		CENTAUR						SOLID															X			X	X	X		
10	294	MX						X	X																	X	X	X		
	295																													
5	296								X(AP)																					
7	297								X(AP)																	X		X		
3	298	MX																												
10	299					X	X	RP-1			X			X												X				
7	300				X		X	RP-1																		X				
8	301	TITAN II											X											X		X		X	X	
8	302	SHUTTLE						SOLID																		X	X	X		
8	303	SHUTTLE						SOLID																		X	X	X		
10	304	SHUTTLE																								X	X	X		
8	305	SHUTTLE/																												
		CENTAUR/																												
		GALILEO				X	X																			X	X			
5	306	TDRSS		GN2						X																				
9	307	TDRSS/IUS																												

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTC	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
10	308	WEAPONS																								X	X	X	X	X
2	309																								X	X				
6	310							SOLID (PBAN)																		X				
5	311																									X	X			
10	312	T-34/TS								X																X		X	X	
5	313	SHUTTLE						SOLID																			X			
5	314	SHUTTLE																								X	X			
7	315							SINGLE & DOUBLE BASE SOLID																		X		X		
6	316																									X		X		
1	317													X																
10	318	GALILEO				X	X																X				X			
10	319							SOLIDS(AP)RDX											X					X	X		X			
8	320																										X			
8	321									X			X													X				X
6	322							SOLIDS																			X	X		
3	323	MARK II																												
8	324							SOLIDS																		X				
5	325							SOLIDS(AP)																				X		
6	326					X	X																			X				
4	327	STS																												
8	328							SOLID(AP)																		X				
6	329					X	X			X														X						
3	330			GH2																										
6	331			LIQUIDS																										
6	332					X	X																					X		
9	333					X	X																			X		X	X	
8	334	GALILEO/																												
		SHUTTLE				X	X			X																X	X	X		
8	335	GALILEO				X	X																			X	X	X		
9	336																						X				X			
9	337																									X	X	X		
10	338							SOLID AND TNT																	X	X	X			

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N204	NH3	CO2	CO	Fe	H202	LF2	LCH4	NF2	Hg	RTG	A50	TNT	EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
7	339																										X	X	X		
9	340							LIQUID AND SOLID																			X	X	X		
10	341							(SOLID AP-PBAN-Al)																	X		X				
9	342							LIQUID & SOLID																	X		X	X	X	X	
10	343							LIQUID & SOLID																	X		X				
8	344					X	X	RP-1																			X				
7	345					X	X																				X		X		
8	346						X	RP-1																	X		X		X		
2	347																										X	X	X		
5	348							SOLID																							
3	349							SOLIDS																							
5	350							SOLIDS																							
5	352							SOLIDS																							
6	353							SOLIDS																							
5	355							SOLIDS																							
8	356	STS	X	LH2						X			X						X			X					X	X	X	X	
9	357	TITAN III						SOLIDS & LIQUID																			X	X	X	X	X
9	358																										X		X		
8	359					X		SOLID																							
7	360				ARGON		LIQUIDS																								
7	361	SHUTTLE				X	X																					X	X		
8	362																														
8	363							NS SOLID																			X		X		
9	364							X																	X		X				
7	365							SOLID HMX																							
6	367	STS																													
10	368						X	RP-1		X														X							
8	369							SOLID PBAN DB BASE																X	X						
1	370																														
10	371	SHUTTLE				X	X	SOLID																			X	X	X		
8	372	STS				X	X	SOLID																							
8	373							SOLID																			X				

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N204	NH3	CO2	CO	Fe	H202	LF2	LCH4	NF2	Hg	NTG	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
1	374	SHUTTLE						SOLID																						
10	375							X	X																	X				
8	376	MINUTEMAN III						SOLID																						
7	377							SOLID																		X				
8	378					X	X																			X		X		
2	379							SOLID (HMX)																						
2	380																													
8	381							SOLID																						
9	382	MINUTEMAN I						SOLID (RDX)																		X	X			
5	383																													
8	384	SATURN IV																							X	X	X			
7	385							COMPOSITE SOLIDS																						
10	386							SOLID																						
4	387	MINUTEMAN WING VI						SOLID																		X				
6	388							SOLID																		X				
7	389	SHUTTLE						SOLID																		X				
9	390	SHUTTLE						SOLID																		X	X			
10	391	STS				X	X																			X				
5	392																								X	X		X		
9	393							LIQUID MONO PROPELLANT																	X		X			
9	394							LIQUID MONO PROPELLANT																	X		X			
10	395							SOLID AND LIQUIDS																		X		X		
8	396					X	X	RP-1					X											X						
9	397					X	X						X													X		X		
6	398						X																			X				
7	399						X																			X		X		
10	400					LIQUID AND SOLID																				X		X		
10	401							SOLIDS																			X	X		
9	402					X	X	RP-1																				X		
10	403					X	X																			X	X			
10	404					X																					X			
	405																								X		X			

CROSS-REFERENCE MATRIX

RK	DGC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H2O2	LF2	LCH4	NF2	Hg	RTG	A50	TNT	EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
8	406																								X		X	X			
8	407																								X			X			
6	408																										X	X			
9	409	SHUTTLE				X	X	SOLIDS																	X		X				
10	410							LIQUID																			X				
8	411	TITAN II								X	X	X	X														X		X		
7	412	SHUTTLE				X	X	RP-1																			X	X			
9	413					LIQUID																			X		X		X		
8	414																											X			
9	415							SOLID																			X				
5	416	TITAN 34D/ IUS						SOLID																			X				
5	417	TITAN 34D/ IUS						SOLID																			X				
8	418																										X				
8	419							LIQUIDS																				X			
8	420							SOLIDS																			X				
8	421					X	X	RP-1																			X		X		
7	422	MINUTEMAN I				X	X																				X				
7	423	GIANT PATRIOT																													
7	424	SHUTTLE				X	X																X				X				
8	425													X																	
10	426							SOLID																	X		X				
8	427							SOLID		238 PuO2																	X				
9	428									238 PuO2																			X		
9	429																														
8	430																														
9	431	SHUTTLE		GN2			LIQUID																								
8	432																								X		X		X		
10	433	GEMINI/ATLAS				X	X			X			X											X							
8	434	ATLAS/CENTAUR				X	X																		X		X		X		
8	435	SHUTTLE				X	X	SOLID																			X				

CROSS-REFERENCE MATRIX

RK	DOC	VEHICLE	He	N2	LHe	LH2	LOX	S1.1	S1.3	N2H4	UDMH	MMH	N2O4	NH3	CO2	CO	Fe	H202	LF2	LCH4	NF2
8	436					X	X						X								
9	437												X								
5	438	SATURN V/																			
		TITAN III						SOLID 238 PuO2													
9	439																				
9	440																				
10	441	SATURN S-14			LIQUIDS AND SOLIDS																
		TITAN I																			
9	442				X		X	RP-1			LN2										
10	443								X												
9	444																				
9	445							EXPLOSIVE													
8	446																				
7	447	SHUTTLE																			
9	448																				
6	449																				
9	450					X	X	RP-1					X								
5	451							EXPLOSIVE													
5	452												X								
9	453																				
9	454	MX																			
9	455																				
8	456																				
9	457																				

RK	DOC	Hg	RTC	A50	TNT EQ	BLST	FRGMNT	FIRE	TOXIC	ACOUST
8	436			X		X				
9	437			X						
5	438	X					X			
9	439									
9	440									
10	441					X		X	X	
9	442									
10	443					X	X			
9	444				X	X				
9	445					X				
8	446					X				
7	447									
9	448				X	X				
6	449					X				
9	450				X	X		X		
5	451					X				
5	452									
9	453									
9	454									
9	455									
8	456									
9	457									

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Appendix B
Summary of Hazardous Materials

APPENDIX B
SUMMARY OF HAZARDOUS MATERIALS

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CHAPTER B1
INTRODUCTION

B1.1 PURPOSE

B1.1.1 Purpose

This Appendix is intended to provide general information to those responsible for minimizing the hazards associated with the handling, storage, use and transportation of liquid propellant and solid propellants, and to those responsible for the design of systems and operations which use them.

B1.1.2 Scope

Information contained in this document has been prepared to assist Federal and Military Departments, Agencies and contractors in the design of systems and operations which include chemical liquid and solid propellants. The user is hereby forewarned that the applicability of the information to any specific location, situation or operation depends upon proper interpretation by experts in the fields of chemical propulsion and safety engineering.

B1.2 BACKGROUND

The data of this document were extracted primarily from CPIA (Chemical Propulsion Information Agency) Publication 394, Hazards of Chemical Rockets and Propellants, Volumes II and III, September 1984. Exceptions to this will be specifically identified.

B1.3 APPLICABILITY

This manual is intended as a source of information and as a general guide for Federal Departments, Agencies and contractors to assist them in understanding the hazardous nature of chemical liquid and solid propellants. Specific design or analysis efforts should include review of CPIA 394 or the references provided.

B1.4 LIQUID MATERIAL CLASSIFICATION TABLE

Information on liquid propellants, explosives and propellant ingredients are included in Figure B1-1. The table consists of the material name, its formula and any synonyms or common names. For each material the storage compatibility group assigned by the DOD is listed, the explosive hazard class, the National Fire Protection Agency hazard identification classification for health, flammability and reactivity, the DOT classification and labeling requirements, the United Nations and Chemical Abstract numbers, and the EPA (RCRA) classification. These classification systems are described below:

B1.4.1 Storage Compatibility Group

To assure a high degree of safety in ammunition storage, a system to classify materials according to characteristics such as chemical and physical properties, quantity-distance criteria, sensitivity and other factors was developed. Dissimilar materials are separated and stored in a safe manner. Seven Storage Compatibility Groups (SCG) have been developed and are described in CPIA 394, Appendix C. The SCG for each material is presented in Figure B1-1.

B1.4.2 Explosive Hazards Class

The DOD has a hazard classification procedure which assigns a group number to denote the types and degrees of hazards other than the chemical compatibility of the liquid propellants. This classification system is described in CPIA 394, Appendix D.

B1.4.3 NFPA Hazard Identification

The National Fire Protection Association has developed a general classification system to describe the inherent hazards of various chemicals and the order of severity of these hazards under emergency conditions. The order of severity of each hazard is indicated by use of five numerical gradings ranging from 4 (indicating a severe or extreme danger) to 0 (indicating no hazard).

The following paragraphs from the National Fire Code (Reference 1) summarize the definitions of the numbers in each hazard category and explain what a number should tell fire fighting personnel about protecting themselves and how to fight fires where the hazard exists.

B1.4.3.1 Health

Level 4. A few sniffs of the gas or vapor could cause death; or the gas, vapor, or liquid could be fatal on penetrating the fire fighters' normal full protective clothing which is designed for resistance to heat. For most chemicals having a Health 4 rating, the normal full protective clothing available to the average fire department will not provide adequate protection against skin contact with these materials. Only special protective clothing designed to protect against the specific hazard should be worn.

Level 3. Materials extremely hazardous to health, but areas may be entered with extreme care. Full protective clothing, including self-contained breathing apparatus, rubber gloves, boots and bands around legs, arms and waist should be provided. No skin surface should be exposed.

Level 2. Materials hazardous to health, but areas may be entered freely with self-contained breathing apparatus.

Level 1. Materials that are slightly hazardous to health. It may be desirable to wear self-contained breathing apparatus.

Level 0. Materials which on exposure under fire conditions would offer no health hazard beyond that of ordinary combustible material.

B1.4.3.2 <u>Flammability</u>

Level 4. Very flammable gases, very volatile flammable liquids, and materials that in the form of dusts or mists readily form explosive mixtures when dispersed in air. Shut off flow of gas or liquid and keep cooling water streams on exposed tanks or containers. Use water spray carefully in the vicinity of dusts so as not to create dust clouds.

Level 3. Liquids which can be ignited under almost all normal temperature conditions. Water may be ineffective on these liquids because of their low flash points. Solids which form coarse dusts, solids in shredded or fibrous form that create flash fires, solids that burn rapidly, usually because they contain their own oxygen, and any material that ignites spontaneously at normal temperatures in air.

Level 2. Liquids which must be moderately heated before ignition will occur and solids that readily give off flammable vapors. Water spray may be used to extinguish the fire because the material can be cooled to below its flash point.

Level 1. Materials that must be preheated before ignition can occur. Water may cause frothing of liquids with this flammability rating number if it gets below the surface of the liquid and turns to steam. However, water spray gently applied to the surface will cause a frothing which will extinguish the fire. Most combustible solids have a flammability rating of 1.

Level 0. Materials that will not burn.

B1.4.3.3 <u>Reactivity</u>

Level 4. Materials which in themselves are readily capable of detonation or of explosive decomposition or explosive reaction at normal temperatures and pressures. Includes materials which are sensitive to mechanical or localized thermal shock. If a chemical with this hazard rating is in a advanced or massive fire, the area should be evacuated.

Level 3. Materials which in themselves are capable of detonation or of explosive decomposition reaction but which require a strong initiating source or which must heated under confinement before initiation. Includes materials which are sensitive to thermal or mechanical shock at elevated temperatures and pressures or which react explosively with water without requiring heat or confinement. Fire fighting should be done from an explosion-resistant location.

Level 2. Materials which in themselves are normally unstable and readily undergo violent chemical change but do not detonate. Includes materials which can undergo chemical change with rapid release of energy at normal temperatures and pressure or which can undergo violent chemical change at elevated temperatures and pressures. Also includes those materials which may react violently with water or which may form potentially explosive mixtures with water. In advanced or massive fires, fire fighting should be done from a protected location.

Level 1. Materials which in themselves are normally stable but which may become unstable at elevated temperatures and pressures or which may react with water with some release of energy but not violently. Caution must be used in approaching the fire and applying water.

Level 0. Materials which are normally stable even under fire exposure conditions and which are not reactive with water. Normal fire fighting procedures may be used.

B1.4.4 DOT Classification

The Department of Transportation has established a hazards classification system for regulated materials and requires special colorcoded labeling on packages containing these materials. The following descriptions are used. These classifications of specific materials are found in 49 CFR 172.101 and 172.102 (Reference 2).

Explosives Class A, B or C
Flammable gas
Non-flammable gas
Flammable liquid
Flammable solid
Oxidizer
Poison A or B
Radioactive material
Corrosives
Irritating substances
Other Regulated Material A, B, C, D or E
Combustible liquid or solid
Forbidden

B1.4.5 United Nations Number and Chemical Abstract Registry Number

The United Nations (UN) has developed a serial number system to describe specific materials. This number is required on all packages containing regulated material. A complete listing of these codes is available in "Transport of Dangerous Goods" published by the United Nations (Reference 3). Those numbers preceded by a "UN" are associated with descriptions considered appropriate for international shipments as well as domestic shipments. Those preceded by an "NA" are associated with descriptions that are not recognized for international shipments, except to and from Canada. If an identification number is in the "NA9000" series, it is either associated with the description or a material that is not appropriately covered by international hazardous materials (dangerous goods) shipping standards or not appropriately addressed by such standards for emergency response information purposes, except for transportation between the United States and Canada.

Chemical Abstracts maintains a registry system for chemicals that assigns a unique number to specific chemicals and chemical compounds. Specific information on the chemical or compound can be referenced through use of the registry number.

B1.4.6 RCRA Classifications and Reportable Quantities

In Figure B1-1, Column 8, code letters and limits are listed, that describe the waste category of a material as established by the Environmental Protection Agency under the Resource Conservation and Recovery Act (RCRA) of 1976.

RCRA Class has six categories established and are identified by letter codes. The tests and conditions which are used to classify wastes as hazardous are detailed in 40 CFR 261.20 through 261.33 and summarized below (Reference 4).

a. Code I - Ignitable Waste. A waste with the characteristic of ignitability is:

   (1) a liquid with a flash point less than 333 K (140°F)

   (2) not a liquid, but is capable of causing fire

   (3) an ignitable compressed gas as defined in 49 CFR 173.00, or

   (4) is an oxidizer as defined in 49 CFR 173.151.

   Ignitable wastes are given an EPA hazardous waste number of D001 unless it is listed in Subpart D of 40 CFR 261.

b. Code C - Corrosive Waste. A waste with the characteristic of corrosivity has a pH less than 2 or greater than 12.5 or is a liquid.

c. Code R - Reactive Waste. The characteristic of reactivity is present if the solid waste is:

   (1) Normally unstable and undergoes violent change without detonating, or

   (2) reacts violently with water, or

   (3) forms potentially explosive mixture with water

   (4) generates toxic gases, vapors or fumes when mixed with water, or

   (5) is a cyanide or sulfide which generates toxic gases, vapors or fumes upon exposure to a pH of less than 2 or greater than 12.5

   (6) is capable of detonation or explosive reaction when initiated or heated in confinement, or

   (7) is capable of detonation or explosive charge at standard temperature and pressure, or

   (8) it is a Class A or B explosive.

Reactive solid wastes are given an EPA Hazardous Waste number of D003.

d. Code E - EP Toxic Waste. The characteristic of EP toxicity is present in a waste whose exempt quantity contains more than prescribed amounts of specified metals or organic compounds. The materials, limits and hazardous waste numbers are given in Figure B1-1 or 40 CFR 261.24.

e. Code T - Toxic Waste. Wastes are considered acutely toxic or toxic by listing in a series of Tables in 40 CFR 261.30-261.33. These tables are also used to define the limits of exclusion for a site and contain the hazardous waste numbers for the compounds in the table.

f. Code H - Acute Hazardous Waste.

LIST OF REFERENCES

1. NATIONAL FIRE CODES, Vol. 13, "Fire Hazard Properties of Flammable Liquids, Gases, Volatile Solids," NFPA - 325M, National Fire Protection Association, Quincy, MA, 1983.

2. CODE OF FEDERAL REGULATIONS, Title 49, TRANSPORTATION, Parts 172.101 and 172.102, 1981.

3. "Transport of Dangerous Goods," United Nations Committee of Experts on the Transport of Dangerous Goods, International Regulations Publishing and Distributing Organization, Second Revised Edition, 1982.

4. CODE OF FEDERAL REGULATIONS, Title 49, PROTECTION OF THE ENVIRONMENT, Part 261.20, Vol. 45, 1983.

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION
				Health	Flamma-bility	Reac-tivity			REPORTABLE QUANTITY (LB)
Aerozine 50 (See Hydrazine Fuels)									
Alcohols:									
a) Ethyl	C_2H_5OH/Ethanol, Type IIIA Alcohol grain alcohol (clear liquid)	C	I	0	3	0	Flammable liquid, Red Label.	UN1170 [64-17-5]	I;1000
b) Furfuryl	C_5H_5OOH/ furfurol (clear straw colored liquid)	C	I	1	2	1	Optional material, St. Andrews Cross Label.	UN2874 [98-00-0]	I,T;1000
c) Isopropyl	C_3H_7OH/ Isopropanol, rubbing alcohol (clear liquid)	C	I	1	3	0	Flammable liquid, Red Label.	UN1219 [67-63-0]	I,T;1000
d) Methyl	CH_3OH/Methanol, wood alcohol (clear liquid)	C	I	1	3	0	Flammable liquid, Red Label.	UN1230 [67-56-1]	I,T;1000
Ammonia, Anhydrous, liquid or gas	NH_3/Refrigerant Ammonia (colorless, low boiling liquid)	C	I	3	1(1)	0	Nonflammable gas, Green Label	UN1005	None;100
Argon (See Noble Gases)									
Boranes:									
a) Diborane Diborane (6) Diborane (4)	B_2H_6/Boro-ethane diboron hexahydride (low boiling liquid)	D	III	3	4	0	Flammable gas, Red Label and Poison Label	UN1911 [19287-45-7] [10099-46-1]	R;1000
b) Pentaborane Pentaborane (9) Pentaborane (11)	B_5H_9/Pentaborane 9 Pentaboron, enneahydride (clear, colorless liquid)	D	III	3	3	2	Flammable liquid, Red Label and Poison Label.	UN1380 [19624-22-7] [18433-84-6]	R;1000

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION
				Health	Flamma-bility	Reac-tivity			REPORTABLE QUANTITY (LB)
Carbon Dioxide	CO_2			—	—	—	Nonflammable gas, Green Label.	UN2187 [124-38-9]	
Carbon Monoxide	CO			2	4	0	Flammable gas, Red Label.	UN1016 [630-08-0]	
Carbon Tetrachloride (See Halocarbons)									
Chlorine	Cl_2			—	—	—	Nonflammable gas, Green Label and Poison Label.	UN1017 [7782-50-5]	10
Chlorine Pentafluoride (See Halogen Fluorides)									
Chlorine Trifluoride (See Halogen Fluorides)									
Chloroform (See Halocarbons)									
Deuterium	D_2	C		0	4	0	Optional material, Flammable gas Red Label.	UN1957 [7782-39-0]	NRC
Diborane (See Boranes)									
Ethyl Alcohol (See Alcohols)									
Ethylene (See Hydrocarbon Fuels)									
Ethylene Oxide	C_2H_4O/ETO, (colorless liquid)	D	III	2	4	3	Flammable liquid, Red Label.	UN1040 [75-21-8]	I,T:U11 10.0
Fluorine and Oxygen	F_2 and F_2-O_2 (FLOX) yellowish	A	II	—	—	—	Flammable gas, Red Label as gas; special		H,(P056), 1.0
Fluorine	F_2	A	II				Nonflammable gas, poison and oxidiser label	[7782-41-4]	
Freons (See Halocarbons)									

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPATIBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO	RCRA CLASSIFICATION
				Health	Flammability	Reactivity			REPORTABLE QUANTITY (LB)
Fuming Nitric Acid	HNO_3	A		—	—	—	Oxidizer, Yellow Label and Poison Label	UN2032 [7697-37-2]	H,(P078), 1.0
Furfuryl Alcohol (See Alcohols)									
Halocarbons:									
Fire Extinguishing Agents: Halons									T;1000
a) Halon 1202	CF_2Br_2 Dibromodifluoromethane			—	—	—	ORM-A No label required.	UN1941	
b) Halon 1211	$CBrClF_2$ Chlorodifluorobromomethane			—	—	—	Optional material, Nonflammable gas, Green Label.	UN1974	
c) Halon 1301	$CBrF_3$ Bromotrifluoromethane			—	—	—	Optional material, Nonflammable gas, Green label.	UN1009	
Refrigerants:									T(U121); 1000
a) Halon 1130 (Freon 11)	CCl_3F Trichlorofluoromethane			—	—	—	Optional material, Nonflammable gas. Green Label.	[75-69-4]	
b) Halon 1220 (Freon 12)	CCl_2F_2 Dichlorodifluoromethane			—	—	—	Optional material, Nonflammable gas, Green Label.	UN1028 [75-71-8]	

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION
				Health	Flamma-bility	Reac-tivity			REPORTABLE QUANTITY (LB)
c) Halon 1120 (Freon 21)	$CHCl_2F$ Dichloromono-fluoromethane			—	—	—	Optional material, Nonflammable gas, Green Label.	UN1029 [75-43-4]	
d) Halon 1210 (Freon 22)	$CHClF_2$ Chlorodifluoro-methane			—	—	—	Optional material, Nonflammable gas, Green Label.	UN1018 [75-45-6]	
Solvents:									
a) Carbon Tetrachloride	CCl_4 Tetrachloro-methane			—	—	—	ORM-A No label required.	UN1846 [56-23-5]	T(U211), 5000
b) Chloroform	$CHCl_3$ Trichloromethane			—	—	—	ORM-A No label required.	UN1888 [67-66-3]	I,T(U044), 5000
c) Methylene Chloride	CH_2Cl_2 Dichloromethane			2	0	1	ORM-A No label required.	UN1593 [75-09-2]	T(U080), 1000
d) Tetrachloro-ethylene	$(CCl_2)_2$ Perchloroethylene			—	—	—	ORM-A No label required	UN1897 [127-18-4]	T(U210), 1000
e) Trichloroethane no specific items				—	—	—	ORM-A No label required	[25323-89-1]	T(U210), 1000
1,1,2 trichloro-ethane	$CH_2ClCHCl_2$						ORM-A No label required	[79-00-5]	T(U226) (U227), 1000
1,1,1 trichloro-ethane	CH_3CCl_3							UN2831 [71-55-6]	
Trichloroethylene	$CHClCCl_2$ Ethylene trichloride			1	1(2)	0	ORM-A	UN1710 [79-01-6]	T(U228), 1000

(2) Practically Non-flammable

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION REPORTABLE QUANTITY (LB)
				Health	Flamma-bility	Reac-tivity			
Halogen Fluorides:									
a) Bromine pentafluoride	BrF_5/BPF	A	II	—	—	—	Oxidizer, Yellow Label.	UN1745 [7789-30-2]	
b) Chlorine Pentafluoride	ClF_5/CPF	A	II	—	—	—	Oxidizer, Yellow Label and Poison Label.	UN2548 [13637-63-3]	
c) Chlorine Trifluoride	ClF_3/CTF	A	II	—	—	—	Oxidizer, Yellow Label and Poison Label.	UN1749 [7790-91-2]	
d) Perchloryl fluoride	ClO_3F/PF	A	II	—	—	—	Non-flammable gas, Green Label.	[7616-94-6]	
Helium (See Noble Gases)									
Hydraulic Fluids:									
a) Normal Synthetic				—	—	—			None
b) Phosphate Ester	$(PO_4)_3$			—	—	—			
c) Tricresyl Phosphate (NATO C-635, H-515, H-537, H-573)	$(CH_3C_6H_4O)_3PO$			—	—	—		[1330-78-5]	
Hydrazine Fuels:									
a) Aerozine 50	(50/50 UDMH/ Hydrazine mix)	C		—	—	—	Flammable liq., Poisonous nos. Flammable liq. and poisonous labels		R,T;1000
b) H-70	70%Hydrazine-30%water mixture	C		—	—	—	Corrosive material, corrosive label		R,T;1000

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION REPORTABLE QUANTITY (LB)
				Health	Flamma-bility	Reac-tivity			
c) Hydrazine (anhydrous)	N_2H_4	C		3	3	2	Flammable liq., Red Label and Poison Label.	UN2029 [302-01-2]	R,T(U133), 1000
d) MAF-1, 3, 4	$C_3H_{22}N_6O_6$	C		—	—	—	Flammable liq., Red Label and Poison Label.		R,T;1000
e) Monomethyl Hydrazine	CH_3NHNH_2 MMH, methyl-hydrazine	C		3	3	1	Flammable liq., Red Label and Poison Label.	UN1244 [60-34-4]	R,T;1000
f) Unsymmetrical Dimethyl Hydrazine 1,1 dimethylhydrazine	$(CH_3)_2NNH_2$ UDMH	C	III	3	3	1	Flammable liq., Red Label and Poison Label.	UN1163 [57-14-7]	R,T;1000
Hydrocarbon Fuels:									
a) Aircraft Turbine and Jet Engine, JP-4,5,6,7,8,9 JP 10	MIL J-5624E	C	I	1 0	2 2	0 0	JP-7 Combus.liq JP-10 Combus.liq JP-9 Flam.liq	UN1863 UN1993 UN1863	I;1000
b) Antarctic Multi-purpose MP-1		C		—	—	—			I;1000
c) Compression Ignition and Turbine		C		—	—	—			I;1000
d) Ethylene	C_2H_4	C		1	4	2	Flammable gas, Red Label.	UN1962 [74-85-1]	I;1000
e) Thermally Stable Jet Engine(JPTS)		C		—	—	—	Combustible liquid	UN1863	I;1000
f) Methane	CH_4	C		1	4	0	Flammable gas, Red Label.	UN1971 [74-82-8]	I;1000
g) Propane	C_3H_8	C		1	4	0	Flammable gas, Red Label.	UN1978 [74-98-6]	I;1000

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION / REPORTABLE QUANTITY (LB)
				Health	Flamma-bility	Reac-tivity			
h) Ramjet Engine RJ-1, 4, 5 (Shelldyne H)		C		—	—	—		[20387-62-7]	I;1000
i) Rocket RP-1		C		—	—	—			I;1000
j) Kerosene				0	2	0	Combustible No label required	UN1223 [8008-20-6]	
Hydrogen	H_2	C		0	4	0	Flammable gas, Red Label.	UN1049 [1333-74-0]	None
Hydrogen Peroxide	H_2O_2 Hydrogen dioxide	A		—	—	—	Oxidizer Yellow Label.	UN2014 or UN2015 [7722-84-1]	
Iodine	I			—	—	—		[7553-56-2]	
Isopropyl Alcohol (See Alcohols)									
Kerosene (See Hydrocarbon Fuels)									
Krypton (See Noble Gases)									
Methane (See Hydrocarbon Fuels)									
Methyl Alcohol (See Alcohols)									
Methylene Chloride (See Halocarbons)									
Neon (See Noble Gases)									
Nitrogen	N_2	E		—	—	—	Nonflammable gas, Green Label.	UN1066 [7727-37-9]	
Nitrogen Oxides:									
a) MON-1,3,10,25		A		—	—	—	Poison A. Poison gas and oxidizer labels.		

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION / REPORTABLE QUANTITY (LB)
				Health	Flamma-bility	Reac-tivity			
b) Nitrogen Tetroxide	N_2O_4	A		—	—	—	Poison A. Poison gas label and oxidizer label.	NA1067 [10544-72-6]	H(P080), 1.0
Nitrogen Tetroxide (See Nitrogen Oxides)									
Nitrogen Trifluoride	NF_3			—	—	—	Nonflammable gas. Green Label.	UN2451 R [7783-54-2]	
Nitromethane	CH_3NO_2	F		1	3	4	Flammable liquid. Red Label.	UN1261 I [75-52-5]	
Nitrous Oxide	N_2O			—	—	—	Nonflammable gas. Green Label.	UN1070 [10024-97-2]	
Noble Gases:									None
a) Argon	A			—	—	—	Nonflammable gas, Green Label.	UN1006 [7440-37-1]	
b) Helium	He			—	—	—	Nonflammable gas. Green Label.	UN1046 [7440-59-7]	
c) Krypton	Kr			—	—	—	Optional material. Nonflammable gas. Green Label.	UN1056 [7439-90-9]	
d) Neon	Ne			—	—	—	Nonflammable gas. Green Label.	UN1065 [7440-01-9]	
e) Xenon	Xe			—	—	—	Nonflammable gas. Green Label.	UN2036 [7440-63-3]	

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

PROPELLANT OR ITEM	FORMULA/ SYNONYM	STORAGE COMPAT-IBILITY GROUP	EXPLOSIVE HAZARDS CLASS	NFPA HAZARD IDENTIFICATION			DOT CLASSIFICATION SHIPPING LABEL	UN NO./ CAS NO.	RCRA CLASSI-FICATION
				Health	Flamma-bility	Reac-tivity			REPORTABLE QUANTITY (LB)
Otto Fuel II		G		—	—	—			
Oxygen	O_2	A(liquid)		—	—	—	Nonflammable gas. Yellow oxidizer Label.	[7782-44-7]	I,R
Pentaborane (See Boranes)									
Perchlorylethylene (See Halocarbons)									
Propane (See Hydrocarbon Fuels)									
Trichloroethane (See Halocarbons)									
Trichloroethylene (See Halocarbons)									
Xenon (See Noble Gases)									

Figure B1-1 Liquid Propellant, Explosives and Propellant Ingredients Classifications (continued)

CHAPTER B2
HYDROCARBON FUELS (RP-1, METHANE)

B2.1 PROPERTIES

B2.1.1 Identification

A list of the hydrocarbon fuels discussed in CPIA 394 is given in Table B2-1, with the pertinent military specifications. In this Appendix to SPHAM, the physical properties for RP-1 and methane only are provided.

In this chapter, the phrase "liquid hydrocarbon fuel" refers to those hydrocarbon fuels which are liquid at ambient temperatures RP-1 is included. The phrase "cryogenic fuel" refers to liquid methane which is a gas at ambient temperatures and requires refrigeration for storage as a liquid. Ethylene is a liquified compressed gas. The phrase "hydrocarbon fuel" is applied to all the fuels identified.

B2.1.2 General Appearance

All of the liquid hydrocarbon fuels are clear liquids ranging in color from waterwhite to a very pale yellow.

Methane and ethylene are colorless gases at ambient conditions (References 1 and 2). Liquified fuels boil vigorously at ambient temperature and pressure, creating a voluminous cloud of condensed water-fuel vapor. Liquid methane is the primary constituent of liquefied natural gas, LNG.

B2.1.3 Physical and Chemical Properties

The specifications listed in Table B2-1 require that hydrocarbon fuels "shall consist entirely of hydrocarbon compounds," with certain additives permitted. The JP-5 and RP fuels may be described as high-boiling kerosene fractions, and JP-4 as a wide cut containing both kersene and gasoline fractions. RJ-4 is also known as Tetra-dimethylcyclopentadiene or TH-dimer. Hydrocarbon fuels vary greatly in volatility.

B2.1.3.1 Physical Properties - The physical properties of RP-1 methane are given in Figures B2-1 and B2-2, respectively.

B2.1.3.2 Solubility - All of the hydrocarbon fuels are insoluble in water. They are soluble in many organic solvents and are, themselves, excellent solvents for many organic materials.

B2.1.3.3 Stability - These fuels are chemically stable and insensitive to shock. They are stable over a wide range of storage temperatures; exposure to high temperatures accelerates the formation of gum and sediment.

B2.1.3.4 Reactivity - Some hydrocarbon fuels react violently under strong oxidizing conditions or at extremes of pressure and temperature. They are flammable at ambient conditions and their vapors form explosive mixtures with air. Ethylene is the most reactive hydrocarbon fuel; it can react explosively with strong oxidizers. Other hydrocarbon fuels may also react violently with strong oxidizers at specific operating conditions. The flashpoint of the fuel is an indicator of the order of reactivity.

B2.1.3.5 Environmental Fate - Hydrocarbons contaminate the air, water, land and sediments through disposal of waste by-products from manufacturing, spills or leaks during transportation, use or storage and as chemical transformation by-products during and after disposal (Reference 3). The major sources of airborne hydrocarbons in the atmosphere is from automobiles and industrial processes.

Airborne hydrocarbons resist washout and undergo photochemical transformations. Nonaromatic hydrocarbons decompose in water; aromatic hydrocarbons are relatively persistent (References 4 and 5). Some exchange of hydrocarbons between air and water occurs, dependent on the hydrocarbon, temperature, mixing rates, turbulence, water quality, and concentrations.

In soil, hydrocarbons are degraded by soil organisms or evaporate. Recent data indicates that some migration of the water table may occur (Reference 6).

B2.2 HAZARDS

B2.2.1 Health Hazards

B2.2.1.1 Toxicity - The hydrocarbon fuels are moderate skin-irritants; repeated contact can cause scaling and fissuring of skin and, rarely, blistering (Reference 7).

Inhalation of aliphatic hydrocarbon vapors can cause narcosis; however, the most severe hazards occur if they are ingested. Aliphatic hydrocarbons are not especially toxic; however, gasping while swallowing or aspiration from vomiting can introduce the liquid into the lungs and cause chemical pneumonitis.

Liquid hydrocarbon fuels with a high aromatic content are more toxic. Exposure to aromatic components, such as benzene, can cause toxic effects on blood-forming tissues. At this time no aromatics have been demonstrated harmless although some, such as benzene, are considered more toxic than others.

The cryogenic fuels have low toxicity and their health hazards are related to their low temperatures and asphyxiant potential. N-hexane, often a major component of JP-4 and possibly JP-5 (though less likely), can cause motor neuropathy to exposed workers when inhaled.

B2.2.1.2 Exposure Limits - Exposure limits for the liquid hydrocarbon fuels are composition dependent. TLVs for fuels not containing aromatic hydrocarbons are higher - usually 500 ppm or more, while those for the aromatic hydrocarbons toluene and xylene are 100 ppm. The TLV for benzene is, at 10 ppm, the lowest. Values for mixtures may be derived as described in Appendix C.1 of Reference 8.

Table B2-1 Identification of Hydrocarbon Fuels

	NAME	SPECIFICATION
FUEL:	Aircraft Turbine and Jet Engine, Thermally Stable	MIL-F-25524
	Compression Ignition and Turbine Engine	MIL-F-46005
	Multipurpose, Antarctic MP-1	MIL-F-23188
	Ramjet Engine, Grade RJ-1	
	Ramjet Engine, Grade RJ-4	MIL-F-32522
	Ramjet Engine, Grade RJ-5	
JET FUEL:	Grades JP-4 and JP-5 (Turbine Fuel, Aviation)	MIL-T-5624
	Grade JP-6 (Jet Fuel)	
	Grade JP-7	MIL-T-38219A
	Grade JP-8	MIL-T-83133A
	Grade JP-9	MIL-P-87107B
	Grade JP-10	MIL-P-87107B
	Grades I and II (Jet Fuel, Referee)	MIL-J-5161
ROCKET FUEL:	RP-1 (Propellant, Kerosene)	MIL-P-25576
OTHER:	Methane	
	Ethylene	

Exposure limits for cryogenic fuels are set at 10 percent of their lower explosion limits (LELs) in air to mitigate hazards of fire and explosion (See Section B2.3.6). Concentrations in air of one percent or lower for methane and 0.6 percent for ethylene are recommended.

Specific Emergency Exposure limits also are composition dependent for hydrocarbon fuels; operational limits will be defined by the safety officer, industrial hygienist and other cognizant personnel (Reference 9).

B2.2.2 Fire and Combustion Product Hazards

Hydrocarbon fuel vapors readily form ignitable mixtures with air. The ignition hazard of the fuels varies with temperature and fuel. Toxic products can be produced in a fuel rich fire. The figures at the end of this chapter give temperature ranges over which the fuels form vapors that can burn or explode.

B2.2.3 Explosion Hazards

Hydrocarbon fuel vapors can form explosive mixtures with air.

If hydrocarbon fuels and rocket oxidizers are spilled and they or their vapors mix, the resultant mixtures can be exploded by mechanical shock, heat, or spark.

B2.2.4 Environmental Effects

The principal environmental effects of hydrocarbons are in air and are associated with their conversion in the presence of light and nitrogen oxides to photochemical oxidants. These oxidants are responsible for the plant damage and eye irritation associated with smog. Since methane is not photochemically active, hydrocarbon emission standards are divided into methane and non-methane hydrocarbon emissions.

B2.3 RESERVED

B2.4 RESERVED

B2.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B2.5.1 Materials

Except for cryogens the prime consideration is choosing materials for use with hydrocarbon fuels is related to their solvent action on most organic matter. Corrosion associated with hydrocarbon fuels at ambient temperatures is negligible for storage and handling of liquids. Limitations which should be observed are outlined in this section.

B2.5.1.1 Metals - Common ferrous and non-ferrous alloys are suitable for the fabrication of containers (fixed or mobile drums and tanks), associated piping and fittings, pumping equipment, valves, and other metal parts. Long term storage of hydrocarbon fuels may require special consideration of compatible metal containers and associated equipment or parts.

B2.5.1.2 Non-Metals - Listed below are some of the recommended and prohibited non-metals:

Table B2-2 Material Compatibility

RECOMMENDED	PROHIBITED
Cork, or paper gasket materials designed for this service Nitrile, Buna N, or NBR Fluorocarbons (Teflon, Kel-F, Halon TFE, Fluorel, Viton) Fluorosilicone, or LS (for static operations) Polyamides Polyethylene Polyurethane (high tensile strength, shock loads), or Adeprene, Cyanoprene, Formrex Neoprene (especially good in refrigeration systems) Vinyls Vamac, or Ethylene Acrylate (excellent resistance to oxidation)	Acrylics Polyisobutylenes Polyurethane, soften at high temperatures Methylene Propylene, or Nordel, Epear, Vistalon, Epsyn, Royalene Natural Rubber Synthetics, other than those listed as usable

B2.5.1.3 Lubricants - Graphite-base, molybdenumdisulfide, some silicone and fluorocarbon lubricants may be used.

Since hydrocarbon fuels are excellent solvents for most organic matter, petroleum lubricants are not recommended.

B2.5.2 Equipment

B2.5.2.1 Containers - Hydrocarbon fuels are drummed or stored in tanks of approved design and construction, which may be either permanent or transportable. See DOT regulations for container requirements. Portable containers should be gas-tight (See Venting, Section B2.5.2.8).

B2.5.2.2 Pumps - Since storage tanks in this service are usually built with bottom outlets, standard centrifugal pumps may be used. Permanently installed pumps in main storage systems may also be equipped with a liquid reservoir to serve as a primer for the pump used to empty tank cars, trucks, drums, etc., not equipped with bottom outlets.

Only pumps specifically designed and qualified by tests should be used for cryogenic liquids.

B2.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 23).

B2.5.2.4 Pipes and Fittings - Pipes and fitting should be made of approved materials (See Section B2.5.1), and should be hydrostatically tested. Although threaded connections with the proper thread sealing compound are permitted for this service, flanged or welded connections are preferred. Top loading lines should extend into the bottom of the tank. For cryogen fuel service, liquid piping sections which can be accidentally blocked should be provided with small pressure-relief valves discharging to the atmosphere to prevent destructive overpressure.

B2.5.2.5 Hose and Gaskets - Hoses used in hydrocarbon fuel service may be constructed of materials as listed in Section B2.5.1.2. Gaskets may be made of any of the following materials:

Commercial cork or paper gasket material
Fluorocarbons (Teflon Kel-F, TFE or equivalent)
Neoprene
Polyethylene

Hoses for cryogenic fuels should be of a proper design and engineered specifically for this service.

B2.5.2.6 Pressure Gages - Standard-type pressure gages can be used in hydrocarbon fuel service. In order to minimize operator reading errors, all pressure gages used for a common purpose should have identical scales.

Liquid cryogen equipment should be monitored with approved types of pressure gauges as required. Gages should be protected with blowout relief backs or plugs.

B2.5.2.7 Valves - A steel or TFE (Teflon or equivalent) plug valve is recommended as the most suitable closure for liquid hydrocarbon fuels. Valves once used for liquid hydrocarbon fuel service must not be used in the transfer or storage of oxidizers unless thoroughly cleaned.

Valves must conform to particular specifications for their use with cryogenic fuels. Extended-stem globe or gate-type valves are recommended but plug or ball-type valves may be used. The valves should be provided with venting devices. Relief valves should preferably discharge vertically upward above the top of the protected vessel through stacks that are the same size as the outlet connection of the relief valve. Vessels which contain cryogenic fuels should be individually relieved. However, where two or more vessels are connected by piping without valves, the group may be treated as a whole. Relief valves should be connected into the vapor space. The relief valve setting should be such as to protect the vessel or vessels involved and should be sized for operating emergencies or exposure to fire.

B2.5.2.8 Venting Systems and Pressure Relief - All openings in the storage system should terminate outdoors, and should be protected by approved flame arrestors. Vents should be sized according to the specifications given in the National Fire Codes (Reference 24). All vents and pressure-relief systems will terminate at a height and location that will give adequate protection for personnel and buildings. Vents on atmospheric tanks should be of the pressure-vacuum type to avoid collapse of tanks when withdrawing fuel and to relieve pressure when filling.

Relief valves should preferably discharge vertically upward above the top of the protected vessel through stacks that are the same size as the outlet connection of the relief valve. Vessels which contain cryogenic fuels should be individually relieved. However, where two or more vessels are connected by piping without valves, the group may be treated as a whole. Relief valves should be connected into the vapor space. The relief valve setting should be such as to protect the vessel or vessels involved, and should be sized for operating emergencies or exposure to fire.

B2.5.2.9 Grounding - Since hydrocarbon fuels are highly flammable, all stationary or mobile tanks should be bonded and grounded to prevent fuel ignition by static electricity. The ground resistance should be monitored regularly; it should not exceed 25 ohms.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		350 - 525°F	450 - 547 K
Freezing Point		-40°F	233 K
Density (Gas)			
(Liquid)		6.7 - 6.8 lb/gal at 68°F	0.80 - 0.81 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)		42.0 - 45.5 API	0.799 - 0.801
Vapor Pressure		0.20 - 0.8 psia at 50°F	1.4 - 5.5 kPa at 283 K
		0.35 - 1.2 psia at 100°F	2.4 - 8.3 kPa at 311 K
Coefficient of Viscosity	Kinematic	16.5 centistokes at -30°F	1.65×10^{-6} m^2/s at 293 K
	Absolute		
Autoignition Temperature			
Flash Point, Closed Cup (P-M(1))		110°F	316.5 K
Flammability Range	Lean:	110°F	316.5 K
	Rich:	175 - 185°F	352.6 - 358 K

(1)Pensky-Martin Closed Cup

Figure B2-1 Physical Properties of Rocket Fuel RP-1 (Propellant, Kerosene)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-258.7°F	111.63 K
Freezing Point		-296.5°F	90.65 K
Density (Gas)		1.57×10^{-2} lb/ft^3	0.252 kg/m^3
(Liquid)		3.54 lb/gal at -258.7°F	0.451 Mg/m^3 at 111.65 K
Specific Gravity - Vapor (relative to STP air)		0.554 at 70°F	0.554 at 293 K
Critical Density		1.35 lb/gal	0.163 Mg/m^3
Critical Pressure		667 psia	4.60 MPa
Critical Temperature		-116.67°F	190.555 K
Vapor Pressure		32.5 psia at -238°F 14.7 psia -258.7°F	224.1 kPa at 123 K 101.4 kPa at 111.63 K
Coefficient of Viscosity	Kinematic	0.0252 centistokes at 68°F	2.52×10^{-8} m^2/s at 293 K
	Absolute	0.0107 centipoises at 77 °F	1.07×10^{-6} Pa·s at 298 K
Autoignition Temperature		999°F	810 K
Flash Point		-305°F	85.9 K
Flammability Limits	Lower: Upper:	5 vol %in air 14 vol %in air	

Figure B2-2 Physical Properties of Methane (MW=16.04g) Reference 1,31

CHAPTER B3
ANHYDROUS AMMONIA

B3.1 PROPERTIES

B3.1.1 Identification

Propellant grade anhydrous ammonia is 99.5 percent (by weight) basic ammonia (NH_3); the remainder is primarily water. Military Specification MIL-P-27406 Propellant, Ammonia (Reference 1) covers this grade. The molecular weight (MW) of ammonia is 17.03 g mol.

B3.1.2 General Appearance and Common Uses

At atmospheric temperatures and pressures, ammonia is a pungent, colorless vapor. The liquid is clear and colorless. It is used in the manufacture of nitric acid, explosives, synthetic fibers and fertilizers (References 2, 3, 4 and 5).

B3.1.3 Physical and Chemical Properties

See Figure B.3-1.

B3.1.3.1 Solubility - Ammonia is soluble in water, alcohol, ether, and many other solvents. Ammonia is an ionizing solvent for a large number of inorganic and organic salts.

B3.1.3.2 Stability - Ammonia is very stable and is not shock sensitive. It is thermally stable at temperatures as high as 755 K (900°F) with dissociation to nitrogen and hydrogen beginning above this temperature.

B3.1.3.3 Reactivity - Ammonia is highly reactive, alkaline in nature, and a reducing agent. Concentrations of 16 of 27 percent burn in air. Ammonia reacts with organic or inorganic acids and forms white fumes in the presence of acid vapors. Moist ammonia corrodes copper, zinc, and many alloys, particularly copper alloys. The reaction of ammonia with oxidizing agents may be violent.

B3.1.3.4 Environmental Fate - Any spill or leak of anhydrous ammonia, would vaporize into the atmosphere. Once in the atmosphere, it would be absorbed rapidly into rain droplets. After absorption, the ammonia would react with the constituents of the water droplets such as sulfates and nitrates. These forms would be re-deposited on land in a precipitation event. Small amounts of ammonia not absorbed by the rain droplets may difuse into the stratosphere.

B3.2 HAZARDS

B3.2.1 Health Hazards (References 6, 7, 8, 9, 10, and 11)

B3.2.1.1 Toxicity - Anhydrous liquid ammonia produces severe burns on contact, due to its caustic action and freezing during evaporation.

Ammonia gas at concentrations of 1 percent by volume can cause death in a few minutes; 0.2 percent can cause fatal respiratory tract irritation in 30 minutes. Irritation of the eyes, respiratory tract, and throat can result from concentrations of 0.05 to 0.1 percent. The maximum concentration tolerated by the skin is 2 percent by volume in air (Reference 8). A concentration of 500 ppm (350 mg/m^3) is considered "immediately dangerous to life or health" (Reference 11).

B3.2.1.2 Exposure Limits

B3.2.1.2.1 Threshold Limit Values (References 8, 9 and 10) - Ammonia: 25 ppm (18 mg/m^3). NIOSH (reference 11, 12) recommends a ceiling value of 50 ppm (35 mg/m^3) for 5 minute exposures. The TLV-STEL for ammonia is 35 ppm (27 mg/m^3) and the IDLH is 500 ppm (350 mg/m^3).

B3.2.1.2.2 Emergency Exposure Limits - The emergency exposure limit (EEL) defines the "single brief accidental exposures to air-borne contaminants that can be tolerated without permanent toxic effects. These limits are not intended to replace accepted safe practices and should be accompanied by appropriate medical surveillance" (Reference 13).

10 min. 500 ppm (350 mg/m^3)
30 min. 300 ppm (210 mg/m^3)

B3.2.1.3 Special Medical Information - The immediate threat to life is respiratory tract irritation and severe skin damage. Recommendations for prevention and medical care are contained ni the References 14 through 21.

B3.2.2 Fire and Combustion Product Hazards - The flammability range of ammonia (16 to 25 percent by volume) in air is higher than for most hydrocarbons but large spills of anhydrous ammonia will represent a fire hazard. In the presence of a high oxygen concentration, ammonia vapors burn vigorously. The flammable concentration range in oxygen is broader than in air.

B3.2.3 Explosion Hazards - The ready detection of ammonia at low concentrations due to its odor and prompt irritant action affords enough warning so that explosive concentrations can normally be prevented. Pressure ruptures of explosive violence can result from overfilling, dropping, or subjecting containers to high temperatures. Also, violent pressure ruptures can result from using equipment and piping that do not have safety valves to relieve excessive pressure, and from repairs in inadequately purged equipment.

The explosive range of ammonia is broadened by the admixture of oxygen to air and by temperatures and pressures above one atmosphere. Contact of ammonia with certain chemicals, including mercury, chlorine, iodine, bromine, calcium, silver oxide, or hypochlorite, can form explosive compounds. Mercury instruments should never be employed in contact with liquid or gaseous anhydrous ammonia.

B3.2.4 Environmental Effects - A background concentration of ammonia always exists in the atmosphere, however is on the order of ppb. Ammonia, as such, would be expected to exist only for a short time in the atmosphere before transformation. For an atmospheric release, therefore, other than occupational effects resulting immediately from the release, the environmental exposure effects are associated with the return of the ammonium compounds to the land in precipitation.

Ammonia and other nitrogen compounds are natural constituents of natural water systems. A sudden addition of ammonia, however, to a surface water would be toxic to aquatic life in large amounts, and cause biostimulation of aquatic life in small amounts. The more toxic form is the ionized ammonia the amount of which is a function of the pH, but is also enhanced by dissolved oxygen, carbon dioxide, elevated temperatures and bicarbonate alkalinity. Acute toxicity has been reported at levels of 10 mg/m^3 to 2 g/m^3 (6.24×10^{-7} to 1.25×10^{-4} lb/ft^3) of ammonia-nitrogen.

B3.3 RESERVED

B3.4 RESERVED

B3.5 MATERIALS AND EQUIPMENT COMPATIBILITY (Reference 23)

B3.5.1 Materials - Ammonia should be stored in cylinders of approved design, materials and construction and suitably housed. When selecting materials for liquid service, consideration should be given to physical properties at high pressures and the reactivity of the material with liquid ammonia. Elevated temperatures may cause containers to explode. Contact with strong oxidizers may cause fire and explosions. The presence of calcium, hypochlorite bleaches, gold, mercury, and silver may lead to highly explosive products. Contact with halogens may cause violent spattering (Reference 12).

B3.5.1.1 Metals - Moist ammonia will not corrode iron, steel, or aluminum but will react rapidly with copper, zinc, and many alloys, especially those containing copper. Only steel should be used for ammonia containers, piping, and pipe fittings. Such items as gauging devices and safety relief valves may be made of aluminum or other approved materials. Inconel is the best metal for high temperature operation.

a. Metals compatible with liquid anhydrous ammonia or ammonia gas:

   nickel (all temperatures)
   stainless steel, 300 and 400 series (all temperatures)
   steel (ambient temperatures)

b. Metals compatible with ammonia gas containing significant moisture:

   nickel
   copper-free steel
   stainless steel, 300 and 400 series

   Do not use copper, zinc, or their alloys.

B3.5.1.2 Non-Metals - In the following list are non-metals approved for service with ammonia:

Tetrafluoroethylene polymer
(TFE, Halon TFE, Teflon, or equivalent)
Chlorotrifluoroethylene polymer
(Kel F, Halon CTF, or equivalent)

Other materials which have been tested and approved for ammonia service may be used.

B3.5.1.3 Lubricants - Refrigeration-grade petroleum oil may be used for pumps and compressors. Specialized lubricants, such as the fluorolubes or the perfluorocarbons, are required in missiles contact systems with oxidizers is a possibility, except that they should not be used with aluminum. Silicone greases may be used.

B3.5.1.4 Sealants - Materials listed in Paragraph B3.5.1.2 in tape or paste form may be used as thread sealants. Glycerine-litharge joint cement may be used for semipermanent joints. Also see Sections B3.5.2.4 and B3.5.2.5 for specific material applications.

B3.5.2 Equipment

B3.5.2.1 Containers - Containers for ammonia may be cylinders, portable tanks, tank trucks, or single or multiple unit tank cars. Ammonia may be stored in the foregoing equipment or in refrigerated or non-refrigerated bulk storage tanks.

B3.5.2.2 Pumps and Hose - When pumps are used in the ammonia transfer system, only pumps and shaft seals designed and suitable for liquid ammonia service should be used. Hoses must be suitable for ammonia service. Detailed information for pumps and seals may be procured from manufacturers of liquid ammonia or of ammonia-handling equipment. (Ammonia hose specifications can be obtained from the Rubber Manufacturers Association, 346 Connecticut Avenue N.W., Washington, DC 20036).

B3.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 24).

B3.5.2.4 Pipes and Fittings - Piping should be extra-heavy steel (ANSI B36.10-1975, schedule 80) on the supply side of pressure-reducing valves, and of standard steel (ANSI B36.10-1975, schedule 40) on the discharge side of reducing valves. Galvanized pipe should never be used. Welded joints are preferred to threaded joints. Under no circumstances should brazed joints be used; they will deteriorate rapidly. Where threaded connections must be made, only schedule 80 pipe should be used. Freshly made glycerine-litharge joint cement may be used. There are also plastic-lead thread compounds and tetrafluoroethylene polymer tape which do not freeze thread joints as glycerine-litharge cement does. The joints should be made up tight. Ammonia-type tongue and groove glanges may be used, but other standard flanges are also satisfactory. Two-bolt flanges of this type are satisfactory for pipe sizes under 1 inch in the 2,000-pound class or for pipe sizes under one-half inch n the 6,000-pound class. Above these diameters use 300-pound ANSI flanges in high pressure systems and 150-pound ANSI flanges in reduced pressure systems. Wherever there is a possibility that an ammonia line may be closed at both ends while liquid-filled, the line should be protected by a hydrostatic relief valve. Ammonia piping systems should be clearly identified (Reference 25).

B3.5.2.5 Gaskets - Gaskets of any of the following types may be used:

a. Tetrafluoroethylene polymer (sheets)
b. Tetrafluoroethylene polymer (envelope)
c. Chlorotrifluoroethylene polymer
d. Ammonia-resistant rubber

B3.5.2.6 Pressure Gauges - All iron ammonia-type gauges are suitable for ammonia transfer and storage service. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B3.5.2.7 Valves - Tank and line shutoff valves should be steel or ductile iron (nodular) and designed for ammonia service. No valve part should contain copper, zinc, or other metals or alloys not resistant to ammonia. Valve seats may be steel, tin, or ammonia resistant resilent material. Properly designed lead seats are satisfactory in certain services. The above materials also are desirable for relief valves and pressure reducing valves, although aluminum (noncopper bearing) may be used in the smaller sizes (under 2 in IPS). Ensure that all valves, fittings and regulators are rates at a gauge pressure of 2069 kPa (300 psi).

When ordering such equipment, always specify that it must be suitable for ammonia service and must have no brass or bronze parts.

All liquid and gas connections to containers, except for relief valves and liquid level pressure-gauging devices, should be equipped either with suitable excess-flow check valves, back-pressure check valves, or remotely controlled block valves. In addition, all connections except safety-relief connections should have manually operated shutoff valves. Connections for gauging devices need not include excess-flow check valves if they incorporate orifices no larger than a No. 54 drill size to restrict the flow.

Excess-flow valves are rated by the manufacturer to close automatically at certain flow rates. Connections and fittings should have greater capacity than the rated flow of the excess-flow valve. To permit reopening the excess-flow check valves after closing a downstream valve, excess-flow valves should incorporate a passageway no larger than a No. 60 drill size in order to equalize pressures. Excess-flow and back-pressure check valves should be inside the tank or at a point outside where the line enters the tank. In the latter case, installation should be made so that any unusual strain beyond the excess-flow or the back-pressure check valve would not cause breakage between the tank and the valves.

Each permanently installed tank should have an ammonia pressure gauge with an adequate pressure range, installed to indicate vapor pressure at the tank top.

B3.5.2.8 Venting Systems and Pressure Relief - All storage containers should be protected with relief valves. Relief valves should be set to discharge at pressures which comply with the ASME code (Reference 26); the applicable federal, state, or municipal regulations; and the application insurance restrictions. Initial discharge pressure and maximum control pressure are based on the design working pressure of the container.

Relief valves should be tested at least once each year. Inspection problems can be simplified by using dual relief valves on a three-way valve to meet each relief valve requirement. The size of the relief valves should be in accordance with the flow-rate requirements of the ASME code (Reference 26). The discharge from these valves should be vented upward into the atmosphere so as not to contaminate the area nearby or endanger workers or passers-by. The exhaust end of the discharge pipe should be designed to prevent the entrance of rain. Provision should be made to drain any condensation which might accumulate in the vent lines.

Shutoff valves should not be installed between the safety-valves and the tank. Safety-relief valves should be directly connected with the vapor space of the tank. Safety-relief valves should begin to discharge at a pressure in excess of the design working pressure (to 110 percent), and their total relieving capacity should be sufficient to prevent a maximum pressure of more than 120 percent of the design working pressure. (A convenient table which relates ammonia container surface areas to the size of relief valves, and thus to air relief capacity, is available in "Unfired Pressure Vessel Safety Orders," Title 8, Section 58.7, Paragraph 511, a publication of the California Division of Industrial Safety.)

On refrigerated storage, there should be at least two safety-relief valves on every storage container, each set to open at the design working pressure and to maintain the pressure at not more than 120 percent of the design working pressure.

B3.5.2.9 Grounding - All electrical equipment should be properly grounded.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-28°F	239.8 K
Freezing Point		-107.9°F	195.4 K
Density (Gas)			
(Liquid)		5.7 lb/gal at -28°F	0.682 Mg/m^3 at 240 K
Specific Gravity - Vapor (relative to STP air)		0.59	0.59
Critical Density		14.71 lb/ft^3	0.235 Mg/m^3
Critical Pressure		1657 psia	11.427 MPa
Critical Temperature		271.4°F	406 K
Vapor Pressure		124.5 psia at 68°F	0.8578 MPa at 293 K
		225.5 psia at 104°F	1.554 MPa at 313 K
Coefficient of Viscosity	Kinematic		
	Absolute	0.26 centipoises at -28°F	2.6×10^{-4} Pa·s
		0.15 centipoises at 68°F	1.5×10^{-4} Pa·s at 293 K
Autoignition Temperature		1204°F	924.3 K
Flash Point			
Flammability Limits	Lower:	16 percent by volume in air	
	Upper:	26 percent by volume in air	

Figure B3-1 Physical Properties of Ammonia (MW = 17.03g)

CHAPTER B4
THE HYDRAZINE FUELS

B4.1 PROPERTIES

B4.1.1 Identification

B4.1.1.1 Hydrazine (Diamide, Diamine, Hydrazine Base, HZ, Hydrazine Hydrate - Propellant-grade hydrazine contains a minimum of 97.5 percent hydrazine (N_2H_4), the remainder is primarily water. Military Specification MIL-P-26536B (USAF), "Propellant, Hydrazine," (Reference 1) covers this grade. The molecular weight of hydrazine is 32.05 g mol.

B4.1.1.2 Monomethylhydrazine - Propellant-grade monomethylhydrazine (MMH) contains a minimum of 98 percent basic MMH (CH_3NHNH_2); the remainder is primarily water. Military Specification MIL-P-27404 (USAF), "Propellant, Monomethylhydrazine," (Reference 2) covers this grade. The molecular weight of MMH is 46.07 g mol.

B4.1.1.3 Unsymmetrical Dimethylhydrazine (UDMH) - Propellant grade UDMH contains a minimum of 98.0 percent UDMH, $(CH_3)_2N_2H_2$; the remainder is related amines and water (0.3 percent maximum). Military Specification MIL-P-25604C, "Propellant, Unsymmetrical Dimethylhydrazine," (Reference 3) covers this grade of fuel. The molecular weight of UDMH is 60.11 g mol.

B4.1.1.4 Mixed Amine Fuels (MAF) - MAF is composed primarily of UDMH and diethylenetriamine (DETA) combined in varying percentages, Technical grade DETA, 97 percent $(NH_2C_2H_4)_2NH$, is covered in Federal Specification O-D-1271 "Diethylenetriamine, Technical" (Reference 4). The mixed amine fuels MAF-1, MAF-3, and MAF-4 are described in the following paragraphs:

B4.1.1.4.1 MAF-1 - This fuel is covered in Military Specification MIL-P-23741A, "Mixed Amine Fuel, MAF-1" (Reference 5). It has the following composition:

UDMH	39.0 percent by weight
DETA	49.7 percent by weight
Acetonitrile (CH_3CN)	10.1 percent by weight
Water	1.0 percent by weight (maximum)

B4.1.1.4.2 MAF-3 - This fuel is covered in Military Specification MIL-P-23686A "Mixed Amine Fuel, MAF-3" (Reference 6). It has the following composition:

UDMH	20.0 percent by weight
DETA	80.0 percent by weight
Water	1.0 percent by weight (maximum)

B4.1.1.4.3 MAF-4 - This fuel, also called U-DETA and Hydyne (Reference 7), has the following composition:

UDMH	60 percent by weight
DETA	40 percent by weight

B4.1.1.4.4 Hydrazine-Unsymmetrical Dimethylhydrazine Mixture - The mixture of hydrazine and UDMH (also know as Aerozine-50) is a 1:1 blend of hydrazine and UDMH containing a minimum of 98.2 percent of the combined liquids (Reference 8). Information and safety procedures in this chapter are applicable for this fuel mixture. Where procedures differ, the more stringent should be followed. If the mixture freezes, the UDMH and hydrazine will separate into two distinct layers which will not blend unless a special mixing operation is completed. See MIL-P-27402A (USAF) "Propellant, Hydrazine - Unsymmetrical Dimethylhydrazine, 50-N_2H_4 50-UDMH" (Reference 9). Vapors from a standing source of A-50 will initially be almost entirely UDMH. The remaining liquid will ultimately be almost entirely hydrazine (Reference 10).

B4.1.2 General Appearances and Uses (Reference 11)

Hydrazine, MMH and UDMH at ordinary temperatures are clear, oily, water-white liquids with fishy amine odor. Mixed amine fuels are water white to light amber in color.

Hydrazine, MMH and UDMH are used as rocket fuels and hydrazine is commonly used as monopropellants.

B4.1.3 Physical and Chemical Properties (References 12, 13 and 14)

The chemical properties of hydrazine, MMH, UDMH and A-50 are shown in Figures B4-1 through B4-4.

B4.1.3.1 Solubility - Hydrazine and MMH mix with water and with low molecular weight alcohols. Hydrazine is insoluble in hydrocarbons while MMH is soluble. UDMH is completely miscible with water, hydrazine, diethylene-triamine, ethanol, and most petroleum fuels. The solubility of mixed amine fuels with gasoline and JP-4 is limited.

B4.1.3.2 Stability - Hydrazine and MMH are stable liquids under the extremes of heat and cold expected in long term storage. They will freeze, but because they contract upon freezing there is no danger to storage vessels. Freezing has no effect upon the chemical properties. Hydrazine, MMH, UDMH and MAF are all stable to friction and shock. Catalysts such as copper, platinum metals, copper alloys, molybdenum, and iron oxide can cause decomposition and possible ignition of hydrazine and MMH. Hydrazine begins to decompose at about 548 K (518°F) (See Figure B4-1). The presence of catalyst reduces this decomposition temperature. MMH is stable up to its boiling point if kept from contact with air. Nitrogen gas blankets should be used on all hydrazines as they are hygroscopic and moisture in air can contaminate fuel. UDMH and MAF show good thermal stability, although some carbonization takes place at 644 to 700 K (700 to 800°F). The spontaneous decomposition temperature of UDMH in an atmosphere of nitrogen or helium has been determined to be 655 to 672 K (740 to 750°F) at 101 kPa (1 atmosphere); the decomposition does not become explosive up to at least 873 K (1112°F).

B4.1.3.3 Reactivity - Hydrazine, MMH, UDMH and the mixed amine fuels are strong reducing agents, weakly alkaline, and very hygroscopic. They will react with carbon dioxide and oxygen in air (Reference 15). Exposure to air from a large surface, as from saturated rags, may result in spontaneous ignition of hydrazine of MMH, due to the heat evolved from oxidation with atmospheric oxygen. A film of hydrazine or MMH in contact with metallic oxides or other oxidizing agents may ignite spontaneously. These fuels decompose on contact with some metals including iron, copper, platinum, molybdenum, and their alloys and oxides.

B4.1.3.4 Environmental Fate - Hydrazine, monomethylhydrazine and unsymmetrical dimethylhydrazine are all heavier than air, and if not oxidized when airborne will react and/or possibly ignite with the porous earth or will form the ammonia, methylamine and dimethyl semine, respectively, and the oxides or nitrogen. All of these substances are soluble in water and are toxic and injurious to plant and lower animal life if present in sufficient concentrations. On further oxidation of the ammonia, methylamine and dimethylamine, the amino substances serve as nutrient to plant life. Airborne nitrogen oxides will return to earth as nitric acid rains in precipitation events and will react with calcium and magnesium to form the nitrates which are also utilized as fertilizer for plant life.

B4.2 HAZARDS

B4.2.1 Health Hazards

B4.2.1.1 Toxicity - Hydrazine, MMH and UDMH are convulsant agents, irritants to the respiratory tract and eyes, and may irritate the skin. They are absorbed by the skin, oral, and inhalation routes. Hydrazine fuels form carcinogenic nitrosamine compounds. The hydrazines are classified as animal carcinogens by the Registry of Toxic Effects of Chemical Substances issued by NIOSH, 1981-82, and listed as Industrial Substances Suspected of Carcinogenic Potential to Man by the American Conference of Government Industrial Hygienists, 1982.

If spilled on the skin or splashed in the eyes, liquid hydrazine can cause severe local damage or burns and can cause dermatitis. In addition, it can penetrate skin to cause systemic effects similar to those produced when the compound is swallowed or inhaled. If inhaled, the vapor causes local and systemic effects. On short exposure, systemic effects involve the central nervous system. Resultant symptoms include tremors; on exposure to higher concentrations, convulsions and possibly death follow. Repeated exposures may cause toxic damage to the liver (fatty liver) and kidney (interstitial nephritis), as well as anemia.

Monomethylhydrazine is a volatile caustic liquid which can cause systemic toxicity through cutaneous absorption, as well as by inhalation. It is irritating both to the skin and pulmonary tract since it is alkaline.

Diethylenetriamine (DETA), also known as one of the major constituents in MAF causes burns in contact with skin or eyes. Repeated skin contact with DETA may lead to skin sensitizations while an asthmatic type response may result from inhalation of the vapors. Due to the low vapor pressure of DETA it is not a significant contributor as a toxic agent in mixed amine fuels via the inhalation route.

B4.2.1.2 Exposure Limits

B4.2.1.2.1 Threshold Limit Values-Time Weighted Average (TLV-TWA) (References 18 and 19).

Hydrazine: The TLV-TWA is 0.1 ppm (0.1 mg/m^3), Skin warning, Suspected carcinogen. The IDLH is 80 ppm (80 mg/m^3).

MMH: The TLV-TWA is 0.2 ppm (0.35 mg/m^3), Ceiling value, Skin warning, Suspected carcinogen. The IDLH is 5 ppm (8.8 mg/m^3)

UDMH: The TLV-TWA is 0.5 ppm (1 mg/m^3), Skin warning, Suspected carcinogen. The IDLH is 50 ppm (100 mg/m^3).

DETA: TLV-TWA is 1.0 ppm (4 mg/m^3) with skin warning.

B4.2.1.2.2 Emergency Exposure Limits - The emergency exposure limit (EEL) defines the "single brief accidental exposures to air-borne contaminants that can be tolerated without permanent toxic effects. These limits are not intended to replace accepted safe practices and should be accompanied by appropriate medical surveillance" (Reference 17).

Hydrazine:

10 min. 30 ppm (39 mg/m^3)
30 min. 20 ppm (26 mg/m^3)
60 min. 10 ppm (13 mg/m^3)

MMG:

10 min. 90 ppm (158 mg/m^3)
30 min. 30 ppm (53 mg/m^3)
60 min. 15 ppm (26 mg/m^3)

UDMH:

10 min. 100 ppm (or less) (2490 mg/m^3)
30 min. 50 ppm (125 mg/m^3)
60 min. 30 ppm (75 mg/m^3)

B4.2.1.3 Special Medical Information

B4.2.1.3.1 Hydrazine - If convulsions occur, quick acting barbiturates administered only by direction of a physician, with due regard for synergistic depression of respiration, should help. Induction of diuresis will aid in the elimination of hydrazine from the body (See References 20 through 51).

B4.2.1.3.2 Monomethylhydrazine - Symptomatology following exposure is unknown in man and is quite varied in animals. Central Nervous Symptom (CNS) symptoms ranging from tremors to convulsions are seen in all species but at widely varying doses. The degree of kidney damage and methemoglobin production found at convulsant doses varies with species. Therefore, following human exposure, any or none of the following may occur: CNS convulsions, methemoglobinemia, kidney proximal tubule damage, and anemia. The possibility of deranged carbohydrate metabolism resulting in a hypoglycemia should also be considered.

Pyridoxine may be of value in treatment of the convulsions. If the victim becomes hypoglycemic or develops a methemoglobinemia, treat using accepted practices. CNS depressants should be used with care (References 21, 22, and 52 through 62).

B4.2.1.3.3 UDMH (References 63 through 75) - Barbiturates and muscle relaxants may be of value in treatment of the convulsions. Pyridoxine has been shown to abort convulsions in animals exposed to UDMH. UDMH is rapidly excreted in the urine and elimination can be hastened by the use of diuretics.

B4.2.1.3.4 Mixtures - If a mixture of hydrazines such as Aerozine-50 (UDMH-Hydrazine) is spilled on the skin, the exposure should be regarded primarily as one to hydrazine. The volatility of the two compounds favors greater absorption of the hydrazine into the skin. Caution may be necessary if pyridoxine is used under these conditions since it is of little value in hydrazine toxicity. If 25 mg/kg (2.5×10^{-5} lb/lb) pyridoxine does not abort convulsions, further doses may provoke rather than help the condition.

If a spill of the mixture of hydrazines is the cause of toxicity via the inhalation route, UDMH toxicity is of primarily concern and pyridoxine is the drug of choice. If four doses of 25 mg/kg (2.5×10^{-5} lb/lb) pyridoxine at two hour intervals fail to abort the symptoms, further administration of the compound may actually be of no use or even detrimental.

B4.2.2 Fire and Combustion Product Hazards

Hydrazine, MMH, UDMH and DETA are hypergolic with oxidants, such as hydrogen peroxide, nitrogen tetroxide, fluorine, halogen fluorides, and nitric acid. A film of hydrazine or MMH in contact with metal oxides, such as those of iron, copper, lead, manganese, and molybdenum, may ignite owing to the heat of chemical reaction. Rags, cotton waste, sawdust, excelsior, or other materials of a large surface area that have absorbed hydrazine, MMH, UDMH, or MAF may eventually cause spontaneous ignition. Such materials should not be stored under conditions that prevent dissipation of the heat that can accumulate by the gradual process of air oxidation. In enclosed spaces all personnel must be evacuated when the atmosphere reaches 10 percent of the lower explosive limit (LEL).

B4.2.3 Explosion Hazards

Vapors of hydrazine, MMH, and UDMH can be exploded by an electric spark or by an open flame. Due to high vapor pressure and a wide flammability range, the possibility of an explosive mixture forming over UDMH or MAF is very high. Hydrazines and MAF should be stored and handled at all times under an atmosphere of nitrogen.

B4.2.4 Environmental Effects

The hydrazines and their respective oxidation by-products such as the amines and the oxides of nitrogen (See Section B4.1.3.4) are toxic and injurious to plant and lower animal life if present in sufficient concentration. If dissolved in natural water, the hydrazines would deplete the oxygen supply causing serious effect on the aquatic ecosystems.

B4.3 RESERVED

B4.4 RESERVED

B4.5 MATERIALS AND EQUIPMENT COMPATIBILITY (Reference 78)

B4.5.1 Materials

Hydrazines are a highly reactive propellant and are considered thermodynamically unstable. The rate of decomposition of purified hydrazine is a function of both temperature and presence of a catalyst. At ambient temperatures, 294 K (70°F), and in the absence of a catalyst. The decomposition of hydrazine is minimal.

B4.5.1.1 Metals - Some metals to show excellent corrosion resistance to hydrazine. Some contaminants, (e.g., carbon dioxide, chloride ion) reduce the corrosion resistance of metals. For long-term storage of uncontaminated hydrazine, the major concern is the degree that the metal being considered accelerates the N_2H_4 decomposition rate.

The following metals may be used with hydrazine, MMH, UDMH, Aerozine-50, H-70, and MAF:

Stainless steel - types 304,I(1,2); 304L,I(1); 316,I(1,2); 317(2); 347(1), 355(1,2); 17-4PH(2); 17-7PH (Carpenter 20 Cb[1,2]); 410, 416(2); 430(1,2); A-286.
Aluminum alloys - types 1100(2); 300(2); 2014(1,2),R; 2017(2); 2024, 2219, 4043, 5052, 5456(2); R; 6061(1,2); 6066, 7075, 716, 356, 40E(2).
Miscellaneous - Chromel A, Chromium Plating, Monel(2); Stelite-21, Tantalum, Tin, Titanium 5Al-2.5, Sn(2), Titanium 6Al-4V(4) and Zirconium.

Unsatisfactory metals are:

Cadmium(2)
Cobalt
Copper and its alloys, Brass(4), Bronze(5)
Aluminum (ASME-40E)
Hasteloys
Iron
Lead
Magnesium
Zinc

Molybdenum(4)
Mild Steel(4)
Nickel

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(1) (Reference 80), based on 10-year tests
(2) (Reference 81), A restriction code, R, is used where restricted compatibility exists. A restriction code, I, indicates an insufficient amount of data (Reference 80). This notation is after the appropriate reference listing.
(3) (Reference 82), 20 Dec 78, Chapter 4.
(4) Based on one reference, this metal showed excessive decomposition at 316.5 K(110°F), with 50/50 N_2H_4/UDMH.
(5) The authors (2) stated that these metals were considered unacceptable because their oxides act as catalysts for decomposition of hydrazine at elevated temperatures.

B4.5.1.2 Non-Metals - The attack of storage materials is usually considered a problem for non-metals, most of which show poor corrosion resistance to neat N_2H_4. For long-term storage with non-metals, both catalytic and material attack must be considered.

The following non-metals may be used:

AF-E-332(1),I
High density polyethylene (tested below 344 K [160°F] exposure less than 4 weeks)
Hydropol OT plastic or equivalent
Tetrafluoroethylene (TFR, Halon TFE, Teflon, or equivalent)
Butyl rubber (2) (tested below 368 K) (95°F)
Ethylene propylene rubber

The following non-metals indicate poor compatibility (2):

Viton unplasticized chlorotrifluoroethylene polymer (KEL F, Halon CTF, or equivalent)

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(1) (Reference 80), based on 10-year tests
(2) (Reference 81), A restriction code, R, is used where restricted compatibility exists. A restriction code, I, indicates an insufficient amount of data (Reference 80). This notation is after the appropriate reference listing.

B4.5.1.3 Lubricants - A completely satisfactory lubricant has not been developed for use with these propellants due to their solvent properties. The following materials have been used:

Krytox PR-240AC
Apiezon L
Quigley "Q" seal
Dow Corning silicon compound 11
Fluorolube GR-470
Kel-F grease
Reddy Lube 200

Based on Reference 80 several preferred lubricants include:

Propellant	Lubricant
Hydrazine	Varnation Oyseal Dow Corning DC-11
UDMH and MMH	Mixture of Apiezon L and M Graphite, Versilube F-50

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point (760 Torr)		236.3°F	386.8 K
Freezing Point		35°F	275 K
Density	(Gas)		
	(Liquid)	8.415 lb/gal at 68°F	1.008 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)		1.1 at 32°F	1.1 at 273.15 K
Critical Density		1.93 lb/gal	0.2313 Mg/m^3
Critical Pressure		2135 psia	14.7 MPa
Critical Temperature		716°F	653 K
Vapor Pressure		0.07 psia at 40°F	0.483 kPa at 277.6 K
		0.28 psia at 77°F	1.92 kPa at 298 K
		0.31 psia at 80°F	2.14 kPa at 299.8 K
		1.04 psia at 120°F	7.17 kPa at 322 K
		2.9 psia at 160°F	20 kPa at 344 K
Coefficient of Viscosity	Kinematic	0.952 centistokes at 77°F	9.52×10^{-7} m^2/s at 298 K
	Absolute	0.96 centipoises at 68°F	9.6×10^{-4} Pa s at 293 K
Automdecomposition Temperature (air at 1 atm)	Rust Surface Glass Surface	435°F 518°F	497 K 543 K
Flash Point	Open Cup Closed Cup	100 - 125.6°F 100°F	310 - 325 K 310.9 K
Flammability Limits	Lower: Upper:	4.7 volume %in air at 212°F (373 K) 100 volume %in air at 212°F (373 K)	

Figure B4-1 Physical Properties of Hydrazine (MW = 32.05 g mol)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		189.5°F	360.7 K
Freezing Point		-65°F	210 K
Density	(Gas)		
	(Liquid)	7.334 lb/gal at 68°F	0.8788 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)		1.59 at 32°F	1.59 at 293 K
Critical Density		2.42 lb/gal	0.29 Mg/m^3
Critical Pressure		1194 psia	8.24 MPa
Critical Temperature		593.6°F	585 K
Vapor Pressure		0.31 psia at 40°F	2.14 kPa at 277.6 K
		1.0 psia at 80°F	6.9 kPa at 299.8 K
		3.1 psia at 120°F	21.4 kPa at 322 K
		7.9 psia at 160°F	54.5 kPa at 344 K
Coefficient of Viscosity	Kinematic	0.97 centistokes at 77°F	9.7×10^{-7} m^2/s at 298 K
	Absolute	0.85 centipoises at 68°F	8.5×10^{-4} Pa s at 293 K
Autoignition Temperature (air at 1 atm)		382°F	467.6 K
Flash Point	Closed Cup Open Cup	17°F 34°F	265 K 274 K
Flammability Limits	Lower: Upper:	2.5 volume %in air 98 volume %in air	

Figure B4-2 Physical Properties of MMH (MW = 46.07 g mol)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		146°F	336.5 K
Freezing Point		-71°F	215.9 K
Density	(Gas)		
	(Liquid)	6.6 lb/gal at 77°F	0.791 Mg/m^3 at 298 K
Specific Gravity - Vapor (relative to STP air)		2.1 at 32°F	2.1 at 273.15 K
Critical Density			
Critical Pressure		786 psia	5.42 MPa
Critical Temperature		482°F	523 K
Vapor Pressure		0.3 psia at 0°F	2.07 kPa at 255.4 K
		1.0 psia at 40°F	6.90 kPa at 277.6 K
		3.1 psia at 80°F	21.4 kPa at 299.8
		8.4 psia at 210°F	57.9 kPa at 372 K
Coefficient of Viscosity	Kinematic	0.65 centistokes at 77°F	6.5×10^{-7} m^2/s at 298 K
	Absolute	0.56 centipoises at 68°F	5.6×10^{-4} Pa s at 293 K
Autoignition Temperature		482°F	523 K
Flash Point	Open Cup Closed Cup	5°F 34°F	258 K 274 K
Flammability Limits	Lower: Upper:	2 volume %in air 90 volume %in air	

Figure B4-3 Physical Properties of UDMH (MW = 60.11 g mol)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Points	N_2H_4(1) UDMH(1)	235°F 146°F	386 K 336 K
Freezing Point(2)		18.8°F	266 K
Density	(Gas)		
	(Liquid)(2)	56.1 lb/ft^3	0.899 Mg/m^3
Specific Gravity - Vapor (relative to STP air)			
Critical Density			
Critical Pressure (cal c)		1696 psia	11.7 MPa
Critical Temperature (cal c)		634°F	607.6 K
Vapor Pressure(3)		2.75 psia at 77°F	18.97 kPa at 298 K
Coefficient of Viscosity	Kinematic		
	Absolute(2)	0.817 centipoise at 77°F	8.17 x 10^{-4} Pa s at 298 K
Autoignition Temperature			
Flash Point			
Flammability Limits	Lower: Upper:		

(1) Fuel blend is not a constant boiling mixture

(2) Measured on samples of the fuel blend of typical composition (51.0%) N_2H_4, 48.5%UDMH, and 0.5%H_2O

(3) Fuel blend composition (51.0%N_2H_4, 48.4%UDMH, and 0.6%H_2O)

Figure B4-4 Physical Properties of Aerozine-50 Fuel Blend (Average MW 45.0 g mol)

CHAPTER B5
NITROMETHANE

B5.1 PROPERTIES

B5.1.1 Identification

Nitromethane has a chemical formula of CH_3NO_2. It is commercially available in grades from 95 to 99% purity (References 1, 2). The molecular weight of nitromethane is 61.04 g mol.

B5.1.2 General Appearance and Common Uses

Nitromethane is an oily, colorless liquid with a mild fruity odor. It is used as a solvent, fuel additive and propellant.

B5.1.3 Physical and Chemical Properties

The physical properties of nitromethane are listed in Figure B5-1.

B5.1.3.1 Solubility - Nitromethane is slightly soluble in water: 9.5g/100g water at 298 K (68°F). It sinks and mixes slowly with water. It is soluble in alcohol.

B5.1.3.2 Stability - Conditions contributing to the instability of nitromethane are overheating of closed containers, and exposure to heat or flame. Nitromethane may detonate when shocked.

B5.1.3.3 Nitromethane is incompatible with amines, strong acids and alkalies which may sensitize it so that it will readily explode. Contact with strong oxidizers may cause fires and explosions. Mixtures of nitromethane and hydrocarbons are highly flammable. Contact with some metallic oxides may cause decomposition and development of pressure.

Special precaution should be taken with liquid nitromethane since it will attack some forms of plastics, rubber and coatings.

B5.1.3.4 Environmental Fate - Nitromethane that evaporates into the atmosphere would be absorbed in atmospheric moisture and washed down in a rainfall. In soil environments, the solubilized nitromethane would be biodegraded. In surface waters, the diluted nitromethane solution could also be biodegraded. See Section B5.2.4.

B5.2 HAZARDS

B5.2.1 Health Hazards

B5.2.1.1 Toxicity - Contact with liquid nitromethane may cause irritation of the skin or eyes. Breathing large amounts of the vapor may cause respiratory irritation or breathing difficulty. In cases of overexposure, anesthesia or convulsions may result. It may also cause anorexia, nausea, vomiting, and diarrhea. Based on animal exposures, kidney, liver and spleen damage are also possible. On decomposition nitromethane emits highly toxic fumes of oxides of nitrogen. It is not considered to be carcinogenic (Reference 3).

B5.2.1.2 Exposure Limits - The threshold limit value-time weighted average (TLV-TWA), is 100 ppm (250 mg/m^3) and the IDLH value is 1000 ppm (2500 mg/m^3) (Reference 4, 5, 6 and 7) and the IDLH value is 1000 ppm (2500 mg/m^3). The IDLH value is "immediately dangerous to life or health," meaning that it is the maximum level from which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects (Reference 5). The TLV-STEL is 150 ppm (375 mg/m^3).

B5.2.2 Fire and Combustion Product Hazard

Nitromethane poses a moderate fire hazard when exposed to heat or flame (Reference 8). It has an autoignition temperature of 691 K (784°F) and a flash point of 308 K (95°F). Contact with oxidizing materials may cause fire or explosion. A nitromethane spill is a fire hazard and a persistent source of toxic vapor. Toxic gases such as oxides of nitrogen and carbon monoxide are released in fires involving nitromethane.

B5.2.3 Explosion Hazards

Nitromethane poses a moderate explosion hazard and may be initiated by shock, heat, or flame. Its vapors have a lower explosion limit of 7.3 percent. Overheating in a closed container may cause detonation. Amines, strong acids, and alkalies may sensitize it to initiation of an explosion. Contact with metal oxides may cause decomposition resulting in pressure buildup. It can also react violently with $AlCl_3$ mixed with organic matter, $Ca(OH)_2$ $Ca(OCl)_2$, hexamthylbenzene, hydrocarbons, inorganic bases, hydroxides, organic amines, KOH and NaOH. (Reference 8).

B5.2.4 Environmental Effects

Nitromethane in an equeous waste, would be biodegradable in a biological wastewater treatment facility, or as part of natural wastewater treatment purification capabilities. The bio-oxidation of nitromethane under aerobic conditions would be expected to produce carbon dioxide, nitrates and water, all harmless byproducts that naturally occur in surface water systems. This biological degradation would be expected provided the amount spilled does not exceed the capabilities of the systems.

B5.3 RESERVED

B5.4 RESERVED

B5.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B5.5.1 Materials

Nitromethane in itself is readily capable of detonation or of explosive decomposition or explosive reaction at normal temperatures and pressure (Reference 9). Also, contact with some metallic oxides may cause decomposition and development of pressure (Reference 3, 12, 14 and 15).

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		214°F	374.3 K
Freezing Point		-20.2°F	244 K
Density	(Gas)		
	(Liquid)	9.35 lb/gal at 77°F	1.12 Mg/m^3 at 298 K
Specific Gravity - Vapor (relative to STP air)		1.139	1.139
Critical Density			
Critical Pressure		915.1 psia	6.311 MPa
Critical Temperature		599°F	588 K
Vapor Pressure			
Coefficient of Viscosity	Kinematic		
	Absolute		
Autoignition Temperature		784.4°F	691 K
Flash Point	(Closed cup)	95°F	308 K
Flammability Limits	Lower: Upper:	7.3 percent by volume in air	

Figure B5-1 Physical Properties of Nitromethane (MW = 61.04 g mol)

CHAPTER B6
ETHYLENE OXIDE

B6.1 PROPERTIES

B6.1.1 Identification

Ethylene oxide (Oxirane, 1,2-epoxyethane) is essentially 100 percent C_2H_4O (Reference 1). The molecular weight of ethylene oxide is 44.05 g mol.

B6.1.2 General Appearance and Common Uses

Ethylene oxide is a gas at room temperature and a clear, colorless, mobile liquid at lower temperatures. It has a characteristic ether-like odor that is irritating in high concentrations. Ethylene oxide will volatilize rapidly at atmospheric temperatures and pressures. When used with a catalyst and as a monopropellant, it is a gas generator.

B6.1.3 Chemical and Physical Properties

Refer to Figure B6-1 for the physical properties of ethylene oxide.

B6.1.3.1 Solubility - Ethylene oxide is completely soluble in water at 283 K (50°F).

B6.1.3.2 Stability - Liquid ethylene oxide is stable when pure, but the vapor will explode when exposed to common igniters. See Section B6.1.3.3 below.

B6.1.3.3 Reactivity - Ethylene oxide gas is flammable in air in all proportions above three percent by volume. It is a reactive material, forming compounds with water, alcohols, amines and organic and mineral acids. Ethylene oxide can violently polymerize exothermally in the presence of such materials as pure anhydrous chlorides of iron, tin and aluminum, oxides of iron, aluminum and magnesium, the alkali metal hydroxides and weak acids. Before it becomes nonflammable, liquid ethylene oxide must be diluted with a quantity of water which is 22 times it volume.

B6.1.3.4 Environmental Fate - Ethylene oxide is a colorless gas at room temperature. The general mode of entry into the environment could be either by air or water; however, its major transportation mode would be by air. In air, the lifetime of ethylene oxide would be very short, as it is rapidly degraded by hydroxyl radicals in the atmosphere. Ethylene oxide in surface waters would be quickly hydrolyzed to ethylene glycol.

B6.2 HAZARDS

B6.2.1 Health Hazards

B6.2.1.1 Toxicity - The gas will cause eye and nasal irritation when present in excessive amounts. The liquid or solutions on the exposed skin do not cause skin irritation immediately, but when spilled in the shoes or on the clothing, delayed skin burns can occur if contaminated clothing and shoes are not promptly removed. The liquid or solutions may cause severe eye burns. Contact has been known to produce skin irritation and burns from its absorption by perspiration in areas of moisture and heat about the body. When excessive amounts of ethylene oxide are inhaled, they have a general anesthetic effect as well as causing coughing, due to irritation of the respiratory system. The victim may become nauseated and vomit. "High concentrations can cause pulmonary edema" (Reference 2).

B6.2.1.2 Exposure Limits

B6.2.1.2.1 Threshold Limit Value-Time Weighted Average (TLV-TWA) (References 3, 4, 5 and 6) - The TLV-TWA is 1 ppm (2 mg/m^3). The IDLH is 800 ppm (1600 mg/m^3). Ethylene oxide is now listed as a suspected carcinogen.

B6.2.1.2.2 Emergency Exposure Limits (Reference 6) - The emergency exposure limit (EEL) defines the "single brief accidental exposures to air-borne contaminants that can be tolerated without permanent toxic effects. These limits are not intended to replace accepted safe practices and should be accompanied by appropriate medical surveillance" (Reference 6).

60 minutes 250 ppm (500 mg/m^3)
30 minutes 400 ppm (800 mg/m^3)
10 minutes 650 ppm (1250 mg/m^3)

B6.2.1.3 Special Medical Information - The reactions and injuries resulting from skin, eye and systemic ethylene oxide exposures are nonspecific. Treatment is based on clinical evaluation of the patient and the system affected. Contradictory evidence from animal data suggests the possibility of liver and kidney damage.

B6.2.2 Fire and Combustion Product Hazards

Ethylene oxide has a flash point below 255.4 K (0°F) and an autoignition temperature of 702 K (804°F). It is flammable at any concentration above 3 percent in air. Because of the wide flammability limits and relatively high vapor pressure of the material, flammable mixtures will be produced by spills at all prevailing temperatures. Firefighting personnel must be provided with respiratory and eye protection because of the toxicity and volatility of the material.

Ethylene oxide can react with oxidizing materials. Ignition of ethylene oxide vapor may be prevented by diluting the vapor with gases such as nitrogen to point below the flammable limit. Such a gas must be free of impurities such as air, acetylene, sulfur, hydrogen sulfide, water or ammonia. Ignition from static sparks, fire and heat must be guarded against by adequate electrical insulation, heat insulation, approved (regulating authority) bonding and grounding, and automatic water-spray systems. Fire fighting should be done from an explosion-resistant location. Water should be used from unmanned monitors or hose-holders to keep fire-exposed containers cool.

B6.2.3 Explosion Hazards

Liquid ethylene oxide is not sensitive to mechanical shock, but its vapor explodes when exposed to an electric spark, static electricity, heat, or an open flame. Mixtures of vapor and air are more explosive than vapor alone thus storage tanks and equipment must be kept free from air.

Ethylene oxide may decompose and/or polymerize violently when in contact with active catalytic surfaces, such as anhydrous chlorides of iron, tin, and aluminum, oxides of iron and aluminum, metallic potassium, alkali metal hydroxides, acids and organic bases. Above 302.6 K (85°F) the reactions are accelerated and may result in an explosion.

B6.2.4 Environmental Effects

A static test of ethylene oxide at 40 kg/m^3 (2.49 lb/ft^3) for 96 hours showed no adverse effects on fish. Ethylene glycol, its hydrolysis product, in a similar test at 100 kg/m^3 (6.24 lb/ft^3) showed no adverse effects. Ethylene oxide has a low partition coefficient and is therefore probably not bioaccumulative. The rate of reaction of ethylene oxide in the environment is so rapid, that exposures and effects are considered low level.

B6.3 RESERVED

B6.4 RESERVED

B6.5 MATERIALS AND EQUIPMENT COMPATIBILITY

Ethylene oxide is a very flammable gas at normal conditions. Ethylene oxide is in itself capable of explosive reaction but requires a strong initiating source, or must be heated under confinement before initiation. It is also sensitive to thermal shock at elevated temperatures and pressures. Contact with oxidizing materials may result in fires (Reference 16). Metal fittings containing copper, silver, or mercury or magnesium should not be used in ethylene oxidize service, since traces of acetylene could produce explosive acetylides capable of detonating ethylene oxide vapor (References 18, 19).

B6.5.1 Materials

B6.5.1.1 Metals - Containers for the storage and transfer of ethylene oxide may be constructed of any of the following acceptable metals:

a. Mild steel, properly protected to prevent rust formation.

b. Pure aluminum (99.6% pure or better)

c. Stainless steel (316, 420, 440)

The following metals must not be used with ethylene oxide:

a. Stainless steel (416 and 442)

b. Copper and copper alloys

c. Magnesium and magnesium alloys

d. Silver and silver alloys

e. Cast iron

B6.5.1.2 Non-Metals - Non-metals recommended for use with ethylene oxide are as follows:

a. When temperature does not exceed 344 K (160°F)

(1) Glass

(2) Chlorotrifluoroethylene (Kcl-F), Halon CTF or equivalent

(3) Tetrafluoroethylene (Teflon) TFE, Halon TFE or equivalent

b. For periods of short duration or intermittent use, and at ambient temperatures only:

(1) Buan N synthetic rubber

(2) Nylon

(3) Polyvinyl butyral

B6.5.1.3 Lubricants - Fluorinated hydrocarbon lubricants should be used for ethylene oxide equipment. Petroleum-based lubricants are not recommended.

B6.5.2 Equipment

B6.5.2.1 Containers - Storage tanks for ethylene oxide should be constructed of mild steel, properly protected, stainless steel or pure aluminum. Tanks should be lagged with non-combustible insulation. Glass wool is the preferred insulation material.

B6.5.2.2 Pumps and Hose - Centrifugal pumps, gravity flow, or inert pressurizing gas may be used when the tanks are equipped with bottom outlets. A minimum flow bypass valve on the pump discharge line should be provided.

Commercially available vaporizers may be used for producing gaseous ethylene oxide. The pump should be explosion proof.

Hoses constructed of materials listed in section B6.5.1 may be used.

B6.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flashlights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 20).

B6.5.2.4 Pipes and Fittings - All pipes and fittings should be made of approved materials (See Section B6.5.1), and must be hydrostatically tested for a working pressure of at least 1.034 MPa (150 psig). All pipe connections should be either welded or flanged, the flange being welded to the pipe. Threaded connections should be avoided, especially where directly exposed to liquid ethylene oxide. Liquid ethylene oxide lines should be insulated to prevent condensation of air or excessive heat transfer. Ethylene oxide piping should be clearly identified (Reference 18).

B6.5.2.5 Gaskets - Gaskets may be made of chlorotrifluoroethylene (Kel-F), tetrafluoroethylene (Teflon) TFE, Halon CTE or TFE, or equivalent materials (See Section B6.5.1.2).

B6.5.2.6 Pressure Gauges - Approved-type pressure gauges should be of compatible materials. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B6.5.2.7 Valves - The valves should be fabricated from approved construction materials (See Section B6.5.1) and should be tested for at least 1.034 MPa (150 psig). The valves may be of gate, globe, plug, or ball type; if either gate or globe type valves are used, the stem packing material must be compatible with ethylene oxide.

B6.5.2.8 Venting Systems and Pressure Relief - All vents and pressure-relief systems should be extended to the atmosphere and should be protected by approved flame arresters. Each vent's termination should be at a height and location that will give adequate protection to personnel and buildings; they should be sized according to the specifications given in National Fire Codes (Reference 13, 14 and 15).

B6.5.2.9 Grounding - All vehicles, propellant containers, missiles, filling hoses, electrical equipment, pumps, etc., should be grounded to safely discharge static accumulations. The resistance to grou [illegible] ould be checked and logged regularly.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		51°F	283.7 K
Freezing Point		-170 - -168°F	160.9 - 162 K
Density	(Gas)		
	(Liquid)	7.4 lb/gal at 51°F	0.887 Mg/m^3 at 283.7 K
Specific Gravity - Vapor (relative to STP air)		1.5	1.5
Critical Density		2.7 lb/gal	0.32 Mg/m^3
Critical Pressure		1042 psia	7.19 MPa
Critical Temperature		383°F	468 K
Vapor Pressure		4.2 psia at 0°F	29 kPa at 255.4 K
		11.5 psia at 40°F	79.3 kPa at 277.6 K
		27.0 psia at 80°F	186 kPa at 299.8 K
		55.0 psia at 120°F	379 kPa at 322 K
Coefficient of Viscosity	Kinematic	0.31 centistokes at 51°F	3.1 x 10^{-7} m^2/s at 283.7 K
	Absolute	0.28 centipoises at 51°F	2.8 x 10^{-4} Pa s at 283.7 K
Autoignition Temperature (air at 1.0 atm)		804°F	702 K
Flash Point (Open Cup)		0°F	255.4 K
Flammability Limit (includes explosive range where applicable)		3-100% by volume in air at 68°F	60-100 g/m^3
Decomposition Temperature (pure vapor at 1.0 atm)		1060°F (14.7 psia)	844 K (101 kPa)

Figure B6-1 Physical Properties of Ethylene Oxide (MW = 44.05 g mol)

CHAPTER B7
OTTO FUEL II

B7.1 PROPERTIES

B7.1.1 Identification

Otto Fuel II is a stable, liquid monopropellant composed of a nitrate ester in solution with a desensitizing agent and a stabilizer. The chemical composition and specifications for Otto Fuel II are listed in Table B7-1 and MIL-0-82672)(OS) (Reference 1 and 2).

Table B7-1 Chemical Requirements (Moisture-Free Basis)

COMPONENT	REQUIREMENTS MINIMUM (% by wt)	MAXIMUM (% by wt)
Propylene glycol dinitrate	75.8	76.2
2-Nitrodiphenylamine	1.4	1.6
Di-n-butyl sebacate	22.2	22.8
Sodium		0.8

B7.1.2 General Appearance

Otto Fuel II is a bright-red, free-flowing oily liquid which is heavier than water. When in a thin layer (i.e., spill, stain, or leak), Otto Fuel is a yellow-orange color.

B7.1.3 Chemical and Physical Properties

Refer to Figure B7-1 for physical properties.

B7.1.3.1 Solubility - Otto Fuel II is insoluble in water for all practical purposes. It is also insoluble in ethylene glycol and propylene glycol. It is slightly soluble in fuel oil, heptane, kerosene and petroleum ether. Otto Fuel II is soluble in acetone, alcohols, benzene, carbon tetrachloride, chloroform, dibutyl pthalate, gasoline, methylene chloide, toluene, and trichloroethylene.

B7.1.3.2 Stability - Otto Fuel II is thermally stable at temperatures up to 339 K (150°F) for several years, up to 355 K (180°F) for a few months, and up to 394 K (250°F) for perhaps 30 minutes. Above 394 K (250°F) there is danger of exothermic decomposition.

At temperatures exceeding 416 K (290°F) rapid exothermic decomposition will occur with container rupture and fire as a very likely consequence. The storage life of Otto Fuel II is dependent upon time and temperature as shown in Table B7-2. A sample of Otto Fuel II has been held at a temperature of 323 K (122°F) for over seven years with no sign of deleterious effects. A "safe life" in excess of 100 years, based on high temperature stability studies, has been predicted for Otto Fuel II held at 323 K (122°F).

B7.1.3.3 Reactivity - Otto Fuel II is a noncorrosive liquid monopropellant with an extremely low vapor pressure which minimizes the explosive and toxic hazards usually associated with other monopropellants. Otto Fuel has a high flash point and other safety characteristics which permit it to be classified as a relatively low fire hazard material.

In developing the monopropellant, the ingredients were selected to minimize the possibility of the components being separated and leaving a possibly hazardous material. For example, all components are insoluble in water so that a spill cleaned up with water does not leave an explosive residue. The components have essentially the same vapor pressure; the vapor pressure of the energy containing constituent being slightly higher than that of the desensitizer (di-n-butylsebacate) and stabilizer (2-nitrodiphenylamine). The energetic ingredient would therefore be the first to evaporate on heating and would leave a completely stable residue.

It has been proven that although Otto Fuel can be made to detonate, the conditions and stimulus required are so extreme that it is considered a nonexplosive, relatively low fire hazard, for shipping and storage purposes. Otto Fuel II is routinely shipped by commercial carriers as a nonregulated item, chemical NOIBN (Not Otherwise Indexed By Name).

B7.1.3.4 Environmental Fate - Otto Fuel is a stable liquid that is insoluble in water. Its transport in the environment in water would be expected to be limited, due to this hydrophobicity. In a land spill, the Otto Fuel would probably absorb into clays and biological particles rather than leach downward. In a surface water spill, the Otto Fuel would be sorbed into bottom sediments. In small concentrations, the Otto Fuel should be biodegradable both in surface water and in the soil (Reference 3).

B7.2 HAZARDS

B7.2.1 Health Hazards

Otto Fuel II is relatively safe to handle. The ingredient of medical concern in Otto Fuel II is the nitrated ester, propylene glycol dinitrate (PGDN). Nitrated esters are known for their acute effects on the human body (Reference 4).

Table B7-2 Safe Life of Otto Fuel II at Various Temperatures

STORAGE TEMPERATURE	MINIMUM SAFE LIFE (DAYS)
364 K (195°F)	24
355 K (180°F)	112
347 K (165°F)	500
339 K (150°F)	2,256 (6 yrs)
330 K (135°F)	10,000 (27 yrs predicted)
323 K (122°F)	37,500 (102 yrs predicted)

B7.2.1.1 Toxicity - Toxic effects may occur from the inhalation of Otto Fuel II vapors, absorption from direct skin contact or ingestion. Severity of these effects may vary with the concentration, time of exposure, and temperature of the propellant. OTTO Fuel II causes vasodilatation and may produce low levels of methemoglibin (Reference 4).

Intoxication results in a characteristically intense, throbbing headache, presumably owing to cerebral vasodilatation, often associated with dizziness and nausea, and occasionally with vomiting and abdominal pain. More severe exposure also causes hypertension, flushing palpitation, low levels of methemoglobinemia, delirium, and depression of the central nervous system. Aggravation of these symptoms after alcohol ingestion has been observed. Upon repeated exposure, a tolerance to headache develops but is usually lost after a few days without exposure. Ingestion can cause death or at least severe disorders of the gastrointestinal tract, mucous membranes, and severe nausea (Reference 2).

B7.2.1.2 Exposure Limits

B7.2.1.2.1 Threshold Limit Values - The TLV is 0.05 ppm (0.3 mg/m^3) for propylene glycol dinitrate, the energetic component of OTTO Fuel. The TLV-STEL is 0.1 ppm (0.6 mg/m^3) (References 5, 6, 7 and 8) (skin warning for HCN), (Reference 6). In concentrations above 100 ppm (110 mg/m^3) it can be absorbed through the skin. Measured concentrations of HCN in Otto Fuel operational areas are normally below 10 ppm (Reference 9).

B7.2.1.3 Special Medical Information - In the event of spills solvents must never be used to cleanse Otto Fuel II from the skin because they speed absorption into the skin and accelerate and magnify the effects of the exposure.

B7.2.2 Fire and Combustion Product Hazards

Otto Fuel II is classified as a relatively low fire hazard by military service regulations. Attempts to burn Otto Fuel II in bulk at atmospheric pressure and temperatures under 394 K (250°F) have been unsuccessful; however, a finely dispersed spray of Otto Fuel II can be ignited in an oxygen-containing atmosphere. Furthermore, the propellant may be ignited in bulk when heated above 402 K (264°F). When porous or absorbent materials, such as paper, rags, fiberglass, are present to act as a wick, Otto Fuel II can be easily ignited at ambient temperatures. The by-products from the combustion of Otto Fuel II include carbon monoxide (CO), carbon dioxide (CO_2), hydrogen (H_2), methane (CH_4, and hydrogen cyanide (HCN). The CO, CO_2, and HCN gases present toxic hazzards to personnel.

B7.2.3 Explosion Hazards

Otto Fuel II can be detonated when a sufficiently strong booster is employed. However, an exhaustive series of tests including drop tests, projectile impact tests, bullet impact tests, and card gap tests have resulted in its classification as a nonexplosive (References 10 and 11). Under the recommended storage conditions, Otto Fuel II presents no explosive hazard. However, when highly confined (as in heavy-walled steel containers with no pressure-relief device) it will mass detonate if a sufficiently strong booster is employed, or if it is heated above its decomposition temperature. Tanks must be equipped with pressure relief devices to prevent internal pressure buildup over 414 kPa (60 psig).

B7.2.4 Environmental Effects

The Otto Fuel is a nitrated ester with a stabilizer and a desensitizer. In the environment, a simple nitrated ester would be biodegradable. However the other constituents of the Otto Fuel may cause its somewhat persistence. Biodecomposition products of the nitrated ester would include nitrates, which are very mobile anions in soil-water systems, and hydrocarbons, which are biodegradable. Larger spills into surface waters could have serious effects on the aquatic life, including the fish.

B7.3 RESERVED

B7.4 RESERVED

B7.5 MATERIALS AND EQUIPMENT COMPATIBILITY

Otto Fuel II must be stored in fixed or mobile drums or tanks of approved design and construction and made with properly selected materials (Reference 12).

B7.5.1 Materials

B7.5.1.1 Metals - Most common metals may be used with Otto Fuel II; however, long-term contact with copper-based metals is to be avoided.

B7.5.1.2 Non-Metals - The following non-metals are approved for Otto Fuel II usage:

a. Teflon-impregnated asbestos
b. Fluorocarbons
c. Polyethylene
d. Glass
e. Natural rubber
f. Butyl rubber
g. Ethylene-propylene rubber

The following materials should be avoided:

a. Viton - excessive swelling, but not hazardous
b. Buna-N - excessive swelling, but no hazard
c. Neoprene - swells slightly, but no hazard
d. Polysulfide rubber - chemically incompatible

B7.5.1.3 Lubricants - Most fluorocarbon-based and silicon-based lubricants may be used with Otto Fuel II, petroleum-based lubricants are to be avoided.

B7.5.1.4 Paint - Epoxy paints appear to offer the greatest protection against Otto Fuel II. Any epoxy paint conforming to MIL-P-22808A (Reference 13) is recommended. To ensure satisfactory service, any previously untested paint should be subject to tests with Otto Fuel II prior to use. A minimum of three coats of fully cured epoxy paint has been found to provide adequate protection for concrete surfaces.

Enamel, water-base latex, and varnish should be avoided. Although they do not react chemically with Otto Fuel II, the fuel does remove the paint finish.

B7.5.2 Equipment

Otto Fuel II can be stored in either fixed or mobile drums or tanks of approved design, materials, and construction; and suitably housed. All storage areas should be equipped with impervious flooring and must be kept neat and free from any trash and combustibles. These must be inspected frequently and safety regulations must be strictly enforced. All leaks and spills must be cleaned up at once.

B7.5.2.1 Containers - Storage tanks should be of high-quality welded construction, made of compatible materials, and should be properly tested for leaks prior to service. The tanks must be equipped with pressure relief devices to prevent internal pressure build-up over 414 kPa (60 psig).

Copper-based alloys are not approved for tank construction.

B7.5.2.2 Pumps and Hose - When pumps are used in the Otto Fuel II system, only pumps and shaft materials suitable for this service should be used. In general, ensure cooling oil used in stator housing is compatible with Otto Fuel II. Ensure that internal seals will not allow contaminants to bypass filter element.

Hose should also be compatible with Otto Fuel II and constructed of approved materials listed in Sections B7.5.1.1 and B7.5.1.2.

B7.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 14).

B7.5.2.4 Pipes and Fittings - Pipes and fittings should be made of approved materials and should be pressure tested prior to use. Flanged, welded AN-type connections are suitable. Threaded connections are acceptable but not recommended.

B7.5.2.5 Gaskets - Teflon-impregnated asbestos, Kel-F, Teflon, polyethylene, butyl rubber, and silicon rubber may all be used in Otto Fuel II gaskets. Polysulfide rubber is to be avoided.

B7.5.2.6 Pressure Gauges - Standard-type pressure gauges can be used in Otto Fuel service. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B7.5.2.7 Valves - Valves must always provide positive closure; ball, plug or the balanced poppet designs are recommended. All wetted parts of the valve must be of approved materials. Once used, valves must not be used for transfer or storage of oxidizers.

B7.5.2.8 Venting Systems and Pressure Relief - In a pressure system only, aluminum piping segments should be installed on each side of the transfer pump as detonation traps. These "soft" sections in the system should provide protection against detonation propagation. Design of these sections will be such that they conform in size to the remainder of the system but with wall thickness sufficiently thin to insure that these sections will, in fact, be a "weak link."

B7.5.2.9 Grounding - All electrical equipment should be properly grounded.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point			
Freezing Point (1)		-18.4°F	245 K
Density (Liquid)		10.28 lb/gal at 77°F	1.23 Mg/m^3 at 298 K
Specific Gravity (relative to water = 1)		1.23	
Critical Density		N/A	
Critical Pressure		N/A	
Critical Temperature		N/A	
Vapor Pressure		0.0877 mm Hg 77°F (1.7×10^{-3} psia)	11.7 Pa at 298 K
Surface Tension		6.33×10^3 poundal/in at 77°F	34.45 dynes/cm at 298 K
Water Saturation Point		0.31% at 77°F	
Coefficient of Viscosity	Kinematic	3.285 centistokes at 77°F	3.285×10^{-6} m^2/s at 298 K
	Absolute	4.04 centipoised at 77°F	4.04×10^{-3} Pas at 298 K
Autoignition Temperature			
Flash Point (Cleveland Open Cup)		265°F	403 K
Flammability Limits	Lower: Upper:		

(1) Depending on water content, data given for sample containing 0.1% of water.

Figure B7-1 Physical Properties of Otto Fuel II

CHAPTER B8
HYDROGEN (AND DEUTERIUM)

Compressed hydrogen and deuterium safety should incorporate the fire and explosion hazards discussed below and the precautions given in CPIA 394, Vol. III, Appendix F-29 for compressed gases. Liquified deuterium is similar to liquid hydrogen in its safety considerations.

B8.1 PROPERTIES

Knowledge of the general properties of liquid hydrogenis of value to all personnel concerned with its handling or storage.

B8.1.1 Identification

The hydrogen molecule (H_2) (molecular weight: 2.016 g mol) exists in two forms, ortho and para, depending on the relative direction in which the nuclei of the atoms spin. As ortho hydrogen changes to para-hydrogen, heat is released. This heat would increase venting requirements, but current use of catalysts at production facilities has eliminated the problem by producing almost pure para-hydrogen in liquid form. This is commonly called equilibrium hydrogen (eH_2). Equilibrium liquid hydrogen is 99.79 percent para-hydrogen and 0.21 percent ortho-hydrogen. Military Specification MIL-P-27201B covers liquid hydrogen (Reference 1). Deuterium (D_2 or D) (molecular weight: 4.032 g mol) is a stable isotope of hydrogen with an atomic weight of 2. One atom of deuterium is found mixed with 6000 ordinary hydrogen atoms.

B8.1.2 General Appearance and Common Uses

High-purity liquid hydrogen is a transparent, colorless, odorless liquid. When observable, it is usually boiling vigorously because of its low boiling point, and when exposed to the atmosphere it creates a voluminous cloud of condensed water vapor. Hydrogen is used as a rocket fuel, for welding, for production of hydrochloric acid, and for the reduction of metallic ores.

B8.1.3 Physical and Chemical Properties

The physical properties of hydrogen is given in Figure B8-1 (References 2, 3, 4, 5, 6).

B8.1.3.1 Sclubility - All known substances are essentially insoluble in liquid hydrogen. Helium is slightly soluble (about 1 percent in pressurized liquid storage tanks).

B8.1.3.2 Stability - Liquid hydrogen is chemically stable. Because of its low boiling point, it is physically stable only when stored under suitable conditions. When stored in properly designed containers, the 24 hour evaporation rate may be as low as 1.5 percent, or less, for a 3.784 m^3 (1000 gal) container.

B8.1.3.3 Reactivity - Liquid hydrogen is non-corrosive. It will form combustible mixtures with oxidizers. Hydrogen gas is combustible with air over a wide range of mixtures. Liquid or gaseous hydrogen will ignite spontaneously with either liquid or gaseous fluorine and chlorine trifluoride.

B8.1.3.4 Environmental Fate - There are no concerns in this area.

B8.2 HAZARDS

B8.2.1 Health Hazards

The low temperature of liquid hydrogen may cause frostbite when the liquid, or uninsulated piping containing it, contacts the skin. In the gaseous form, hydrogen acts as a simple asphyxiant. It can be breathed in high concentrations without producing systemic effects. However, if the concentration is high enough to significantly reduce the amount of oxygen normally present in the air, the effects of oxygen deprivation will be produced (Reference 7).

B8.2.2 Fire and Combustion Product Hazards

A fire hazard always exists when hydrogen gas is present. Hydrogen-air mixtures containing as little a 4 percent or as much as 75 percent hydrogen by volume are readily ignited. Hydrogen-oxygen mixtures are flammable over the range of 4 to 94 percent hydrogen by volume. When no impurities are present, hydrogen burns in air with an invisible flame. In enclosed spaces, all personnel must be evacuated when the atmosphere reaches 10 percent of the LEL (lower explosive limit - same as lower limit of flammability).

B8.2.3 Explosion Hazards

Unconfined hydrogen-air mixtures generally burn rapidly but without detonation, when initiated by heat, spark, or flame. Since hydrogen diffuses rapidly by air, it will not form persistent flammable mixtures when the liquid evaporates in open, unconfined areas. However, in confined areas, or when ignition of the hydrogen-air mixture is caused by a shock source equivalent to a blasting cap or small explosive charge, the mixture can detonate.

An explosion hazard can exist if liquid hydrogen is contaminated with solid oxygen or solidified oxygen-enriched air, formed by exposure of the liquid hydrogen to air or oxygen.

B8.2.4 Environmental Effects

There are no concerns in this area.

B8.3 RESERVED

B8.4 RESERVED

B8.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B8.5.1 Materials

The ability of materials to retain satisfactory physical properties at liquid hydrogen temperatures and to withstand thermal stresses caused by large temperature changes is of prime importance. Where hydrogen is used, special equipment complying with pertinent industrial standards must be provided. Buildings where cylinders of hydrogen are stored should be isolated and used solely for that purpose when practicable.

B8.5.1.1 Metals - The ferrous alloys, except for the austenitic-nickel-chromium alloys, lose their ductility when subjected to the low temperatures of liquid hydrogen and become too brittle for this service. Metals used for this service are as follows:

- stainless steel (300 and other austenitic series)
- copper
- bronze
- brass
- Monel
- aluminum, pure
- Everdur (copper alloy)
- Inconel 286

B8.5.1.2 Non-Metals - Nonmetal materials must also be selected to withstand the low temperature of liquid hydrogen. Non-metals found suitable for this service are as follows:

- polyester fiber (Dacron or equivalent)
- tetrafluoroethylene (TFE, Halon TFE, Teflon, or equivalent)
- unplasticized chlorotrifluoroethylene (Kel F, Halon CTF, or equivalent)
- asbestos impregnated with TFE
- Mylar or equivalent
- nylon

B8.5.1.3 Lubricants - Lubricants are generally not practical in the presence of liquid hydrogen, for they solidify and become brittle at liquid hydrogen temperature. Vacuum grease is satisfactory as a sealant with "O" rings. Based on (Reference 11). Molybdate may be used with liquid hydrogen.

B8.5.2 Equipment

B8.5.2.1 Containers - Liquid hydrogen containers should be of approved design and materials.

B8.5.2.2 Pumps and Hose - Only pumps and shaft seals specifically designed and qualified by test for use with liquid hydrogen should be used. Hoses should be of a proper design and engineered specifically for liquid hydrogen service.

B8.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 12).

B8.5.2.4 Pipes and Fittings - Pipes and fittings should be of approved materials and construction and should be hydrostatically tested at prescribed pressures. Welded as well as welded flanged connections are recommended. Eutectic brazing procedures are satisfactory with stainless steel. Threaded connections should be avoided, especially where directly exposed to liquid hydrogen temperatures. Liquid hydrogen lines should be insulated to prevent condensation of air or excessive heat transfer. All lines in which liquid hydrogen may be trapped between closed valves should be equipped with safety relief devices.

B8.5.2.5 Gaskets - Gaskets may be made of the material listed in Sections B8.5.1.1 and B8.5.1.2, depending upon the application.

B85.2.6 Pressure Gauges - Liquid hydrogen systems should be monitored with approved types of pressure gauges, as required. Clean standard gauges equipped with blowout backs and plastic face plates and with proper bourdon tube material are acceptable. In order to minimize operator reading errors, all pressure gauges used for common purpose should have identical scales.

B8.5.2.7 Valves - Valves must conform to particular specifications, for their use with liquid hydrogen. Extended stem, globe-type or fate-type valves are recommended, but plug-type or ball-type valves may also be used. It must be possible to purge valves efficiently. Also, they must have an adequate packing design to provide good sealing and to prevent plugging or air condensation.

B8.5.2.8 Venting Systems and Pressure Relief - In general, vessels which contain liquids (or gases of low boiling point) should be individually relieved. However, where two or more vessels are connected by piping without valves the group may be treated as a whole. Relief valves should be connected into the vapor space. The relief valve setting should be such as to protect the vessel or vessels involved and should be sized for operating emergencies or exposure to fire. Common practice is to vent through an approved flare stack.

B8.5.2.9 Grounding - Since hydrogen-air and hydrogen-oxygen mixtures are highly flammable, all stationary or mobile tanks should be bonded and grounded to prevent ignition by static electricity. The ground resistance should be monitored regularly.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-423°F	20.4 K
Freezing Point		-435°F	13.7 K
Density	(Gas)		
	(Liquid)(1)	0.59 lb/gal at -423°F	0.07077 Mg/m^3 at 20.4 K
Specific Gravity - Vapor (relative to STP air)		0.07 at 32°F	0.07 at 273.15 K
Critical Density(2)		0.262 lb/gal	0.03142 Mg/m^3
Critical Pressure		188 psia	1.30 MPa
Critical Temperature		-400°F	33 K
Vapor Pressure		1.9 psia at -433°F	13.1 kPa at 14.8 K
		14.7 psia at -423°F	101 kPa at 20.4 K
		23.7 psia at -420°F	163 kPa at 22 K
		120 psia at -405°F	827 kPa at 30.4 K
		163 psia at -402°F	1.12 kPa at 32 K
Coefficient of Viscosity	Kinematic	0.20 centistokes at -423°F	2.0×10^{-7} m^2/s at 20.4 K
	Absolute	0.014 centipoises at -423°F	1.4×10^{-5} Pa s at 20.4 K
Autoignition Temperature (air at 1 atm)		1075°F	853 K
Flash Point			
Flammability Limits at 68°F (293°K)	Lower:	4 percent by volume in air (3.2 g/m^3)	
	Upper:	75 percent by volume in air (60 g/m^3)	

Figure B8-1 Physical Properties of Hydrogen (MW = 2.016 g mol)

PROPERTY	ENGLISH UNITS	SI UNITS
Volume Expansion (Liquid)	B.P. to gas at 70°F, 840:1	
Vapor Density(3)	0.083 lb/ft^3 at -423°F	1.3295 kg/m^3 at 20.4 K
Gas Density(4)	0.0056 lb/ft^3	8.97×10^{-2} kg/m^3

(1) Density decreases to 53.9 kg/m^3 (0.45 lb/gal) at 30.4 K (-405°F) where vapor pressure is 827.6 kPa (120 psia)
(2) See Reference 5
(3) 1.02 times heavier than air at 273 K (32°F)
(4) 14.5 times lighter than air at 273 K (32°F)

Figure B8-1 Physical Properties of Hydrogen (MW = 2.016 g mol)(continued)

CHAPTER B9
THE BORANES

B9.1 PROPERTIES

B9.1.1 Identification

Diborane (B_2H_6) is also known as diboron hexahydride and boroethane. The propellant grade contains approximately 99 percent by weight diborane; the remainder is dissolved hydrogen and higher boron hydrides. No military specification covers this grade. The molecular weight of diborane is 27.67 g mol.

Pentaborane (B_5H_9) is also known as pentaboron anhydride, and pentaborane-9. The propellant grade contains 99.0 percent by weight pentaborane and 1.0 percent dissolved solids (maximum). Military Specification MIL-P-27403, (Reference 1) "Propellant, Pentaborane," covers this grade. The molecular weight of pentaborane is 63.13 g mol.

B9.1.2 General Appearance

Diborane is a colorless gas at room temperature and a pressure of one atmosphere, and it has a characteristic pungent odor.

Pentaborane is a clear colorless liquid and has a characteristic pungent odor.

B9.1.3 Physical and Chemical Properties

B9.1.3.1 Solubility - Diborane is completely hydrolyzed in seconds when the gas is brought into contact with water. Diborane is soluble without reaction in dry hydrocarbon solvents such as pentane and hexane. With oxygenated or halogenated solvents, diborane is soluble but may form shock-sensitive mixtures.

Pentaborane is soluble without reaction in hydrocarbon solvents such as kerosene, hexane, benzene, and toluene. With oxygenated or halogenated solvents, pentaborane is soluble but forms shock-sensitive mixtures. It is insoluble in, but reacts with, water.

B9.1.3.2 Stability - In the absence of air or contaminants, diborane is stable indefinitely at 193 K (-112°F). At room temperature, diborane decomposes slowly. Hydrogen inhibits diborane decomposition. At temperatures above 573 K (572°F), diborane decomposes rapidly into its elements. It reacts readily with water and forms a mist of boric acid when in contact with moist air. Diborane, by itself, is insensitive to shock.

In the absence of air or contaminants, pentaborane is stable at room temperature. It decomposes at 423 K (302°F), but not explosively. Small amounts of oxygen or moisture will cause solid deposits to form in pentaborane and metal oxides affect its stability. Pentaborane, by itself, is insensitive to shock.

B9.1.3.3 Reactivity - Diborane is a strong reducing agent and reacts violently with oxidizers. It is pyrophoric in air. Diborane hydrolyzes rapidly in contact with water. It is hypergolic with hydrazine and oxygen difluoride. Diborane is toxic both as a liquid and as a vapor.

Pentaborane will react with any organic compound containing a reducible functional group. It may react explosively or form shock-sensitive solutions with highly halogenated or oxygenated solvents such as carbon tetrachloride, trichlorethylene, acetone and other ketones, aldehydes and carbon disulfide. it reacts with hydrazine and other amines. it is pyrophoric in air. Pentaborane hydrolyzes slowly in water at room temperature but at elevated temperatures it can be hydrolyzed completely. The addition of dioxane or alcohols greatly enhances the hydrolysis.

B9.1.3.4 Environmental Fate - Diborane exists as a gas, and pentaborane a liquid at room temperatures. Both diborane and pentaborane are pyrophoric in air. As atmospheric pollutants, therefore, only borate and other oxidized boron forms would be expected. Although there are occupational suggested limits on boron oxide and borates in the workplace, there are no ambient air quality standards as such. As atmospheric pollutants, however, boron compounds would be expected to wash-out and be returned to land in precipitation events. Di and pentaborane rapidly hydrolyze to borates and hydrogen in water. The hydrogen would probably vaporize into the atmosphere, and the borates would react with substances in the water.

B9.2 HAZARDS

B9.2.1 Health Hazards

B9.2.1.1 Toxicity - Diborane, pentaborane, and related boranes are highly toxic by inhalation, ingestion, and skin and eye contact.

B9.2.1.2 Exposure Limits

B9.2.1.2.1 Threshold Limit Values-Time Weighted Averages and Short Term Exposure Limits (Reference 3, 4 and 5)

diborane(TWA)	0.1 ppm (0.1 mg/m^3)
diborane(IDLH)	40 pppm (40 mg/m^3)
pentaborane(TWA)	0.005 ppm (0.01 mg/m^3)
pentaborane(STEL)	0.015 ppm (0.03 mg/m^3)
pentaborane(IDLH)	3 ppm (6 mg/m^3)

Threshold limit values for all of the alkyl boranes have not been established; in view of their high toxicity, it is likely that any values set (TLV-TWA) will be very low, i.e., on the order of 0.01 parts per million (ppm). Appropriate precautions must be taken to prevent cutaneous absorption.

B9.2.1.2.2 Emergency Exposure Limits (Reference 6)

Diborane	10 minutes - 10 ppm (10 mg/m^3)
	30 minutes - 5 ppm (5 mg/m^3)
	60 minutes - 2 ppm (2 mg/m^3)
Pentaborane	5 minutes - 25 ppm (50 mg/m^3)
(Reference 7)	15 minutes - 8 ppm (16 mg/m^3)
	30 minutes - 4 ppm (8 mg/m^3)
	60 minutes - 2 ppm (4 mg/m^3)

No emergency exposure limits have been set for other related boranes.

B9.2.1.3 Special Medical Information (References 8 through 16) - In acute intoxication, symptoms usually occur promptly after exposure but can be delayed. Initial symptoms are lightheadedness with loss of memory and may be followed by headaches and nausea. Nervousness, muscle tightness, muscle tremors, or convulsions, depending on the degree of exposure, may occur. A reserpine-like depletion of neurotransmitter amines is seen in animals which show an initial hypertension followed by hypotension. The signs of central nervous system (CNS) depression sometimes seen as a sequel to pentaborane poisoning in humans may result from a similar mechanism.

Depressants of the polysynaptic pathways in the spinal cord (such as methocarbamol) and general CNS depressants (such as barbiturates and tranquilizers) may be used to control the peripheral muscular CNS manifestations . Supportive care should be given as indicated, keeping in mind the possibility of a reserpine-like activity similar to the reaction seen in animals. The onset of symptoms is usually delayed and can occur up to 24 hours after exposure. Frontal headache and lightheadedness is followed by general signs of CNS depression such as drowsiness, lethargy, and complete fatigue.

B9.2.2 Fire and Combustion Product Hazards

Diborane is a very flammable gas with extremely wide combustion limits. It is flammable at concentrations between 0.8 and 98 percent by volume in air and has an autoignition temperature of 311 to 325 K (100 to 126°F). Due to contaminants or moisture, it should be considered pyrophoric at about 293 K (68°F). Below this temperature, diborane may not be spontaneously flammable, but may be ignited by a static spark, heat of reaction, or heat of absorption. "Contact with aluminum, and other active metals form hydrides which ignite spontaneously. Diborane reacts with many oxidized surfaces as a strong reducing agent" (Reference 17). Fires involving diborane may produce toxic gases and vapors such as boron oxide smoke (Reference 17).

Pentaborane is a highly flammable compound and, for all practical considerations, is pyrophoric. Its flash point is 303 K (86°F) (pure material in closed cup). Impurities cause it to ignite spontaneously in air. It has an autoignition temperature of 308 K (95°F) for the pure material, and a lower flammable limit of 0.42 percent by volume in air.

Boranes burn more intensely and are more dangerous than burning hydrocarbon fuels. Combustion can be sustained over a wide range of conditions.

B9.2.3 Explosion Hazards

If a significant quantity of diborane becomes mixed with air before spontaneous ignition takes place, as in the case of a leak, there will be an explosive ignition followed by quiet burning.

Pentaborane is usually pyrophoric; therefore, explosive concentrations of vapor are not normally present in air. The decomposition of pentaborane in storage containers liberates hydrogen which will cause pressure build-up in the containers if not relieved. Excessive pressure build-up should be relieved with care to avoid causing a fire or explosion. Pentaborane forms highly explosive mixtures by reaction with oxidizers. Explosions may be caused by contact with halogens or halogenated compounds (Reference 17).

B9.2.4 Environmental Effects

In natural surface waters, borates form complexes with major metal cations in the water. Boron is background in freshwater and seawater in concentrations up to 5 g/m^3 (3.1×10^{-4} lb/ft^3) typically as a sodium or calcium borate salt. Boron is an essential element for growth of plants, although long-term irrigation would show some damage to sensitive crops. A suggested criterion for freshwater to be used for long-term irrigation of sensitive crops is 0.75 g/m^3 (4.7×10^{-5} lb/ft^3). The maximum lethal dose for minnows exposed to boric acid at 293 K (68°F) for 6 hours was 19 to 20.2 kg/m^3 (1.12 to 1.19 lb/ft^3).

B9.3 RESERVED

B9.4 RESERVED

B9.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B9.5.1 Materials

It is difficult to predict the compatibility of materials under all conditions and it is recommended that preliminary tests be conducted prior to their use. Materials should retain satisfactory physical requirements when stored in "dry ice" (i.e., Solid carbon dioxide) or at temperatures greater than 423 K (302°F). It should be noted that diborane reacts with many oxidized surfaces as a strong reducing agent. Pentaborane reacts with oxidizers to form highly explosive mixtures. Contact of diborane and pentaborane with air or halogenated compounds will cause fires and explosions. Both may ignite spontaneously in air (References 26, 27). Materials are listed below that have proven satisfactory under ordinary conditions.

B9.5.1.1 Metals - The following metals and alloys are recommended for use with boranes:

a. Diborane:

brass	low carbon steel
nickel	stainless steel
K-Monel	series 18-8
Monel	stainless steel
lead	series 300
copper	

b. Pentaborane (These alloys may be anodized.):

Aluminum types,	5052-S
	6061-T6
	7075-T6
	2024-T3
	3003-H14
	356-T6
	Cadmium coated (Reference 26)
	Chromated (Reference 26)
Stainless steel types,	302 (Reference 26)
	304 (Reference 26)
	321 (Reference 26)
	Series 18-8
Titanium alloys,	Rem-Cru No. C-110AM
	Rem-Cru No. C-130AM
Magnesium alloys,	Fed-QQ, M-44A
	Fed-QQ, M-56A763 (Reference 26)
	Fed-QQ, M-46A263

Pure aluminum
Low - carbon steels
Cadmium plated steel (Reference 26)
Brass
Chromium - (Reference 27)
Copper cobalt alloy
Iron
Nickel, nickel alloy
Hastelloy No. X-1258 (Reference 26)
K - Monel
Monel M-8330-B, Soft
Nichrome "V"

B9.5.1.2 Non-Metals - The following non-metals have been used successfully with boranes:

a. Diborane:

Saran
Viton A, Fluorel, or equivalent
asbestos-graphite (Garlock or equivalent)
tetrafluoroethylene (TFE, Halon TFE, Teflon, or equivalent)
polychlorotrifluoroethylene (Kel F, Halon CTF, or equivalent)

b. Pentaborane:

polychlorotrifluoroethylene (Kel F, Halon CTF, or equivalent)
fluorosilicone rubbers, Neoprene rubber
tetrafluoroethylene (TFE, Halon TFE, Teflon, or equivalent)
glass
pure carbon
Fluoroflex T or equivalent
Viton A and B or equivalent
Garlock 230 or equivalent
molybdenum disulfide (Reference 26)

B9.5.1.3 Lubricants - The following lubricants may be used with boranes:

a. Diborane:

perfluorocarbon lubricants

b. Pentaborane:

graphite (Reference 27)
perfluorocarbon lubricants
Graphitar No. 39
Rockwell Nordstrom Lube No. 921
Gulf Harmony Oil Nos. 44 and 69
Hercules No. 571 Kaobestos

B9.5.1.4 Prohibited Materials - The use of the following materials with boranes is not recommended:

a. Diborane:

all rubbers

b. Pentaborane:

water-based lubricants
natural rubber
butyl rubber
GR-S rubbers
Garlock Silicone Rubber No. 9383
Buna rubber
nitride rubber or nylon
Mylar
Tygon
Saran
neoprene
silicones
Fiberfrax Nos. XSW, SLF
Rubatex No. 5. G-207-N, R-103-J
Karoseal, Viton A
Dow Corning Foam R-7007, R-7003
Nopco Foam F-10, B-49

Pittsburgh - Corning Foamglas
Dow Corning No. 916
Dow Corning Silastic Nos. 50-24-480, 80-24-480
Garlock Silastic No. 250
Rockwell Nordstrom Lube Nos. 833, P-21, 860, 386, 852-5, P-55, 942-S
Johns-Manville Packing Nos. 2008, C-255
Swedlow Plastic X5G-146-101
Kel-F 5500, 3700 (Reference 27)
vinylidine plastics
epoxy cements
graphite and carbon with binders

c. The following is a partial list of solvents which will form shock-sensitive mixtures with pentaborane:

chloroform	trichlorethane
dioxane	special fluorinated solvents
acetone	trichlorethylene
aldehydes	halogenated compounds
ketones	
carbon tetrachloride	

B9.5.2 Equipment

B9.5.2.1 Containers - Boranes may be stored in either shipping cylinders or storage tanks. The storage tanks are to be designed to the latest requirements of the ASME Code (Reference 28).

a. Diborane is a liquified compressed gas and should be kept at a miximum temperature of 253 K (-4°F) to prevent decomposition.

b. Pentaborane storage tanks should be designed and fabricated in accordance with at least a 10 percent volumetric allowance for ullage. A well must be provided at the bottom of each storage tank to permit almost complete drainage.

B9.5.2.2 Pumps and Hoses

a. Diborane: There is no acceptable pump available at present for diborane. Tetrafluoroethylene (TFE) or polychlorotrifluoroethylene (CTFE) lines or all-metal bellows-type hoses may be used.

b. Pentaborane: Canned rotor or centrifugal pumps with double mechanical seals or packed seals of graphite, TFE, or CTFE impregnated asbestos are acceptable. TFE or CTFE lines or all-metal bellows-type hoses may be used. Hoses of the metal interlocking type should be avoided.

Storage and handling area should be provided with personnel emergency showers, eye baths, fire blankets, portable fire extinguishers, first aid kits, and a water deluge system, preferably of the fog type. Safety equipment should be strategically located and easily accessible. All operating personnel should be thoroughly familiar with the location and operation of each piece of safety equipment. The operation of the equipment should be verified periodically. A reliable borane detector should be used to help in monitoring storage and drainage collection points. An adequate water supply should be provided for flushing, decontamination, and fire fighting.

B9.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flashlights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 29).

B9.5.2.4 Pipes and Fittings - Pipes and fittings will be fabricated of approved material and installed in accordance with pertinent codes. All transfer lines should slope in one direction to prevent the accumulation of boranes in places that cannot be easily drained. Nitrogen taps should be provided in the transfer system as a means of completely purging the system.

Flared, welded, or flanged fittings are preferred. Threaded pipe fittings should be avoided as possible sources of fuel leakage. All joints and fittings should be accessible for pressure and leak tests. Piping and equipment systems should be electrically bonded and grounded. Piping containing boranes should be clearly identified.

B9.5.2.5 Gaskets - Gaskets and O-rings should be fabricated of approved materials listed in Section B9.5.1.2.

B9.5.2.6 Pressure Gauges - Pressure gauges must be of approved materials and should have solid case front, blow-out back, and be provided with a surge shut-off. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B9.5.2.7 Valves - The primary considerations for selection are that the valves be leakproof and made of compatible materials. Non-lubricated valve designs acceptable for use with other toxic and corrosive liquids may be used with boranes. Valves used with boranes must be thoroughly cleaned, inspected, and tested for leaks prior to installation and use.

Particle migration is a problem whenever valve parts rub, turn, or wedge on plastic sealing materials such as polychlorotrifluoroethylene or tetrafluoroethylene. This can cause problems in seal life such as plugging of minute orifices and instruments and fouling of close-tolerance fits. Migration is a particular problem in butterfly and soft-seated gate valves.

Plug, needle, and globe valves have been employed satisfactorily in borane service.

B9.5.2.8 Venting Systems and Pressure Relief - Vent gases containing borane fumes should be flared or scrubbed with a 3 to 5 percent ammonia solution before being released to the atmosphere. For safety relief, any high quality valve with good relief-reset characteristics, constructed of approved materials, may be used. As an additional relief device, rupture discs in parallel with the valve are recommended.

B.9.5.2.9 Grounding - Since borances are highly flammable, all stationary or mobile tanks should be bonded and grounded to prevent fuel ignition bystatic electricity. The ground resistance should be monitored regularly.

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CHAPTER B10
CARBON MONOXIDE

B10.1 PROPERTIES (References 1,2)

B10.1.1 Identification

The chemical formula of carbon monoxide is CO. It is formed as a by-product of combustion and is toxic. It is available commercially in five grades ranging from 98.0 mole % to 99.8 mole % as a nonliquified gas. The molecular weight of CO is 28.01 g mol.

B10.1.2 General Appearance and Common Uses

Carbon Monoxide is a colorless, odorless gas that has no warning properties to alert the human body of its presence. The largest amount of carbon moniside is industrially utilized in fuel gas mixtures with hydrogen and other gases for heating. It is also used in the manufacture of a variety of chemicals.

B10.1.3 Physical and Chemical Properties

The physical properties of carbon monoxide are listed in Figure B10-1.

B10.1.3.1 Solubility - Carbon monoxide is soluble at STP in water at 3.5 g/100ml at 273 K (32°F) and 1.5 g/100ml at 333 K (140°F).

B10.1.3.2 Stability - Carbon monoxide is stable with respect to decomposition into carbon and oxygen. At 673 to 973 K (752 to 1292°F) almost any surface is sufficiently active to cause disproportionation. It oxidizes to carbon dioxide in ambient air.

B10.1.3.3 Reactivity - At temperatures of 573 to 1773 K (572 to 2732°F) carbon monoxide reduces many metal oxides (like those of cobalt, copper, iron, lead, manganese, molybdenum, nickel, silver and tin) to lower metal oxides, metals or metal carbides. Contact with strong oxidizers may cause fires and explosions. Carbon monoxide reacts with steam to give carbon dioxide and hydrogen.

B10.1.3.4 Environmental Fate - Carbon monoxide emitted into the environment is ultimately oxidized to carbon dioxide.

B10.2 HAZARDS

B10.2.1 Health Hazards

B10.2.1.1 Toxicology - Inhalation may cause headache, nausea, dizziness, weakness, rapid breathing, unconsciousness and death. High concentrations may be fatal without producing significant warning symptoms. Exposure to carbon monoxide may aggravate heart disease and artery disease, and may cause chest pains in those with heart disease. Pregnant women are more susceptible to its effects. The effects of exposure are more severe when involved in heavy labor, at high temperatures, or at altitudes above 610 m (2,000 ft). Skin exposure to liquid carbon monoxide may produce frostbite.

Carbon monoxide is eliminated through the lungs when air free form CO is inhaled. Over half the CO is eliminated in the first hour, when the exposure has been moderate.

Concentrations up to 10 percent of CO-hemoglobin in the blood rarely cause noticeable symptoms. Concentrations of 20 to 30 percent cause shortness of breath on moderate exertion and slight headache. Concentrations from 30 to 50 percent cause severe headache, mental confusion and dizziness, impairment of vision and hearing, and collapse and fainting on exertion. With concentrations of 50 to 60 percent, unconsciousness results, and death may follow if exposure is long. Concentrations of 80 percent result in almost immediate death. For additional information see SAX, 1981 (Reference 3).

B10.2.1.2 <u>Exposure Limits</u> - Carbon monoxide has a threshold limit volume (TLV) of 50 ppm (55 mg/m^3) (References 4, 5 and 6) and STEL of 400 ppm (440 mg/m^3). NIOSH gives 35 ppm (44 mg/m^3) as the 10-hour time weighted average (TWA) value and a 200 ppm ceiling. The IDLH of 1500 pm (1875 mg/m^3) value is "immediately dangerous to life or health," meaning that it is the maximum level from which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects (Reference 7).

B10.2.1.3 <u>Special Medical Information</u> - Carbon monoxide can enter the body by inhalation or by contact with the skin when in liquid form. The organs affected are the cardiovascular system, lungs, blood, and the central nervous system. Symptoms include headache, tachypnea, nausea, weakness, dizziness, confusion, hallucination, cyanosis, depression, ST of electrocardiogram, angina, syncope, and frostbite (for liquid) (Reference 2).

B10.2.2 <u>Fire and Combustion Product Hazard</u>

Carbon monoxide has an autoignition temperature of 882 K (1128°F). Its flammable limits are 12.5 percent to 74 percent in air. It is a fire hazard when exposed to flame, but does not have hazardous composition products.

B10.2.3 <u>Explosion Hazard</u>

Reference 3 categorizes CO as posing a severe explosion hazard when it is exposed to heat or flame. On contact with strong oxidizers a fire or explosion may result.

B10.2.4 <u>Environmental Effects</u>

Environmental effects of carbon monoxide are limited to areas of high concentration where it can pose a threat to animal life. It is one of the ambient pollutants used to indicate pollution alerts, since excess ambient levels impose a strain on human respiration. The effects are greater on persons with impaired respiratory sytems and among the elderly.

B10.3 <u>RESERVED</u>

B10.4 <u>RESERVED</u>

B10.5 MATERIALS AND EQUIPMENT COMPATIBILITY (References 1, 11, 12, 13, 14, 15, 17)

B10.5.1 Materials

Carbon monoxide, a very flammable gas, should be stored in tanks or cylinders of approved design and construction, made with properly selected materials and suitably housed. (See 49 CFR 178.36, 178.37, 178.42 adn 178.45, and Section B10.9.2.1.)

Conditions contributing to instability consist of elevated temperatures which may cause cylinders to explode (NIOSH). Contact with strong oxidizers may cause fires and explosions

No other information is available at this time.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-312.7°F	81.65 K
Freezing Point		-326.2°F	74 K
Density	(Gas) at b.p.	0.272 lb/ft^3	4.36 kg/m^3
	(Liquid) at b.p.	49.41 lb/ft^3	0.7915 Mg/m^3
Specific Gravity - Vapor (relative to STP air)		0.9678 at 70°F and 14.7 psia	0.9678 at 294 K and 101 kPa
Critical Density		19.13 lb/ft^3	0.3064 Mg/m^3
Critical Pressure		504.6 psia	3.48 MPa absolute
Critical Temperature		-220.4°F	132.9 K
Vapor Pressure		14.7 psia at -312.8°F	101 kPa at 81.6 K
Coefficient of Viscosity	Kinematic	0.0219 centistokes at 32°F and 14.7 psia	2.1 x 10^{-8} m^2/s at 273 K and 101 kPa
	Absolute	0.0166 centipoise at 32°F and 14.7 psia	1.66 x 10^{-5} Pas at 273 K and 101 kPa
Autoignition Temperature		1128°F	882 K
Flash Point			
Flammability Limit	Lower: Upper:	12.5% by volume in air 74% by volume in air	

Figure B10-1 Physical Properties of Carbon Monoxide (MW = 28.01 g mol)
(Reference 1)

CHAPTER B11
OXYGEN

B11.1 PROPERTIES

B11.1.1 Identification

Propellant grade liquid oxygen (LOX or LO_2) contains a minimum of 99.5 percent oxygen (O_2). The Marshall Space Flight Center requirement for space vehicle grade oxygen is 99.6 percent minimum. The major impurity is argon. Military Specification MIL-P-25508E "Propellant, Oxygen," and MSFC SPEC-339 Grade 1, 2, or 3 (References 1 and 2) covers propellant grade oxygen. The molecular weight of oxygen is 32 g mol.

B11.1.2 General Appearance and Common Use

High purity liquid oxygen is a light-blue transparent liquid. It boils vigorously at ambient conditions and uninsulated containers are usually frosted. Oxygen has no odor. It is used as a cryogenic rocket propellant oxidizer.

B11.1.3 Physical and Chemical Properties (Reference 3)

The major physical properties of oxygen are listed in Figure B11-1.

B11.1.3.1 Solubility - Most common solvents are solid at liquid oxygen temperatures. Liquid oxygen is completely miscible with liquid nitrogen and methane. Light hydrocarbons are usually soluble in liquid oxygen; acetylene is soluble only to about 4 parts per million.

B11.1.3.2 Stability - Liquid oxygen is chemically stable. It is not shock-sensitive and will not decompose. At ordinary temperatures in properly designed containers, the 24-hour evaporation rate may be as low as 1.4 percent from a 1.7 m^3 (450 gal) container, 0.4 percent from a 5.1 m^3 (1,350 gal) container.

B11.1.3.3 Reactivity - In either gaseous or liquid form, oxygen is a strong oxidizer which vigorously supports combustion. The violence of some reactions involving liquid oxygen is due to the highly reactive oxygen-rich atmosphere surrounding the liquid.

B11.1.3.4 Environmental Fate - Oxygen becomes part of the atmosphere.

B11.2 HAZARDS

B11.2.1 Health Hazards

The health hazards of liquid oxygen are associated with its very low temperature. The low temperature of liquid oxygen may cause frostbite when the liquid, or uninsulated piping containing it, contacts the skin. Oxygen gas will not cause toxic effects in propellant operations, except that inhalation of very cold oxygen gas may cause some irritation to the upper respiratory tract.

B11.2.2 Fire and Combustion Product Hazards

Liquid oxygen does not burn but vigorously supports combustion. Normally it is not hypergolic with fuels. Liquid oxygen will cause liquid fuels to cool and freeze if both liquids are brought together. Such a mixture of frozen fuel and liquified oxygen is shock sensitive and can react with the violence of a detonation. This hazard must be considered in fire control and preventive measures taken in connection with spills of liquid oxygen. Fire blankets must not be used to cover personnel whose clothing is impregnated with oxygen. Some of the materials that can react violently with oxygen are oil, grease, asphalt, kerosene, cloth, wood, paint, tar, and dirt. When working in an oxygen-rich environment, clothing may become saturated and may readily ignite.

B11.2.3 Explosion Hazards

When mixed with liquid oxygen, all materials that burn represent explosion hazards. These mixtures can usually be exploded by static electricity, mechanical shock, electrical spark, and other similar energy sources, particularly when the mixtures are frozen. All personnel involved in handling liquid oxygen should be made aware that the ordinary burning of rocket fuels or other combustile materials mixed with liquid oxygen, may progress to a detonation.

Leaking or spilled liquid oxygen can form potentially dangerous high concentrations of oxygen gas. During transfer operations, especially when liquid oxygen enters a warm system, large volumes of gas may be formed from "boil off." In confined areas, static electricity, an electrical spark, or flame will cause mixtures of gaseous oxygen and fuel vapors to explode.

When liquid oxygen is trapped in a closed system and refrigeration is not maintained, pressure rupture may occur. Oxygen cannot be kept liquid if its temperature rises above 155 K (-180.4°F) regardless of the confining pressure. Liquid oxygen trapped between valves can cause the pipe or tube to rupture violently. Loss of refrigeration may lead to storage tank rupture if the oxygen is not dumped or pressure-relieved by suitable devices. The loss of vacuum in vacuum jacketed tanks can cause increased evaporation and failure of the venting system if the system is inadequate for the extra flow.

B11.2.4 Environment Effects

None

B11.3 RESERVED

B11.4 RESERVED

B11.5 MATERIALS AND EQUIPMENT COMPATIBILITY

Liquid oxygen is classified as a liquified compressed gas for transfer and storage purposed, and must be handled and stored in fixed or mobile containers of approved design, materials, and construction.

B11.5.1 Materials

When selecting materials for liquid service, consideration should be given to physical properties at low temperature, and the reactivity of the material with liquid oxygen. The ability to withstand stress concentrations, particularly those resulting from sudden temperature changes, is important. Oxygen in air may create a dangerous fire hazard if it escapes or leaks into combustible materials. Compressed oxygen in the presence of oils and greases is almost certain to cause fire (Reference 6). Liquid oxygen in contact with oxidizing materials cause explosion (Reference 7).

B11.5.1.1 Metals - Metals to be used in liquid oxygen equipment should possess satisfactory physical properties at extremely low operating temperatures. The following metals are recommended for service with liquid:

a. Aluminum and aluminum alloy types

1000	3000	5083	5454	6063
2014	5050	5085	5456	7075
2024	5052	5154	6062	

b. Stainless steel types

304	316	304L
310	321	304ELC

c. 9 percent nickel steel alloy

d. Copper and copper alloys

copper	aluminum bronze
naval brass	cupro-nickel

e. Nickel and nickel alloys

nickel	Inconel-X
Rene 41	Hastelloy B
K-Monel	Inconel 718

f. Titan alloys should not be used.

B11.5.1.2 Non-Metals (Reference 8) - The number of acceptable non-metals is small due to the extremely low temperatures encountered. The following list contains the acceptable non-metals:

tetrafluoroethylene Polymer (TFE, Halon TFE, Teflon, AFLAS or equivalent)
Viton
unplasticized chlorotrifluoroethylene Polymer (Kel F, Halon CTF, or equivalent)

B11.5.1.3 Lubricants - Some petroleum-based lubricants are not recommended because of their reactivity with compressed and/or liquid oxygen. Special lubricants such as the fluorolubes or the perfluorocarbons are recommended.

B11.5.2 Equipment

B11.5.2.1 Containers - Liquid oxygen should be stored in stationary or mobile tanks of approved materials and construction. Storage and shipping drums for other propellants are not to be used in this service. To insure against defects in material or fabrication, the storage tanks should be tested as required by the provisions of applicable ASME or DOT specifications for unfired pressure vessels. Materials used for pressure vessels operating at temperatures less than 244 K (-20°F) should be impact-tested in accordance with Paragraph UG-84, Section VIII, of the ASME Boiler and Pressure Vessel Code (Reference 9). Containers for the shipment, storage, and transfer of liquid oxygen fabricated in accordance with any standard that meets pertinent structural requirements. Storage containers should be vacuum-jacketed; the vacuum space may contain reflective insulation or powders. The storage tank itself should be of welded construction and should be equipped with an appropriate pressure-relief system (Section B11.5.2.8). Bottom outlets on storage tanks are recommended, since they materially simplify the transfer system design and the selection of pumping equipment.

B11.5.2.2 Pumps and Hoses - Since the storage tanks may be designed with bottom outlets, flooded-suction centrifugal pumps may be used when gravity flow is not applicable. Only pumps and shaft seals designed for liquid oxygen service should be used. Details on these pumps and hoses may be secured from manufacturers of oxygen handling equipment. Hoses should be of proper design and engineered specifically for liquid-oxygen service.

B11.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Mine Safety and Health Administration (MSHA).

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the MSHA or as accepted by the current edition of the National Electrical Code (Reference 10).

B11.5.2.4 Pipes and Fittings - The pipes and fittings should be of approved material and construction, and should be hydrostatically tested at specified pressures. The use of welded and flanged connections whenever possible is recommended.

B11.5.2.5 Gaskets - Gaskets may be made of soft metals selected from those listed in Section B11.5.1.

B11.5.2.6 Pressure Gauges - Liquid oxygen equipment should be monitored with acceptable, LOX-clean types of pressure gauges as required. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales. Gauges should be protected with blowout relief backs or plugs.

B11.5.2.7 Valves - The use of extended stem gate, globe, or ball valves provided with venting devices is recommended.

B11.5.2.8 Venting Systems and Pressure Relief - The storage container itself should be equipped with a bursting disc and a pressure-relief valve inn parallel, both discharging to the outdoor atmosphere through an adequately sized vent line. The insulated area, between the inner and outer shells, should be equipped with either a rupture disc or a pressure-relief device, so that pressure cannot build up and rupture the vessel. All lines and vessels in which liquid air may be trapped between closed valves should have pressure-relief valves; if it is likely that the relief valve may freeze, a rupture disc should also be provided.

B11.5.2.9 Grounding - Since oxygen supports combustion, all stationary or mobile tanks should be properly bonded and grounded.

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14. MILITARY STANDARD, COLOR CODE FOR PIPELINE OR COMPRESSED GAS CYLINDER, Department of Defense, MIL-STD-101.

15. MILITARY STANDARD, COLOR CODE FOR CONTAINERS OF LIQUID PROPELLANTS, Department of Defense, MIL-STD-172.

16. CODE OF FEDERAL REGULATIONS, Title 40, PROTECTION OF THE ENVIRONMENT, Parts 260 to 265, Vol. 45, 1980.

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-297.4°F	90.2 K
Freezing Point		-361°F	54.8 K
Density	(Gas)	0.083 lb/ft^3 at 70°F and 14.7 psia	1.3296×10^{-2} Mg/m^3 at 294.3 K and 101 kPa
	(Liquid)	9.53 lb/gal at -297.4°F	1.14 Mg/m^3 at 90.2 K
Specific Gravity - Vapor (relative to STP air)		1.105 at 32°F	1.105 at 273.15 K
Critical Density		26.8 lb/ft^3	0.430 Mg/m^3
Critical Pressure		737 psia	5.09 MPa
Critical Temperature		-181.1°F	154.8 K
Vapor Pressure		37 psia at -280°F	2.55 kPa at 99.8 K
		167 psia at -240°F	1.15 kPa at 122 K
		492 psia at -200°F	3.39 kPa at 144.3 K
		615 psia at -190°F	4.14 kPa at 149.8 K
Coefficient of Viscosity	Kinematic	0.17 centistokes at -297.4°F	1.7×10^{-7} m^2/s at 90.2 K
	Absolute	0.19 centipoises at -297.4°F	1.9×10^{-4} Pa s at 90.2 K

Figure B11-1 Physical Properties of Oxygen (MW = 32 g mol)

CHAPTER B12
FLUORINE AND FLUORINE MIXTURES

B12.1 PROPERTIES

Information in this chapter is applicable to fluorine and fluorine-oxygen (FLOX) mixtures unless otherwise specified.

B12.1.1 Identification

The propellant fluorine (F_2) is used in its elemental form as a cryogenic liquid. It is essentially pure containing only traces of oxygen, nitrogen, hydrogen fluoride, and carbon tetrafluoride. MIL-P-27405 (Reference 1) is the applicable specification covering this grade. The molecular weight of fluorine is 37.997 g mol.

The propellant FLOX is a mixture of liquified fluorine (F_2) and oxygen (O_2) containing only traces of argon, nitrogen, hydrogen fluoride, and carbon tetrafluoride. FLOX, a mixture ratio of 30 percent fluorine and 70 percent oxygen, is optimum mixture in reaction with RP-1 for the maximum specific impulse, I_{sp}. FLOX containing a greater fluorine content than 30 percent should be considered as fluorine for safe handling. Military Standard MIL-P-25508E and NASA SP-3037 (References 2 and 3) are the applicable oxygen specification to be blended with fluorine to produce FLOX.

B12.1.2 General Appearance

Fluorine is a yellowish gas over a wide range of temperatures and pressures. The liquid exhibits an amber color. The solid form is yellow when near the freezing point. At lower temperatures fluorine solid undergoes a change in crystalline structure and is white in color.

FLOX has a bluish gray color in either the vapor or liquid for. Both FLOX and fluorine have a characteristic pungent odor.

B12.1.3 Physical and Chemical Properties (Reference 4)

The physical properties of propellant fluorine is given in Figure B12-1.

B12.1.3.1 Solubility - Fluorine reacts with nearly all materials and is completely miscible with liquid oxygen and liquid oxygen and liquid nitrogen in all proportions.

B12.1.3.2 Stability - Fluorine and fluorine-oxygen mixtures are not affected by heat or shock. Liquid and gaseous fluorine and FLOX are stable indefinitely when stored in containers properly constructed of approved materials.

B12.1.3.3 Reactivity - Fluorine is the strongest oxidizing agent and one of the most reactive materials known. Under proper conditions, fluorine reacts with practically every known element and most compounds. Exceptions are some of the rare gases, and some completely fluorinated compounds. (See Table B12-1 for examples).

FLOX is a strong oxidizing agent which vigorously supports combustion. The degree of reactivity with other material is related to the percentage of fluorine in the FLOX mixture.

B12.1.3.4 Environmental Fate - Because of the highly reactive nature of fluorine and fluorine mixtures, fluorine is almost never found in nature in its elemental form. Inorganic fluorine compounds in the atmosphere are rapidly hydrolyzed in water vapor and converted to less volatile compounds. An air sampling program conducted by the NAPCA (National Air Pollution Control Association) in 1966-67, revealed that 97% of the samples in non-urban areas had fluoride concentrations below the detectable limits. The highest non-urban air concentration of fluoride was 0.16 mg/m^3 (Reference 5). Gaseous fluorine compounds in the atmosphere can either be washed out by precipitation or absorbed by plants.

Background concentrations of fluorides in soils are common from the weathering of natural deposits of fluorospar (calcium fluoride) and fluorapatite. High concentrations of fluoride (4 ppm) in drinking water have been associated with water pumped from wells ranging from 305 to 762 m (1,000 to 2,500 ft) in depth (Reference 6). Surface water concentrations of fluoride are usually low, less than 1 ppm.

B12.2 HAZARDS

B12.2.1 Health Hazards

B12.2.1.1 Toxicity - Fluorine and FLOX are severely toxic in both liquid and gaseous forms. The primary toxic effects are local discomfort and irritation to the eyes, lungs, and skin. Liquid fluorine in direct contact with the skin will cause severe burn. Industrial experience and animal studies indicate that acute exposures to fluorine or FLOX cause pathological lung changes prior to liver damage, kidney damage, or significant biochemical, hematological, weight, or skeletal changes (References 7 and 8). Concentrations in the order of 25 ppm (42 mg/m^3) are slightly irritating to the eyes after 5 minutes but could be inhaled without respiratory discomfort; 50 ppm (85 mg/m^3) is very pungent and moderately irritating; 60 to 70 ppm (102 to 199 mg/m^3) is very irritating and extremely uncomfortable in a few seconds.

Static testing of hycrogen-fluorine engines will produce hydrogen fluoride exhaust products which are toxic. Hydrogen fluoride is less toxic than fluorine and the threshold limit value-time weighted average 3 ppm (2.5 mg/m^3) is 3 times that of fluorine (Reference 9).

Hydrogen fluoride is also formed when fluorine reacts with water. The rate at which water and fluorine react increases with high concentrations of fluorine. Laboratory measurements indicate that there is almost no reaction under conditions simulating a fluorine spill into moist air. At fluorine pressures up to 40 mm of mercury (53,000 ppm or 90 g/m^3 in air at 1 atmosphere), negligible reaction occurred with water vapor, also present at pressures up to 40 mm of mercury. The gas mixture was heated to 308 K (95°F) in these laboratory tests, at which temperature a water vapor pressure of 4 mm of mercury corresponds to 95 percent relative humidity. Large scale spill tests 1361 kg (3000 lbs) of 70/30 FLOX indicate that hydrolysis does occur readily with atmospheric moisture. This is significant in evaluating downwind toxic concentrations from a spill since hydrogen fluoride (HF) gas is less toxic than fluorine (F_2) gas. The emergency exposure limits of hydrogen fluoride (HF) gas are as follows:

60 min	8 ppm (7 mg/m^3)
30 min	10 ppm (9 mg/m^3)
10 min	20 ppm (18 mg/m^3)

B12.2.1.2 Exposure Limits

B12.2.1.2.1 Threshold Limit Values-Time Weighted Average (References 9, 10, 11, 12 and 13).

ACGIH 1981 (Reference 9) recommends a TLV-TWA of 1 ppm (2 mg/m^3) for an eight-hour exposure. The TLV-STEL is 2 ppm (4 mg/m^3).

NIOSH (Reference 11) recommends a permissible exposure limit (PEL) of 0.1 ppm (0.2 mg/m^3) with an IDHL level of 25 ppm (42 mg/m^3). The IDLH value is "immediately dangerous to life or health," meaning that it is the maximum level from which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects.

B12.2.1.2.2 Emergency Exposure Limits - (Reference 14)

10 min	15 ppm (25 mg/m^3)
30 min	10 ppm (17 mg/m^3)
60 min	5 ppm (8 mg/m^3)

B12.2.1.2.3 Special Medical Information (Reference 11) - Application of iced 70 percent isopropyl alcohol to the involved skin may provide considerable relief from pain. Chemical injury to the respiratory tract by these compounds should be treated as with any primary lung irritant. Steroids should be considered. Skin contact with wetted hydrogen fluoride may require the injection of calcium ion to inhibit subcutaneous fluoride ion migration (Reference 15). Further information should be obtainable from a medical text on the treatment of hydrofluoric acid burns.

B12.2.2 Fire and Combustion Product Hazards

Fluorine and FLOX are highly reactive oxidizing agents. As such, they must be considered fire hazards. They will react with many materials not normally considered combustible such as sand and glass at elevated temperatures and asbestos at room temperature. High concentrations of gaseous fluorine, as well as the liquid itself, will spontaneously initiate combustion with any flammable material.

The reaction of fuels with liquid fluorine or FLOX is frequently rapid, and attempts to extinguish the resulting fire may not be successful. After the fluorine-fed fire has subsided or the fluorine has evaporated, efforts can be made to control or extinguish the secondary fires, provided there is no residual hydrogen fluoride or fluorine gas trapped within building structures. With the exception of fire fighters and damage control personnel, no one should be permitted in the area. Fire fighters entering the area must wear protective clothing and self-contained air breathing apparatus until there is no further indication of residual fluorine gas. After the fire, containers and associated systems must be purged with an inert gas and responsible personnel must insure that fluorine gas or liquid is not trapped in the system.

If a fluorine or FLOX spill occurs in an unconfined, open area, water fog or soda ash should be remotely applied to promote smooth and rapid combustion of the fluorine. Areas surrounding and downwind of the spill should be evacuated.

B12.2.3 Explosion Hazards

Liquid fluorine and rocket fuels ignite hypergolically upon contact. Uncontrolled mixing of liquid fluorine and liquid fuels will invariably result in an explosive reaction. The possibility exists that gaseous hydrogen and fluorine can form explosive mixtures. There have also been delayed explosive reactions with water and liquid fluorine (excluding water fog which burns smoothly).

Pressure rupture may occur when liquid fluorine, FLOX, or any cryogenic liquid is trapped in a closed system and refrigeration is not maintained. Liquid fluorine or FLOX trapped between closed valves, if allowed to vaporize, can cause a violent rupture of the pipe and subsequent fires.

Hydrocarbon fuels and the hydrazines have been found to be hypergolic with FLOX mixtures having greater than ten percent fluorine content. Fuels and 30/70 FLOX mixtures have very short ignition delays resulting in low order detonations and fires. Hydrogen, however, to be reliably hypergolic with FLOX, requires at least 35 percent fluorine in the FLOX mixture. A liquid hydrogen-30/70 FLOX mixture could have significant ignition delay and result in an explosion.

B12.2.4 Environmental Effects

Atmospheric releases of fluorine compounds pose an indirect threat to public welfare through their adverse effects on animals and vegetation. Major emitters of fluorine compounds include phosphate fertilizer, aluminum, steel, fluorinated plastics and hydrocarbon manufacturers (Reference 16). Relatively small quantities of fluorine gas and halogenated fluorine compounds are released through rocket engine test firings.

Gaseous fluorine compounds are responsible for damage to vegetation, since they are more easily absorbed through the leaves from the air. Soluble fluorine forms are the most easily absorbed by the plants and therefore cause the most damage. The sensitivity of plants to gaseous fluorine compounds varies. Sensitive varieties exhibit damage at concentrations as low as 0.5 to 1.2 ppm (0.8 to 2 mg/m^3) for several consecutive days. The most obvious damage is manifested in tip burn or necrosis of the plant tissue. Fluorine and halogenated fluorine compounds cause injury to plants by oxidation of tissues and deposition of fluoride ion after hydrolysis.

Adverse effects to animals is usually not associated with the ingestion of gaseous fluorine contaminants, rather with the ingestion of water and vegetation containing high fluoride concentrations. In man, the soluble forms of fluorides are absorbed in the gastrointestinal tract, 50% is excreted and the remainder is accumulated in the bones. High intake of fluorides in cattle is related to a loss of appetite and other symptoms of fluorosis. High concentrations of fluorides in drinking water are associated with mottled teeth (2-7 ppm). See Reference 6.

Activated alumina and bone char have been used successfully for de-fluoridation of drinking water supplies (Reference 17).

B12.3 RESERVED

B12.4 RESERVED

B12.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B12.5.1 Materials

The selection of the materials suitable for fluorine systems must be governed by two important factors:

a. the resistance of materials to fluorine attack

b. the material's mechanical strength characteristic at cryogenic temperatures.

It is important to note that elevated temperatures may cause fluorine cylinders to burst. Since fluorine is a dangerously reactive gas, precautions should be taken against contact with water which causes formation of toxic hydrogen fluoride gas. Other incompatibilities include: (1) vigorous reaction with most oxidizable materials at room temperature, frequently with ign' 'on, and (2) reaction with nitric acid to form fluorine nitrate, an explosive gas (Reference 18). From References 18 and 20, a list of compatible and a few examples of reactive materials is provided in Table B12-1.

B12.5.1.1 Metals - Metals considered satisfactory for handling gaseous fluorine at temperatures up to 344 K (160°F) are as follows:

nickel	copper
stainless steel	magnesium
mild steel	brass
Monel	aluminum

NOTE

These and other metals may also be suitable for use at higher temperatures and with specific operating conditions. Spray-coated or calcinated aluminum oxide is resistant to gaseous and liquid fluorine under flow conditions at both low and high temperatures 1973 K (3092°F).

For liquid fluorine, the following metals are recommended:

nickel	stainless steel types
aluminum	304L, 316 and 321
copper	brass
Monel	

B12.5.1.2 Non-Metals - Polytetrafluorethylenes (Teflon TFE, KEL F-81, Halon TFE, or equivalent) are acceptable for use at moderate pressures and low flow rates with gaseous fluorine. TFE tape has been used extensively as a thread sealant for gaseous service, if care is taken not to cover the first three internal threads.

There are no non-metals recommended for use with liquid fluorine under flow conditions.

Table B12-1 Typical Compatible and Reactive Chemicals with Fluorine.

MATERIAL	TYPE OF INCOMPATIBILITY
Compatible Chemicals:	
Oxygen, all rate gases with the exception of xenon	
Hydrogen fluoride	
Tetraflurometnane and other perfluorocarbon compounds	
Reactive Chemicals:	
Ammonia	Ignites on contact, explodes in admixture.
Phosphorous Pentachloride	Ignites on contact.
Trichyloride	Ignites on contact.
Chromyl chloride	Ignites at high concentrations.
Cyanoguanidine	By-products of reaction are extremely explosive in gas, liquid and solid states.
Carbon Tetrachlor- ide, Chloroform, etc.	Explosive reaction on direct, local contact with gaseous fluorine.
Bromine, Iodine, Dicyanogen	Ignites at ambient temperatures.
Chlorine	Needs sparking before ignition occurs, with explosion immediately following.
Liquid hydrocarbons	Violent explosions with direct contact.

B12.5.1.3 Lubricants - There are no lubricants suitable for fluorine service.

B12.5.2 Equipment

B12.5.2.1 Containers - Liquid fluorine may be stored in either fixed or mobile tanks of approved design and materials. Storage and shipping containers designed for noncryogenic fluids should not be used in this service. To insure against material or fabrication defects, storage tanks should not be used in this service. To insure against material or fabrication defects, storage tanks should be tested in accordance with the provisions of applicable ASME, ASTM, or DOT specifications for pressure vessels. Materials used for pressure vessels containing liquid fluorine should be impact-tested in accordance with the ASME Code (d), Paragraph UG-84, Section VIII (Reference 19).

Containers for shipping, storing, and transferring liquid fluorine should be fabricated in accordance with the physical and structural requirements dictated by the service for which it is intended. The tanks for storing liquid fluorine are usually constructed with three horizontal, concentric shells forming an outer shell, an intermediate shell, and an inner shell. As dictated by the user, the two inner shells should be constructed of either Monel or stainless steel. The inner shell contains liquid fluorine and the intermediate shell contains liquid nitrogen. The outer shell forms a vacuum jacket and may contain an insulating material of two thermal conductivity such as Santocel or Perlite.

B12.5.2.2 Pumps and Hoses - When pumps are used in a fluorine/FLOX transfer system, only pumps and shaft seals designed and suitable for fluorine/FLOX service should be used. Detailed information for pumps and seals may be procured from manufacturers of fluorine/FLOX-handling equipment.

Hoses should be of proper design and engineered specifically for liquid fluorine service.

B12.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 24).

B12.5.2.4 Pipes and Fittings - Pipes and fitting used with liquid fluorine should be fabricated form nickel, Monel, stainless steel (304, 321, 316), copper, or aluminum. All pipes and fittings should be cryocycled, hydrostatically tested and leak tested at the prescribed pressures. Welded and welded-flanged connections are recommended. All welds should be made using inert gas shielding; welds should be of the butt type and should have full penetration verified by X-ray as no occlusions can be tolerated. Threaded connections, particularly where directly exposed to liquid fluorine, should be avoided. Usually, liquid fluorine lines should be insulated to prevent condensation of air or excessive heat loss. Metal tubing with compression AN-type fittings of suitable materials are also recommended. These AN-type fittings are suitable up to one inch size. Fittings should be used with soft copper or aluminum fitting rings and conical shaped seals to prevent leakage. As no lubricants should be allowed, silver plating of threads eliminates galling.

B12.5.2.5 <u>Gaskets</u> - Gaskets made from any of the following materials are recommended:

Gaseous service: aluminum, copper, lead, polytetrafluorethylene (TFE)

Liquid service: aluminum, copper

Liquid-fluorine equipment should be monitored through the use of approved pressure gauges as required. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales. Pressure gauges should be fluorine clean and the bourdon tube should be made of Monel or stainless steel of all welded construction. Pressure measuring devices should be used in gas-phase service as much as possible. Special care in selection, cleaning, and passivation should be exercised in all devices.

B12.5.2.6 <u>Valves</u> - Materials recommended for valve fabrication are forged stainless steel, Monel, or nickel. Packless-type bellow valves have proved to be more reliable in both liquid and gaseous fluorine service and should be used in fluorine installations.

In general, valves should be of an approved design for fluorine service; they should be installed so they can be operated remotely, with a barrier between the operator and the valve.

B12.5.2.7 <u>Venting Systems and Pressure Relief</u> - Large storage containers should be insulated, vacuum-jacketed, nitrogen-cooled tanks. The vacuum-insulated space between the intermediate and outer shells should be equipped with either a rupture disc or a pressure-relief device. The storage container itself should be of welded construction. The liquid nitrogen system should be equipped with an adequate vent line and a pressure-relief valve, but no pressure-relief devices are to be installed in the inner fluorine tank itself.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-306.64°F	85.02 K
Freezing Point		-362.32°F	54.08 K
Density	(Gas)		
	(Liquid)	12.5 lb/gal at -306.32°F	1.50 Mg/m^3 at 85.02 K
Specific Gravity (relative to STP air)		1.3	1.3
Critical Density		5.1 lb/gal	0.61 Mg/m^3
Critical Pressure		809.7 psia	5.583 MPa
Critical Temperature		-199.57°F	144.5 K
Vapor Pressure		19.7 psia at -300°F	135.8 MPa at 88.7 K
		164.7 psia at -250°F	1.136 MPa at 116.5 K
		794.7 psia at -200°F	5.479 MPa at 144.3 K
Coefficient of Viscosity	Kinematic	0.127 centistokes at -306.5°F	1.27×10^{-7} m^2/s at 85 K
	Absolute	0.257 centipoises at -306.3°F	2.57×10^{-4} Pas at 85.2 K
		0.414 centipoises at -333.4°F	4.14×10^{-4} Pas at 70 K

Figure 12-1 Physical Properties of Fluorine (MW = 37.997 g mol)

CHAPTER B13
THE HALOGEN FLUORIDES

B13.1 PROPERTIES

Information in this chapter is applicable to Chlorine Trifluoride (CTF) and Chlorine Pentafluoride (CPF). Their molecular weights are 92.45 and 130.45 g mols, respectively.

B13.1.1 Identification

Chlorine Pentafluoride (CPF) is covered by Military Specification MIL-P-27413 (Reference 1) and its chemical formula is ClF_5 Chlorine Trifluoride (CTF) is covered by Military Specification MIL-P-81399A (Reference 2) and its chemical formula is ClF_3.

B13.1.2 General Appearance and Common Uses

Chlorine Trifluoride is a corrosive, colorless gas at room temperature and pressure. It has a somewhat sweet odor and is highly irritating even at low concentrations. Chlorine Trifluoride is the most reactive compound of the halogen fluorides. It is a powerful oxidizing agent and is used as an igniter and propellant in liquid propellant engines. It is also used as an incendiary and fluorinating agent (References 3, 4). Chlorine Pentafluoride is a colorless vapor at temperatures above 291.3 K (64.6°F). The liquid is clear with a slight greenish cast. It is a toxic and corrosive oxidizing agent.

B13.1.3 Physical and Chemical Properties (Reference 5)

B13.1.3.1 Solubility - Chlorine Trifluoride and Chlorine Pentafluoride react with solvents rather than dissolve in them. Under normal conditions they react violently with ice or water.

B13.1.3.2 Stability - Chlorine Trifluoride decomposes above 493 K (428°F) and may pressure rupture its container at that temperature. Chlorine Pentafluoride is insensitive to shock, heat and electrical sparks.

B13.1.3.3 Reactivity - Chlorine Trifluoride and Chlorine Pentafluoride react with almost all elements except the inert gases, nitrogen and a few metals. They are not combustible by themselves but are very reactive and may cause fire on contact with organic matter. They also react violently with water, sand, silicon containing compounds, glass and asbestos. They are corrosive oxidizing agents and similar to elemental fluorine in reactivity.

B13.1.3.4 Environmental Fate - Both Chlorine Trifluoride and Pentafluoride exists as colorless vapors that are extremely toxic and corrosive. Both compounds are strong oxidizers and hypergolic with water and most materials. In water, hypochlorous, hydrochloric, and hydrofluoric acids are formed. In the atmosphere, the halogen fluorides would be absorbed into atmospheric moisture and form acids. These would be washed out in precipitation events. The acids in the moisture would react with a number of compounds on land forming chloride and fluoride salts. These salts would be very soluble and mobile in soil-water systems.

B13.2 HAZARDS

B13.2.1 Health Hazards

B13.2.1.1 Toxicity - Chlorine Trifluoride (CTF) and Chlorine Pentafluoride (CPF) are highly reactive oxidizing compounds and the gases can cause irritation of the skin, eyes and respiratory tract. Direct contact with either of these liquids will cause severe chemical burns. Since they are low boiling liquids, a dangerous atmospheric concentration may result from improperly sealed containers or loss of refrigeration. The odor of the gases in air may not be objectionable to some persons at concentration levels which are dangerous. The physiological effects of these compounds closely resemble those of fluorine and hydrofluoric acid. Almost all of the toxicity data comes from investigations with experimental animals, thus the recommended threshold limit values and emergency exposure limits are tentative and the latest revision of the values should be obtained by users of these propellants.

B13.2.1.2 Exposure Limits

B13.2.1.2.1 Threshold Limit Values-Time Weighted Average (References 6, 7 and 8)

Chlorine Trifluoride:	TLV-TWA = 0.1 ppm (0.4 mg/m^3) ceiling value
Chlorine Pentafluoride:	No data

B13.2.1.2.2 Emergency Exposure Limits (Reference 9) - The emergency exposure limit (EEL) defines the "single brief accidental exposures to airborne contaminants that can be tolerated without permanent toxic effects. These limits are not intended to replace accepted safe practices and should be accompanied by appropriate medical surveillance" (Reference 9).

a. Chlorine Trifluoride

10 min 7 ppm (27 mg/m^3)
30 min 3 ppm (11 mg/m^3)
60 min 1 ppm (4 mg/m^3)

Recent experimental work with fluorine inhalation by Keplinger has resulted in an increased emergency exposure limit value for fluorine. See Reference 4 and Fluorine and Fluorine-Oxygen Mixtures, Chapter B12 of this volume. Data on human exposures with CTF and CPF are not available, but it is believed that the CTF value given above is conservative.

b. Chlorine Pentafluoride (Reference 10)

10 min 3.0 ppm (13 mg/m^3)*
30 min 1.5 ppm (7 mg/m^3)*
60 min 0.5 ppm (2 mg/m^3)*

*These limits are tentative.

B13.2.1.3 Special Medical Information - Exposure to CTF and CPF vapor can cause severe irritation of the eyes, mucous membranes and the entire respiratory tract. Exposure to high concentrations may result in acute respiratory distress, pulmonary edema, and death. For most individuals the odor of CTF and CPF is objectionable well before the emergency exposure limit is reached.

Oxygen may be administered under positive pressure to counteract chemical pneumonitis and pulmonary edema. The application of iced magnesium sulfate (Epsom salt) solution or iced 70% isopropyl alcohol to involved skin may provide considerable relief from pain.

A momentary contact with the skin may produce a thermal burn, whereas if a quantity of any of these halogen fluorides are trapped in contact with the skin, a deep, painful chemical burn may result (References 11 to 16). For first aid information, see Appendix E13.

The imminent danger of exposure to halogen fluoride compounds is that they react violently in contact with water to form chlorine and hydrogen fluoride gases. The hydrogen fluoride vapor is very soluble in water, whereas chlorine (and possibly chlorine dioxide) is soluble to an appreciable extent about 0.7 to 0.8 g per 100 g of water. Extremely severe injuries caused by the halogen fluorides can be attributed more to the hydrolysis of the hydrogen fluoride gas than the less soluble chlorine product. Inhalation of halogen fluoride gas in producing pulmonary edema was probably caused by this mechanism of reaction. Severe inflammation of all mucosal surfaces resulting in lacrimation, corneal ulceration, and burning of exposed area of skin may also be attributed to the hydrofluoric acid reactant (Reference 16). Externally the skin whitens; the tissues beneath the skin are destroyed, the destruction spreading even to the bones. Skin which has been exposed to the hydrofluoric acid attack should be thoroughly washed with cold water until the whitening of the tissue disappears. An ointment consisting of 0.085 kg (3 oz) of magnesium oxide, 0.113 kg (4 oz) of heavy mineral oil, and 0.312 kg (11 oz) of white Vaseline should then be applied (Reference 17). Medical assistance should be obtained as soon as after an exposure to halogen fluoride vapor as possible.

B13.2.2 Fire and Combustion Product Hazards

CTF and CPF are hypergolic with most materials. Both react violently with small quantities or pools of water, but with a water spray or copious amounts of water, CTF and CPF react smoothly forming hydrogen fluoride and chlorine as major products of the reaction. Both compounds react strongly with silicon-containing compounds and support combustion of sand, asbestos, ground glass, etc.

The main product of reaction with water, water vapor, and combustibles is hydrogen fluoride which is also toxic and very corrosive (refer to Chapter 12 of this volume).

B13.2.3 Explosion Hazards

CTF and CPF are shipped as liquids in cylinders. They are hypergolic and react violently with small quantities of water or ice and vigorously with most combustible surfaces at room temperature, frequently igniting immediately. They react with most metals and metal oxides at elevated temperatures. CTR and CPF react on contact with water vapor, ammonia and hydrogen and with most fuel and organic vapors.

B13.2.4 Environmental Effects

Gaseous halogen fluorides released into the atmosphere would be a short-lived, nonpersistent contamination. The persistence of the hydrolysis products in the atmosphere before washout, however, is a function of the climate. It is likely that the compounds would be washed out before they diffuse into the stratosphere. Return to earth of the moisture with the acids, would result in the formation of the fluoride and chloride salts. Hydrogen chloride and fluoride would be expected to react with the major cations in the water and soil and form calcium, sodium and magnesium salts. These salts would be carried down into groundwater and through runoff into surface water. Chloride and fluoride salts are found abundantly in natural surface and ground waters.

B13.3 RESERVED

B13.4 RESERVED

B13.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B13.5.1 Materials

Nearly all materials not passivated will ignite when heated in the presence of CFT and CPF. The amount of heating required for ignition varies with the materials. "Hot Spots" developed from a trace of combustible materials or by a compression wave or friction from such as in turning a valve may be a source of ignition (References 21, 22).

B13.5.1.1 Metals - Passivated metals considered satisfactory for handling liquid and gaseous CTF and CPF at temperatures below 338.7 K (150°F) are as follows:

nickel	Aluminum 2024	Copper
Monel	Aluminum 5052	Brass
Aluminum 1160	Aluminum 6063	Silver Solder
Aluminum 3003	All stainless steels	Pure Aluminum
Aluminum 6061	Inconel	Magnesium
Aluminum 6066		

NOTE

These and other metals may also be suitable for use at higher temperatures and or specific operating conditions. Spray-coated or calcinated aluminum oxide is resistant to gaseous and liquid halogen fluorides under flow conditions at both low and high temperatures; 1973 K (3092°F) is the high temperature limit.

The following metals are not satisfactory for handling liquid or gaseous CTF and CPF.

Tantalum	Niobium	Titanium
Molybdenum	Tungsten	

B13.5.1.2 Non-Metals - Polytetrafluoroethylenes (Teflon TFE, KEL F-81, Hal TFE, or equivalent) are acceptable for use at moderate pressures and low flow rates with gaseous halogen fluorides. TFE tape has been used extensively as a thread sealant for gaseous service, if care is taken not to cover the first three internal threads.

There are no plastics acceptable for use with liquid halogen fluorides under flow conditions (See Section B13.5.2.5).

In general, Chlorine Trifluoride will attack all forms of plastics, rubber, and resins, except the highly fluorinated polymers "Teflon" and "KEL-F."

Refer to Section B13.5.2.5 for the metal laminated and non-metal gaskets which can be used.

B13.5.1.3 Lubricants - The use of standard petroleum-base lubricants is prohibited. Fluorinated hydrocarbons may form detonable mixtures. Fluorocarbons may be used sparingly to lubricate seals if the amount of lubricated surface is limited and in the gas phase exposure. Fluorocarbon lubricants should not be used under conditions of dynamic flow (Reference 23).

B13.5.2 Equipment

All equipment shall be cleaned and passivated.

B13.5.2.1 Containers

The halogen fluorides may be stored in either fixed or mobile tanks of approved design and materials. Storage and shipping containers furnished by the producer are not generally used as a part of a test system. An ASME code pressure vessel should be employed and tested for defects in accordance with the applicable specifications. If transfer by condensation is employed, the materials for constructing CTF and CPF vessels should be impact tested in accordance with the ASME code, Section VIII, Paragraph UG-84 (Reference 24). The physical and structural requirements will be dictated by the service for which the container is intended.

B13.5.2.2 Pumps and Hose - Standard pumps possessing a stuffing box are not suitable for CTF and CPF, but diaphragm pumps such as the Lapp Pulsafeeder (R) and the Chempump (R) may be used.

B13.5.2.3 Lights - Temporary, portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines. Flash lights or storage battery lamps, where permitted for use, shall be of the safety types approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (see Reference 25).

B13.5.2.4 Pipes and Fittings - Pipes and fittings for liquid halogen fluorides shall be fabricated from nickel, Monel, stainless steel (304, 321, 316), copper, or aluminum. All pipes and fittings should be properly tested for at least 2.07 MPa (300 psi) working pressure for CTF, and 3.45 MPa (500 psi) for CPF. Piping systems and cylinders should be clearly identified. Welded and welded-flanged connections are recommended. All welds shall be made using inert gas shielding; welds shall be of the butt type and shall have full penetration. Threaded connections, particularly where directly exposed to halogen fluorides, should be avoided. Metal tubing with compression AN-type fittings of suitable materials are also recommended. These AN-type fittings are suitable up to one-inch size. Fittings should be used with soft copper or aluminum fitting rings and conical shaped seals to prevent leakage.

B13.5.2.5 Gaskets - Gaskets for CTF and CPF service may be made from soft copper, 1100 aluminum, or Teflon laminated with copper or another metal, provided that no point on the exposed Teflon surface is more than 0.05 to 0.08 mm (0.002 to 0.003 in) removed from a metal heat conductor. Highly fluorinated polymers such as Teflon, Kel-F and Halon, which are resistant only to the vapor at ordinary temperatures, can be used as gasket materials if properly cleaned and vacuum dried. Halogenated solvents are absorbed strongly by the fluorinated polymers; therefore, solvent degreasing should be avoided.

B13.5.2.6 Pressure Gauges - Halogen fluoride equipment should be monitored using compatible pressure gauges as required. Pressure measuring devices should be used in gas-phase service only. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B13.5.2.7 Valves - Packless type valves have proven to be more reliable and are recommended for this service. Recommended materials for valve fabrication are forged stainless steel, Monel or nickel. Valves should be of an approved design and installed so they can be operated remotely, with a barrier between the operator and the valve.

B13.5.2.8 Venting System and Pressure Relief - Vents and pressure relief systems, as required, should be extended to the atmosphere. Their termination should be at a height and location that will give adequate personnel protection.

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CHAPTER B14
NITROGEN OXIDES

B14.1 PROPERTIES

B14.1.1 Identification (References 1 and 2)

Nitrogen tetroxide is an equilibrium mixture of nitrogen tetroxide (N_2O_4) and nitrogen dioxide (NO_2) and is also known as dinitrogen tetroxide, nitrogen peroxide or liquid nitrogen dioxide. Propellant grades of mixed oxides of nitrogen are identified by percent weight N_2O_4 with no more than 0.17 percent water equivalent (See Table B14-1 for applicable grades).

The MON (mixed oxides of nitrogen) propellants consist of dinitrogen tetroxide with 1 to 25 percent pungent nitric oxide (NO). The composition of the various grades are listed in Table B14-1.

B14.1.2 General Appearance and Common Uses

N_2O_4 is a volatile reddish-brown liquid. It is a powerful oxidizing agent containing about 70% available oxygen. The gas is yellowish to reddish brown and has a characteristic pungent odor. The characteristic color is due to the NO_2 resulting from the N_2O_4-NO_2 equilibrium mixture. Propellant mixed oxides of nitrogen, containing 99% N_2O_4 and 10% NO, resembles nitrogen tetroxide in color and odor. Nitric oxide is a volatile blue liquid. The gas is colorless.

B14.1.3 Physical and Chemical Properties (References 3 and 4)

Refer to Figure B14-1 for the physical properties of N_2O_4.

Nitrogen tetroxide when mixed with nitric oxide forms MON-type oxidizers. MON-type oxidizers have lower freezing points than N_2O_4. MON's density is lower than the density of N_2O_4 and MON's higher vapor pressure requires the use of larger and heavier pumps.

Dry N_2O_4 (less than 0.1 percent water equivalent) may be stored in low-pressure carbon-steel containers, since the vapor pressure at 333 K(140°K) is only 510 kPa (74 psia) and the corrosiveness at this water content is negligible for an indefinite period.

MON-type oxidizers, MON-1 and MON-3, are currently being specified for rocket oxidizers because the addition of NO acting as an inhibiter against corrosion of propellant tanks is anticipated. MON-3 is specified for Space Shuttle; to assure that the NO content of the oxidizer will remain above 0.5 percent NO after handling and storage of the oxidizer and ventings of the oxidizer tanks. The slight addition of NO does not change the thermophysical properties of N_2O_4.

The mixed oxides of nitrogen, MON-10 and MON-25, have higher vapor pressures than that of nitrogen tetroxide at 338.71 K (150°F), and require containers rated for these higher pressures.

B14.1.3.1 Solubility - N_2O_4 in water reacts to form nitric and nitrous acids. The nitrous acid decomposes forming additional nitric acid and evolving nitric oxide (NO). On contact with oxygen the NO converts the NO_2.

B14.1.3.2 Stability - N_2O_4 is very stable at room temperature. At 423 K (302°F) it begins to dissociate into nitric oxide and free oxygen, but upon cooling it reforms into N_2O_4.

B14.1.3.3 Reactivity - Oxides of nitrogen are corrosive oxidizing agents. In the presence of water the corrosiveness of these oxidizing agents are enhanced. N_2O_4 may react the combustible materials and is hypergolic with UDMH. NO is stable at room temperature. As NO concentrations are increased, the N_2O_4 mixture shows decreased hypergolicity. Oxides of nitrogen are not sensitive to mechanical shock, heat or detonation. They are nonflammable with air, but can support combustion.

B14.1.3.4 Environmental Fate - Nitrogen tetroxide is a highly toxic gas with corrosive fumes. In water, nitrogen tetroxide reacts to produce nitric and nitrous acids.

$$N_2O_4 + H_2O \quad HNO_3 + HNO_2$$

In the atmosphere, nitrogen dioxide is very rapidly photochemically decomposed to nitrogen oxide and atomic oxygen. The typical half life for nitrogen dioxide in sunlight is about 2 minutes. Atomic oxygen reacts with water for form hydroxyl radicals which are significant in atmospheric reactions and the formation of photochemical smog. Nitrogen oxides react with atmospheric moisture to form nitric acid rain which can be returned to the land in precipitation events.

B14.2 HAZARDS

B14.2.1 Health Hazards

B14.2.1.1 Toxicity - Because N_2O_4 and mixed oxides of nitrogen in liquid form are corrosive, severe burns of the skin and eyes can result unless the material is immediately removed. The liquid volatilizes readily, giving off yellowish to reddish-brown vapors containing a mixture of N_2O_4 and NO_2. The inhalation of toxic vapors is normally the most serious hazard in handling N_2O_4 and mixed oxides of nitrogen. The color of the vapors is not a reliable index of a degree of toxic hazard. The initial symptoms of poisoning - irritation of the eyes and throat, cough, tightness of the chest, and nausea - are slight and may not be noticed. Then, hours afterward, severe symptoms begin; their onset may be sudden and precipitated by exertion. Coughing, feeling of constriction in the chest, and difficult breathing are typical. Cyanosis (a blue tinge to the mucous membranes of the mouth, eyelids, lips, and fingernail beds) may follow. Persons with such symptoms are in great danger. Milder cases may show signs of bronchitis with cyanosis, and other may vomits and suffer nausea and abdominal pain.

The health hazards associated with mixed oxides of nitrogen are the same as those for nitrogen tetroxide.

Table B14-1 Identification of N_2O_4 and Mixed Oxides of Nitrogen (References 1 and 2)

COMPOSITION	NTO (Red-Brown)	MON-1 (Green)	MON-3 (Green)	MON-10	MON-25
Nitrogen tetroxide assay (N_2O_4 percent by weight)	99.5			88.8 min	73.8 min
Nitric oxide (NO) content-max percent by weight-minimum	(1)	1.0 0.6	3.0 1.5	11.0 10.0	26.0 25.0
N_2O_4 + NO - percent by weight minimum	99.5	99.5			
Water equivalent - percent by weight max	0.17	0.17	0.17	0.17	0.17
Chloride content - percent by weight max	0.040	0.040	0.040	0.040(2)	0.040(2)
Iron (Fe)					
Particulate g/m^3	10	10	10	10	10

(1) The NO content should be limited to that which does not change the specified Red-Brown color of the propellant.
(2) This test need not be performed on propellant manufactured by the ammonia-oxidation process.

B14.2.1.2 Exposure Limits

B14.2.1.2.1 Threshold Limit Values-Time Weighted Average (TLV-TWA) (References 5, 6 and 7)

Nitrogen tetroxide: 3 ppm (6 mg/m^3) (Given as the concentration of NO_2 gas).

TLV-STEL:	5 ppm (10 mg/m^3)
NIOSH:	1 ppm (1.8 mg/m^3)
IDLH:	50 ppm (90 mg/m^3)

Nitric Oxide: 25 ppm (30 mg/m^3) (Present in mixed oxides of nitrogen).

TLV-STEL:	35 ppm (45 mg/m^3)
IDLH:	100 ppm (129 mg/m^3)

B14.2.1.2.2 Emergency Exposure Limits (Reference 8) - The following recommended emergency exposure limits for NO_2 have been set:

10 minutes	30 ppm (60 mg/m^3)
30 minutes	20 ppm (40 mg/m^3)
60 minutes	10 ppm (20 mg/m^3)

The emergency exposure limit (EEL) defines the "single brief accidental exposures to air-borne contaminants that can be tolerated without permanent toxic effects. These limits are not intended to replace accepted safe practices and should be accompanied by appropriate medical surveillance" (Reference 9).

B14.2.2 Special Medical Information (Reference 9, 10, 11)

The development of pulmonary edema is the principal danger associated with the inhalation of N_2O_4 gas. A person may breath an atmosphere containing a dangerous concentration of N_2O_4 gas without serious discomfort at the time, only to suffer severe effects several hours later. A detailed therapeutic regimen is described in Reference 10.

Repeated exposure to low level concentrations of N_2O_4 gas may cause ulceration of the nose and mouth, wearing down and decay of teeth and chronic irritation of the entire respiratory tract. Bronchitis, bronchiectasis, and secondary pulmonary emphysema may develop.

Treatment for exposure to mixed oxides of nitrogen is the same as that for nitrogen tetroxide.

B14.2.3 Fire and Combustion Product Hazards

N_2O_4 and mixed oxides of nitrogen are normally stored and handled as liquids without refrigeration. Liquid N_2O_4 and mixed oxides of nitrogen will not burn; however, they will support combustion.

The oxygen content of N_2O_4 is about 70 percent by weight. When mixed with a fuel, it readily supports combustion. N_2O_4 is hypergolic with a number of fuels, including MMH, UDMH, hydrazine, aniline and furfuryl alcohol. With increasing concentrations of NO, mixed oxides of nitrogen are less hypergolic.

B14.2.4 Explosion Hazards

N_2O_4 and mixed oxides of nitrogen are oxidizers, but they are not hypergolic with all combustible materials. Such non-hypergolic mixtures, therefore, present an explosion hazard, particularly when subjected to elevated temperatures, pressures or impact. If containers leak, the oxides of nitrogen gases can form explosive mixtures with fuel vapors, especially in confined spaces. N_2O_4 of commercial purity and mixed oxides of nitrogen are stable at ordinary temperatures. There is a possibility that containers in proximity to a fire may pressure rupture and the released gas can form explosive mixtures.

Organic materials and partially halogenated solvents should not be used as flushing or decontamination fluids unless specifically tested previously under the conditions of usage. Mixtures of N_2O_4 and mixed oxides of nitrogen and the following partially halogenated solvents can be initiated by heat and shock, yielding violent explosions.

Methylene chloride
1,1,1 - Trichloroethane (Methylchloroform)
Trichloroethylene
Perchloroethylene
Chloroform
Carbon Tetrachloride
Dichloroethylene
1,2 - Dichloroethane
Asymmetrical Tetrachloroethane

Mixtures of N_2O_4 and ethylene glycol when confined will react explosively.

B14.2.5 Environmental Effects

Because of human health and other risks associated with nitrogen oxides, the Federal Ambient Air Quality Standard on nitrogen dioxide is 0.05 ppm. Other environmental effects are associated with the generation of photochemical smog and acid rain. Nitric acid in surface and soil water systems would react with metals such as calcium and magnesium. The nitrate is a very common constituent of surface water systems. Nitrate in drinking water supplies is limited because of health effects associated with consumption of high concentration of nitrate, particularly for infants.

B14.3 RESERVED

B14.4 RESERVED

B14.5 <u>MATERIALS AND EQUIPMENT COMPATIBILITY</u>

B14.5.1 <u>Materials</u>

B14.5.1.1 <u>Metals</u> - The selection of metals for oxide of nitrogen service should be governed by the oxidizer's moisture content. The following metals are suitable:

a. When moisture is 0.1 percent or less

Carbon steels
Aluminum
Stainless steels
Nickel
Inconel

b. When moisture content is above 0.1 percent

Stainless steel (300 series)

B14.5.1.2 <u>Non-Metals</u> - The following non-metals may be used:

Ceramic (acid-resistant)
Pyrex glass
Polytetrafluoroethylene (TFE or equivalent)
Polyethylene (limited use)

B14.5.1.3 <u>Lubricants</u> - Hydrocarbon lubricants react with oxidizers, and therefore must be avoided. The following lubricants are resistant to strong oxidizers and may be used:

Fluorocarbon oils, greases and waxes
Nordco seal-147 and DC 234S or equivalent
Polytetrafluoroethylene tape

B14.5.2 <u>Equipment</u>

B14.5.2.1 <u>Containers</u> - Oxides of nitrogen are shipped in cylinders, tank trucks and tank cars. They may be stored in cylinders and tanks (main storage and mobile). See Section B14.5.1.1 for the approved metals for container construction.

The tanks should be of welded construction and should be constructed according to the ASME Boiler and Pressure Vessel Code (Reference 15). Since oxides of nitrogen do not present a corrosion hazard, the tank may be equipped with bottom outlets for transfer and cleaning. The tanks should be equipped with adequate pressure relief valves.

Storage tanks for service in mixed oxides of nitrogen require higher pressure ratings than if used only for NTO. Storage tanks must be equipped with both topfill lines and fume-return (or vent) and pressure-balance lines. This design is necessary to keep the system closed during transfer and filling operations. The tanks should also be equipped with liquid-level gauges and, if practical, a high- and low-level alarm system.

The design and construction of mobile tanks should, insofar as possible, comply with DOT regulations. This would avoid the use of low-pressure tanks which may be satisfactory for stationary storage but too light for safe handling in transit.

B14.5.2.2 Pumps and Hose - Transferring nitrogen tetroxide and mixed oxides of nitrogen by pump is preferred because it minimizes the possibility of introducing moisture into the system. The pumps shall be constructed of an approved material and may be of the centrifugal or rotary type. A seal-less pump of stainless steel construction is recommended; as the design includes no exposed shafts. The gear type of rotary pump is widely used for small volume propellant service. This type of pump can use mechanical seals in place of packing, and is quite reliable. The seal material should not contain carbon or graphite. Hoses should be fabricated of polytetrofluoroethylene with stainless steel braid and fittings. Flexible metal hoses of stainless steel construction are also satisfactory. The hose must be designed for the service intended.

B14.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 16).

B14.5.2.4 Pipes and Fittings - The pipes and fittings should be of approved construction materials (Section B14.5.1.1) and should be tested for the design working pressure. Whenever possible, piping and fittings shall be welded. Threaded connections should be sealed with thread sealing compound of water glass (disodium silicate) and graphite and polytetrafluoroethylene tape.

B14.5.2.5 Gaskets - Gaskets and "O" rings may be fabricated from any of the recommended non-metals listed in Section B14.5.1.2.

B14.5.2.6 Pressure Gauges - Standard types of pressure gauges made of compatible materials and cleaned for oxidizer service should be used with NTO and MON. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B14.5.2.7 Valves - Leakage cannot be tolerated in valves selected for mixed oxides of nitrogen service. Where full flow with minimum pressure drop is required, ball valves may be used; however, seal design is critical. Needle valves can be used for bleed, sampling, and purge systems. Polytetrofluoroethylene chevron packing is recommended for valve stem seals. Valve body construction should be of stainless steel. Valve trim should be polytetrafluoroethylene or polychlorotrifluoroethylene. Gate and butterfly valves should be avoided for nitrogen tetroxide service due to particle migration problems in a soft sealed valve of this type.

B14.5.2.8 Venting Systems and Pressure Relief - Venting and safety relief equipment should be constructed of compatible materials as given in Section B14.5.1, and provided as required.

B14.5.2.9 Grounding - All tanks should be properly grounded.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		70.1°F	294.2[illegible] K
Freezing Point		11.8°F	261.95 K
Density (Gas)		0.206 lb/ft³ at 70°F	[illegible] kg/m³ 294 K
(Liquid)		12.1 lb/gal at 68°F	1.45 Mg/m³ at 293 K
Specific Gravity (Vapor) (relative to STP air)		2.8	2.8
(Liquid) [Water = 1 at 293 K (68°F)]		1.45	1.45
Critical Density		4.65 lb/gal	0.550 Mg/m³
Critical Pressure		1464.5 psia	10.1 MPa
Critical Temperature		316.8°F	431.4 K
Vapor Pressure		5.1 psia at 32°F	35 kPa at 273.15 K
		11.31 psia at 50°F	78 kPa at 288.15 K
		56.26 psia at 112°F	338 kPa at 323.15 K
		290 psia at 212°F	2.0 MPa at 373.15 K
Coefficient of Viscosity	Kinematic	0.287 centistokes at 68°F	2.87×10^{-7} m²/s at 298 K
	Absolute	0.416 centipoises at 68°F	4.16×10^{-4} Pa·s at 293 K
Autoignition Temperature		Non flammable	
Flash Point		Non flammable	
Flammability Limits	Lower: Upper:	Non flammable in air	

Figure B14-1 Physical Properties of N_2O_4 (NTO) (MW = 92.014 g mol)

CHAPTER B15
FUMING NITRIC ACID

B15.1 INTRODUCTION

B15.1.1 Identification

The fuming nitric acids, FNA, as specified in Military Specification MIL-P-7254 series (Reference 1), are identified by chemical composition (percentage by weight) with the limits as listed in Table B15-1. Type IIIB is also known as inhibited red fuming nitric acid (IRFNA). Type IV is high density acid (HDA) or inhibited maximum density fuming nitric acid (IMDFNA).

B15.1.2 General Appearance and Common Uses

The nitric acids discussed in this chapter are fuming liquids which, depending upon the amount of dissolved nitrogen oxides, vary from colorless to reddish brown. Fuming nitric acid (FNA) is a light orange to orange-red (Reference 2), clear, strongly fuming, very corrosive liquid which evolves toxic nitric acid vapor and yellow-red vapors of nitrogen oxides. They are used as Rocket Propellant oxidizers.

B15.1.3 Physical and Chemical Properties

Fuming nitric acids (FNA) consist of concentrated nitric acid containing added nitrogen dioxide. The water content is low, and hydrogen fluoride is added as an inhibitor. Vapors from these acids have a characteristic pungent odor.

B15.1.3.1 Solubility - Nitric Acids are soluble in water in all proportions: there is an accompanying evolution of heat and oxides of nitrogen.

B15.1.3.2 Stability - Fuming nitric acids are unstable releasing nitrogen dioxide, nitric oxide and mists of nitric acid to the atmosphere.

B15.1.3.3 Reactivity - The fuming nitric acids are highly corrosive oxidizing agents and will vigorously attack most metals. They reach with may organic materials, spontaneously causing fire. In rare instances, on gross contact with certain materials (e.g., hydrazine) and when spontaneous ignition is delayed because of degraded materials, an explosion may occur. The nitric acids will react with sea water, releasing large quantities of toxic nitrogen oxides. THese chemicals are hygroscopic.

B15.1.3.4 Environmental Fate - Nitrogen tetroxide is a highly toxic gas with corrosive mist. Nitrous oxide is relatively unreactive and inert to the halogens, alkali metals and ozone at room temperature. In water, nitrogen tetroxide reacts to produce nitric and nitrous acids.

$$N_2O_4 + H_2O \quad HNO_3 + HNO_2$$

In the atmosphere, nitrogen dioxide is very rapidly photochemically decomposed to nitrogen oxide and atomic oxygen. The typical half life for nitrogen dioxide in sunlight is about 2 minutes. Atomic oxygen reacts with water to form hydroxyl radicals which are significant in atmospheric reactions and in the formation of photochemical smog. Nitrogen oxides are caught up in an oxidation cycle in the atmosphere. Within this cycle nitric acid is formed, which can be washed out and returned to the land in precipitation events.

B15.2 HAZARDS

B15.2.1 Health Hazards

B15.2.1.1 Toxicity - FNA, in contact with any surfaces of the body (skin, mucous membranes, eyes) destroys tissue by direct action. It stains the skin and tissue yellow or yellowish brown. Sustained contact, more than instantaneous, will result in a chemical burn. FNA vapors are highly irritating and toxic to the respiratory tract. Immediately after exposure to dangerous concentrations, there may be coughing, increased respiratory, rate, asthmatic-type breathing, nausea, vomiting, and marked fatigue. A fatal pulmonary edema may develop.

B15.2.1.2 Exposure Limits

B15.2.1.2.1 Threshold Limit Value-Time Weighted Average (TLV-TWA) (See References 4, 5 and 6) - FNA contaminates the surrounding atmosphere with nitric acid mists nitric oxide, the nitrogen dioxide for (its dimer nitrogen tetroxide). Consequently, a threshold limit value for FNA has not been established. However, the atmospheric threshold limit values for its more toxic components are as follows:

Nitric acid mist	2 ppm (5 mg/m^3)
Nitrogen dioxide	3 ppm (6 mg/m^3)
Nitric Oxide	25 ppm (30 mg/m^3)

Table B15-1 Fuming Nitric Acids - Chemical Composition
(Percentage by Weight)

TYPE	COMMON NAME	NITROGEN DIOXIDE (NO_2)	WATER (H_2O)	NITRIC ACIDS (HNO_3)	SOLIDS AS NITRATES	HYDROGEN FLUORIDE INHIBITOR
IIIB	Inhibited red fuming nitric acid (IRFNA)	13-15	1.5-2.5	81.6-84.8	0.04 max	0.7
IV	High Density Acid (HDA)	42-46	0.5 max	52.7-57.4	0.04 max	0.7

The following IDLH values are:

Nitric acid mist	100 ppm (250 mg/m^3)
Nitrogen dioxide	50 ppm (100 mg/m^3)
Nitric Oxide	100 ppm (120 mg/m^3)

The short term exposure limit (TLV-STEL) is as follows:

Nitric acid mist	4 ppm (10 mg/m^3)
Nitrogen dioxide	5 ppm (10 mg/m^3)
Nitric Oxide	35 ppm (45 mg/m^3)

B15.2.1.2.2 Emergency Exposure Limits (See Reference 7) - The following recommended emergency exposure limits for nitrogen dioxide have been set:

10 minutes	30 ppm	(54 mg/m^3)
30 minutes	20 ppm	(36 mg/m^3)
60 minutes	10 ppm	(18 mg/m^3)

B15.2.1.3 Special Medical Information (See References 7 and 8) - Exposure to dangerous atmospheric concentrations of the oxides of nitrogen may cause spasm of the terminal bronchioles and disturbance of reflexes affecting respiration. Circulatory collapse may ensue, or the symptoms may subside and reappear several hours later with the onset of pulmonary edema. Complete rest from the moment of exposure is essential even if the initial symptoms are not particularly alarming. The individual should be hospitalized promptly. Certain signs indicating that severe lung damage has occurred may appear within the first few hours. They are an increase in platelets in the venous blood, often as great as 60 to 100 percent, a decrease in blood pressure, and an increase in the hemoglobin content of the blood. Spasmodic cough and dyspnea appearing several hours after the exposure are evidence of the development of pulmonary edema. Bronchopneumonia may be a complication.

Oxygen therapy is indicated if there are signs of hypoxia or if respiratory distress is present. The use of respiratory depressant drugs should be avoided and the use of cardiac drugs should be limited to threatened heart failure. Antibiotics may be indicated in combating complications such as pneumonia.

If no symptoms develop during a 24-hour observation period, the individual may be allowed to resume his normal activity, but he should remain under periodic medical surveillance for at least two weeks as a precautionary measure against the development of complications.

Chronic exposure to low concentrations of the oxides of nitrogen may produce wearing down and decay of the teeth, pulmonary emphysema, and chronic inflammation of the respiratory passages, often with ulceration of the nose or mouth.

FNA in contact with the eye causes irreparable damage in a matter of seconds; therefore, emphasis should be on immediate eye irrigation and other adequate first aid or self aid measures. After that, the individual should be seen promptly by the physician and eye irrigation should be continued utilizing sodium bicarbonate as a chemical antagonist. Subsequent treatment should be determined by an ophthalmologist. It is difficult to tell soon after exposure how great the risk has been. Despite their protestations, persons so exposed should be kept under observation for at least 24 hours; those known to have been seriously exposed should be removed to a hospital. Absolute rest is essential. Patients should be kept warm but not overheated. The administration of oxygen is often desirable.

B15.2.2 Fire and Combustion Product Hazards

Nitric acids by themselves will not burn. The fumes liberated by the acids support combustion. The type of fire that may occur in the presence of nitric acid is governed by the combustible material or fuel involved. Acid spills may ignite materials such as wood or rope, and the fire will be typical of the materials burning. Aniline and other hypergolic fuels quickly ignite on contact with acid. Both hypergolic and nonhypergolic fuels, once ignited, undergo flare burning in contact with the acids. The intensity of this type of fire depends upon the rate at which fuel and acid mix and, if large quantities of fuel and acid are mixed rapidly, there may be a violent reaction.

B15.2.3 Explosion Hazards

Although nitric acid is stable to mechanical shock and impact, upon contact with certain fuels (such as aniline or furfuryl alcohol) it will react violently. Nitric acid will form explosive mixtures with nonhypergolic fuels (such as hydrocarbons) and with hypergolic fuels if either the fuels or the nitric acid are not diluted with an excess of water. In rare instances, as in gross contact with certain materials (e.g., H_2) and when spontaneous ignition is delayed because of degraded materials, an explosion may occur. High temperatures in confined spaces may cause containers or other equipment to rupture from pressure build-up.

B15.2.4 Environmental Effects

Because of human health and other risks associated with nitrogen oxides, the Federal Ambient Air Quality Standard on nitrogen dioxide is 0.05 ppm (0.1 mg/m^3). Other environmental effects are associated with the generation of photochemical smog and acid rain. Nitric acid in surface and soil water systems would react with metals to form calcium and magnesium nitrates. The nitrate, are very common constituents of surface water systems. Nitrates in drinking water supplies are limited because of health effects associated with consumption of high concentration of nitrate, particularly for infants.

B15.3 RESERVED

B15.4 RESERVED

B15.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B15.5.1 Materials

B15.5.1.1 Metals - The following types of metals are approved for use with fuming nitric acids:

Aluminum	Stainless Steel
1060	347
1100	19-9 DL
3003	19-9 DX
3004	304 ELC
6061	321
5052	303
5154	316
	Durimet 20

All other ferrous and nonferrous metals and their alloys are prohibited because they react with fuming nitric acid, producing toxic oxides of nitrogen as well as failures from corrosion. Titanium metal and alloys of which titanium is a major constituent must be particularly avoided in FNA and IFNA systems because of a possible fire hazard.

B15.5.1.2 Non-Metals - The following non-metals are approved for use:

a. Kel-F-81, Teflon TFE, Halon TFE or equivalent

b. Resin-X (concrete protective coating) Epoxy, type MIL-P-22808A

B15.5.1.3 Lubricants - Three types of lubricants approved for use with fuming nitric acids are as follows:

a. Nordcoseal-147-S

b. Fluorolube

c. Perfluorocarbons

B15.5.2 Equipment

B15.5.2.1 Containers - All storage tanks should be resistant to fuming nitric acids. See Section B15.5.1.1 for the approved metals for container construction. Tanks should be constructed according to the ASME Boiler and Pressure Vessel Code (See Reference 11). The tanks should be equipped with adequate pressure relief devices.

B15.5.2.2 Pumps and Hose - Acid hoses should be made of flexible braided stainless-steel wire with TFE or stainless-steel bellows inner liner.

B15.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 12).

B15.5.2.4 Pipes and Fittings - Pipes and fittings should be of approved construction materials (Section B15.5.1.1) and should be tested for design working pressure.

B15.5.2.5 Gaskets - Gaskets and "O" rings may be fabricated from any of the recommended non-metals in Section B15.5.1.2.

B15.5.2.6 Pressure Gauges - Acid-storage facilities should be monitored with pressure gauges of approved stainless-steel construction and cleaned for oxidizer service. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B15.5.2.7 Valves - Valve body construction should be of stainless-steel.

B15.5.2.8 Venting System and Pressure Relief - All acid-storage tanks should be provided with a venting system and with a vapor-pressure-relief valve of adequate size, set a safe working pressure which will be determined by the material and construction of the tank.

B15.5.2.9 Grounding - All tanks should be properly grounded (See Chapter 7, Section B7.6.2.6.

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CHAPTER B16
IODINE

B16.1 PROPERTIES (References 1, 2)

B16.1.1 Identification

The chemical formula for iodine is I_2 and its molecular weight is 253.809 g mol. Iodine, a halogen, occurs naturally in sea water and nitrate bearing earth. Ultrapure iodine can be obtained from the reaction of potassium iodine with copper sulfate.

B16.1.2 General Appearance and Common Uses

Iodine is a violet-black, metallic lustrous solid, subliming at ordinary temperatures into a blue-violet gas with an irritating odor.

Iodine is used in the synthesis of chemical intermediates for the pharmaceutical, photographic and chemical industries. It is also used during the manufacture of photoelectric cells, rectifiers and specialty lubricants. It forms compounds with many elements but is less active than other halogens which displace it from iodides.

B16.1.3 Physical and Chemical Properties

B16.1.3.1 Solubility - Iodine is only slightly soluble in water, but dissolves readily in chloroform, carbon tetrachloride, or carbondisulfide to form beautiful purple solutions.

B16.1.3.2 Stability - Iodine is stable.

B16.1.3.3 Reactivity - Iodine is not combustible by itself, but can react vigorously with reducing materials.

B16.1.3.4 Environmental Fate - Iodine is a solid that is slightly soluble in water and sublimes at atmospheric pressure to a violet vapor. As a vapor emission, iodine would absorb into atmospheric particulates and to a limited extent into atmospheric moisture. In both cases, the iodine will eventually be redeposited on land during precipitation. In surface water, iodine would dissociate over time into the iodate form. Iodate forms complexes with metal ions, particularly calcium and sodium.

B16.2 HAZARDS

B16.2.1 Health Hazards

B16.2.1.1 Toxicity - Iodine can enter the body by inhalation of the gas, ingestion, and contact with the skin or eyes. The organs affected included respiratory system, lungs, eyes, skin, central nervous system and the cardiovascular system. Symptoms include eye or nose irritation, lacrimation, headache, chest constriction, skin burns, cutaneous hypersensitivity, mouth burning, vomiting, abdominal pain, diarrhea and skin rash (Reference 1). It may cause severe breathing difficulty after some delay. The crystalline material or strong solutions may cause severe irritation and skin burns. It is difficult to remove from the skin. An allergic skin rash may develop. Ingestion of iodine or solutions may cause burning in the mouth, vomiting, abdominal pain and diarrhea, and can be fatal. Chronic exposure to iodine may cause insomnia, eye or nasal inflammation, bronchitis, tremor, rapid heart beat, diarrhea, and weight loss. Some persons may develop skin sensitization.

When heated, it emits highly toxic fumes of iodine and iodine compounds. However, serious exposures seldom occur in industry due to the solid materials low volatility at room temperatures (Reference 3).

B16.2.1.2 Exposure Limits - The threshold limit value (TLV) for iodine is 0.1 ppm (0.1 mg/m^3) (References 4, 5 anD 6). The IDLH value is 10 ppm (100 mg/m^3). The IDLH value is "immediately dangerous to life or health," meaning that it is the maximum level from which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects (Reference 7).

B16.2.1.3 Threshold Limit Value-Time Weight Average (TLV-TWA)

0.1 ppm (1 mg/m^3) (ceiling value)

B16.2.1.4 Special Medical Information - For first aid information, see Appendix E16, CPIA 394.

B16.2.2 Fire and Combustion Product Hazards

Iodine is non-combustible, but can react vigorously with reducing materials.

B16.2.3 Explosion Hazards

Contact with gaseous or aqueous ammonia forms a water-insoluble, very shock sensitive explosive solid. Contact with acetylene, acetaldehyde, powdered aluminum or other active metals may cause fires and explosions.

B16.2.4 Environmental Effects

Iodine is found in nature in brines as iodide and as sodium and calcium iodates. Elemental iodine is a strong oxidizing agent and has been considered as an alternative to chlorine as a disinfectant. In saturated solutions in water iodine dissociates to form hypoiodous acid and hypoiodate. Hypoiodous and hypoiodate acids are unstable, however, and rapidly react to form complexes with soluble metals.

B16.3 RESERVED

B16.4 RESERVED

B16.5 MATERIALS AND EQUIPMENT COMPATIBILITY (References 1, 2)

B16.5.1 Materials

Iodine is not combustible by itself, but can react vigorously with reducing agents.

Contact of iodine with gaseous or aqueous ammonia forms a water-insoluble solid that is very sensitive to shock when dry and will explode, causing fires. Contact with acetylene, acetaldehyde, powdered aluminum, or other active metals may cause fires and explosions (NIOSH).

B16.5.1.1 Containers - Iodine should be stored in fixed or mobile containers of approved design and construction, made with properly selected material and suitably sealed. Materials that are compatible for chlorine are recommended for use with iodine (See Chapter B17, Section B17.5.1.1).

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CHAPTER B17
CHLORINE

B17.1 PROPERTIES

B17.1.1 Identification

The chemical formula for chlorine is Cl_2 and its molecular weight is 70.906 g mol. It is available commercially in two grades of purity. Research grade has a minimum purity of 99.965 mole percent. High purity grade has a minimum 99.5 mole percent.

B17.1.2 General Appearance and Common Uses

Liquid chlorine is a clear amber-colored liquid. Gaseous chlorine is greenish-yellow. It has a disagreeable and suffocating odor with an irritating effect on the nose and throat. Chlorine has many uses. Some of the most important are the following:

a. for water purification

b. sanitation of swimming pools, industrial waste and sewage

c. manufacture of chemicals

d. bleaching of pulp and textiles

e. oxidizing agent

B17.1.3 Physical and Chemical Properties

B17.1.3.1 Solubility - The solubility of chlorine in water at 293K (68°F) is 0.7 g/100 g.

B17.1.3.2 Stability - Chlorine is not combustible but is a strong oxidizer.

B17.1.3.3 Reactivity - Chlorine combines directly with nearly all elements. It will attack some forms of plastics, rubbers and coatings, and it reacts with practically all metals. Contact with combustible substances (such as gasoline and petroleum products, turpentine, alcohols, acetylene, hydrogen, ammonia and sulfur) and finely divided metals may cause fires and explosions.

B17.1.3.4 Environmental Fate - A spill or leak of chlorine gas would rapidly volatilize into the atmosphere. Chlorine is highly soluble in water, therefore a gaseous emission of chlorine would be rapidly absorbed into atmospheric moisture, and returned to land during precipitation events. In water, chlorine dissociates to form hypochlorous acid, and hypochlorite, as follows:

$$Cl_2 + H_2O = HOCl + H^+ + Cl^-$$
$$HOCl = H^+ OCl^-$$

Chlorine is a strong oxidizer. In natural water systems the chlorine and its forms rapidly react with a variety of organic and inorganic compounds. A major reaction with chlorine is that with nitrogen forming chloramines.

B17.2 HAZARDS

B17.2.1 Health Hazards

B17.2.1.1 Toxicity - "Chlorine is extremely irritating to the mucous membranes of the eyes and respiratory tract. It combines with moisture to liberate nascent oxygen and form hydrochloric acid. Both these substances, if present in quantity, cause inflammation of the tissues with which they come in contact. If the lung tissues are attacked, pulmonary edema may result. A concentration of 3.5 ppm produces a detectable odor; 15 ppm causes immediate irritation of the throat. Concentrations of 50 ppm are dangerous for even short exposures. 1,000 ppm may be fatal, even where the exposure is brief" (Reference 5).

"Because of its intensely irritating properties, severe industrial exposure seldom occurs, as the workman is forced to leave before he can be seriously affected. In cases where this is impossible, the initial irritation of the eyes and mucous membranes of the nose and throat is followed by cough, a feeling of suffocation, and later, pain and a feeling of constriction in the chest. If exposure has been severe, pulmonary edema may follow, with rales being heard over the chest" (Reference 5).

Although chlorine can usually be detected below the TLV, olfactory fatigue develops, so that the odor may not provide adequate warning.

Liquid chlorine can cause eye and skin burns on contact.

When heated, chlorine will react with water or steam to produce toxic and corrosive HCl gas (Reference 4).

B17.2.1.2 Exposure Limits - Chlorine has a ceiling threshold limit value-time weighted average (TLV-TWA) or 3 mg/m^3 (1 ppm). Its threshold limit value-short term exposure limit (TVL-STEL) is given 9 mg/m^3 (3 ppm) (References 6, 7 and 8). NIOSH gives a value of 0.5 ppm (1.5 mg/m^3) for a 15 minute exposure. The IDLH value is given as 25 ppm (79 mg/m^3).

The IDLH value is "immediately dangerous to life or health", meaning that it is the maximum level from which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects.

B17.2.1.3 Special Medical Information - Chlorine can enter the body by inhalation or contact with the skin or eyes. It can cause damage to the lungs and respiratory system. Symptoms include burning of the eyes, nose, and mouth, lacrimation, rhonorrhea, coughing, laryngitis, shortness of breath, choking, wheezing, nausea, vomiting, substernal pain, headache, dizziness, syncope, pulmonary edema, pneumonia, hypoxemia, and dermatitis. If it penetrates into the lower airway, it may produce bronchitis. Eye exposure to high concentrations may cause ulceration of the conjunctive and cornea and destruction of all ocular tissues (References 3 and 9).

B17.2.2 Fire and Combustion Product Hazards

Chlorine is non-flammable, but it is a strong oxidizer and supports combustion even in the absence of oxygen. It can react upon contact with turpentine, ether, ammonia gas, illuminating gas, hydrocarbons, hydrogen, and powdered metals to cause fires or explosions. It does not have hazardous decomposition products (References 3, 5 and 9).

B17.2.3 Explosion Hazard

It reacts vigorously with active metals, and corrodes most others. It reacts with organic compounds, and forms explosive mixtures with flammable gases or vapors. Contact with combustible materials may create a fire or explosion hazard (References 3, 5 and 9).

B17.2.4 Environmental Effects (Reference 10)

Chlorine is not a natural constituent of fresh and marine water systems. Both free and combined chlorine residuals have biocidial properties. Chlorine is toxic to aquatic life. Some fresh water species are killed at concentrations as low as 0.006 mg/l of combined chlorine. Any accidental release of chlorine gas could have serious effects on a surface water (Reference 11).

B17.3 RESERVED

B17.4 RESERVED

B17.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B17.5.1 Materials

Chlorine should be stored in compressed gas cylinders in accordance with 29 CFR 1910.101. Contact with combustible substances (such as gasoline and petroleum products, turpentine, alcohols, acetylene, hydrogen, ammonia and sulfur) and finely divided metals may cause fires and explosions (Reference 13). It is not combustible, but is a strong oxidizer. Reactivity with metals increases with temperature. This may result in a cylinder bursting at elevated temperatures.

B17.5.1.1 Metals - At normal or moderately elevated temperatures, gaseous or liquid chlorine is noncorrosive to most metals. At high temperatures, however, corrosion presents a serious problem. Metals recommended for use with chlorine include:

Steel
Cast Iron
Wrought Iron
Copper alloys (most)
Nickel alloys (most)
Certain stainless steels
Lead

Wet chlorine does not attack:

High silica iron
Monel
Platinum
Silver
Tantalum

More detailed material information may be obtained from manufacturer specifications.

B17.5.1.2 Non-Metals - It is difficult to predict the compatibility of materials in all cases, since chlorine will attack some forms of plastics, rubber and coatings. At low pressures, wet chlorine may be handled in chemical stoneware.

B17.5.2 Equipment

B17.5.2.1 Containers - All tanks should be stored in containers resistant to chlorine. See Section B17.5.1.1 for the approved metals for container construction.

B17.5.2.2 Pipes and Fittings - Extra heavy black iron or steel pipe is recommended for dry liquid or gaseous chlorine.

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CHAPTER B18
NITROGEN TRIFLUORIDE

B18.1 PROPERTIES

The information in this chapter is applicable to gaseous and liquid nitrogen trifluoride unless otherwise specified.

B18.1.1 Identification (References 1 and 2)

The propellant nitrogen trifluoride (NF_3) may be used as an oxidizer in either gaseous or liquid form. Air Products and Chemicals Co. is the only known commercial producer of NF_3 and a typical composition of their product analysis is shown in Table B18-1. The military specification for nitrogen trifluoride was given by the Air Force in its procurement of NH_3 during the 1977-78 fiscal year (See Table B18-2). The molecular weight of nitrogen trifluoride is 71.00 g mol.

B18.1.2 General Appearance (Reference 2)

Pure nitrogen trifluoride is a colorless gas at room temperature and atmospheric pressure. It condenses to a colorless liquid at 144.12 K (-200.25°F). It is nearly odorless and its toxicity requires special precautions to limit the extent of inhalation.

B18.1.3 Chemical and Physical Properties (References 2 and 3)

The chemical and physical properties of nitrogen trifluoride are given in Figure B18-3.

B18.1.3.1 Solubility (Reference 2) - The solubility of gaseous nitrogen trifluoride is very low in liquids such as water and anhydrous hydrogen fluoride. The liquid is completely miscible in all proportions with the following liquids: Oxygen, ozone, fluorine, oxygen fluorides, nitrogen, argon and carbon tetrafluoride.

B18.1.3.2 Stability (Reference 4) - Nitrogen trifluoride is reasonably stable to hydrolysis by pure water and dilute aqueous acid and base. It has been quantitatively recovered from aqueous solutions after a week at 405K (270°F). Oxidation, reduction or fluorination processes are relatively difficult to achieve without a high energy activation sources such as a glow discharge system or high temperature reaction conditions.

When stored in a properly prepared container of approved materials, gaseous and liquid nitrogen trifluoride are stable indefinitely.

Table B18-1 Composition of "Typical" Air Products NF_3(1) (Reference 1)

COMPONENT	%
NF_3	99.34
[illegible]$_4$	0.27
$CO + O_2$	0.30
N_2O	0.038
N_2	0.042
CO_2	0.006
Active Fluorides as HF	0.0024

(1) Mean values based on analyses of nine cylinders

Table B18-2 Air Force Procurement Specification for NF_3 (Reference 2)

COMPONENT	% Weight
NF_3 (min.)	98
Total Impurities (max.)	2
$N_2 + O_2 + CO$ (max.)	1
CF_4 (max.)	1
N_2 (max.)	0.2
CO_2 (max.)	0.1
N_2F_2 (max.)	0.1
Total Active Fluorides as HF (max.)	0.1

B18.1.3.3 Reactivity - Nitrogen trifluoride at high temperatures is a powerful oxidizing agent but is less reactive than fluorine. It is thermodynamically stable and inert to most materials at low and moderate temperatures. It is similar to oxygen in reactivity. At temperatures which cause appreciable dissociation of its molecules, it takes on the reactive character more akin to that of fluorine. Nitrogen trifluoride reacts readily with alkali metals and barium, but even with these elements, gentle heating is required to achieve extensive rapid reactions (Reference 5). Nitrogen trifluoride is also capable of reacting with many inorganic compounds if the energy for activation is available. When any reaction with NF_3 occurs, high heats of reaction are common, many sufficiently energetic to cause ignition. Gaseous mixtures with carbon monoxide are stable at room temperature but are known to explode on warming. Diborane mixtures with NF_3 detonate on condensing the gaseous phase.

B18.1.3.4 Environmental Fate -- Gaseous nitrogen trifluoride being heavier than air will settle in the low-lying areas and can be washed down into the soil where it will eventually be reacted or absorbed by plants. Background concentrations of fluorides in soils are common from the weathering of natural deposits of fluorospar (calcium fluoride) and fluorapatite. High concentrations of fluoride (4 ppm) in drinking water have been associated with water pumped from wells ranging from 304.8 to 609.6 m (1,000 to 2,500 ft) in depth (Reference 6). Surface water concentrations of fluoride are usually low, less than 1 ppm. See Section B18.2.4.

B18.2 HAZARDS

B18.2.1 Health Hazards

Nitrogen trifluoride is nearly odorless but is toxic. Gaseous nitrogen trifluoride is innocuous in skin contact. Direct contact with the liquid, however, would cause a cryogenic burn.

B18.2.1.1 Toxicity (References 2 and 5) - Gaseous nitrogen trifluoride is a minor irritant to the eyes and mucuous membranes. It is nearly odorless but is toxic when inhaled. The acute toxic effect is formation of methemoglobinemia, a modified form of hemoglobin incapable of transporting oxygen to the tissues. High levels of exposure may cause chemical cyanosis. Severe exposure in animals is accompanied by respiratory distress and collapse, however the recovery is complete in the survivors. In chronic intoxication there are no outward symptoms but damage to the liver and kidneys have been observed in rats. Direct contact with liquid nitrogen trifluoride will cause a cryogenic burn.

Additional information on the mean lethal dosage of NF_3 in animals can be found in Pisacane and Baroody, 1976 (See Reference 5).

B18.2.1.2 Exposure Limits

B18.2.1.2.1 Threshold Limit-Time Weighted Average (TLV-TWA) (References 2, 5, 7, 8 and 9) - For NF_3 the TLV-TWA is 10 ppm (30 mg/m^3) as established by the American Conference of Governmental Industrial Hygienists (ACGIH) and the corresponding permissible exposure limit (OSHA) is 20 ppm (58 mg/m^3). The IDLH value is 2000 ppm (5.8 g/m^3).

B18.2.1.2.2 Emergency Exposure Limits (EEL) (References 1, 4 and 10) - The emergency exposure limit (EEL) defines the "single brief accidental exposures without permanent toxic effects. These limits are not intended to replace accepted safe practices and should be accompanied by appropriate medical surveillance" (Reference 11). National Academy of Sciences, National Research Council (Reference 12) Committee on Toxicology recommends that the EEL for nitrogen trifluoride be the following:

10 min	2250 ppm (7.13 g/m^3)
30 min	750 ppm (2.38 g/m^3)
60 min	375 ppm (1.19 g/m^3)

B18.2.1.3 Special Medical Information - Nitrogen trifluoride is a pulmonary irritant and also produces kidney, liver and possible spleen injuries following sublethal exposures. Upon removal from the contaminated atmosphere, the methemoglobin formed in the blood as a result of exposure spontaneously reverts back to hemoglobin. Oxygen therapy should be started at once by trained personnel and preferable under the direction of a physician. This therapy should continue for periods up to 6 hours. Allow the patient to rest under close supervision until all possibility of secondary complications can be discounted. Be aware of possible anemia and impaired kidney functions and seek competent medical follow-up.

B18.2.2 Fire and Combustion Product Hazards (Reference 2)

Nitrogen trifluoride is a powerful oxidiser and can react violently with fuels, e.g., grease, oil, etc. The energy released from the combustion of fuels with NF_3 is much greater than with O_2. Burning rates of NF_3-hydrocarbon mixtures are much higher than for the corresponding O_2 mixtures, and the flammability ranges are broader. Like O_2, NF_3 can be mixed with gaseous fuels at room temperatures without reacting. However, exposing the mixtures to an ignition source such as a hot surface, a spark or flame can initiate a violent reaction. This violent reaction may be characterized as a rapidly propagating flame front, an explosion, or even a detonation depending upon the nature of the mixture and its composition, temperature, pressure, and degree of confinement.

Static testing of engines using nitrogen trifluoride produces hydrogen fluoride as the major exhaust product. Hydrogen fluoride is more toxic than nitrogen trifluoride and currently acceptable TLV is 3 ppm as compared to 10 ppm for NF_3. The EEL's (NAS/NRC 1968, Reference 12) of the exhaust gas product hydrogen fluoride are as follows:

10 min	20 ppm	18 mg/m^3
30 min	10 ppm	9 mg/m^3
60 min	8 ppm	7 mg/m^3

Nitrogen trifluoride is unlike fluorine in that it is relatively insoluble in water and does not react spontaneously with atmospheric moisture. However, an electric spark will ignite gaseous NF_3 in atmospheric moisture causing violent reaction (Reference 4).

B18.2.3 Explosion Hazards (Reference 4)

Nitrogen trifluoride is not shock sensitive. It is less reactive than fluorine and when dry, it will not detonate with a spark input. In a moist atmosphere and with a spark input, the following explosive reaction will proceed:

a. $H_2O + 2/3\ NF_3 = 2\ HF + 1/3\ NO + 1/3\ NO_2 + 33.9$ kcal (proceeds with brown flame at 572 K)

b. $H_2O + 2/3\ NF_3 = 2\ HF + 1/2\ O_2 + 1/3\ N_2 + 41.5$ kcal (thermodynamically favored in explosion)

Pressure rupture may occur when liquid nitrogen trifluoride is trapped in a closed system and refrigeration is not maintained. Liquid nitrogen trifluoride trapped between closed valves can cause violent rupture of the valves and associated piping. Storage and transfer equipment designed for supercritical pressures must take these considerations into account.

B18.2.4 Environmental Effects

Nitrogen trifluoride gas because of its relative inertness compared to other inorganic fluoride compounds will in time degrade and react with the soil and plant life. Damages to the plant and animal life are practically nil because of this degree of inertness and its relative insolubility in water (Reference 2). Its degradation products eventually form the nitrates and fluoride salts in reactions with the soil (See Section B18.1.3.4).

B18.3 RESERVED

B18.4 RESERVED

B18.5 MATERIALS AND EQUIPMENT COMPATIBILITY (Reference 15)

B18.5.1 Materials

The selection of materials suitable for nitrogen trifluoride systems must be governed by two important factors:

a. The resistance of the materials to nitrogen trifluoride attack

b. The material's mechanical strength and characteristics at cryogenic temperatures.

B18.5.1.1 Metals - Metals considered satisfactory for handling gaseous nitrogen trifluoride at temperatures up to 344 K (160°F) are as follows:

nickel	stainless steel
mild steel	Monel-K
copper	aluminum
inconel	

NOTE

These and other metals may also be suitable for use at higher temperatures and with specific operating conditions. For liquid nitrogen trifluoride the following metals are recommended:

aluminum	stainless steel
nickel	types 301, 304L,
copper	316, 321, 309Cb,
Monel-K	310, and 347

In low temperature service, stainless steel has been used successfully.

B18.5.1.2 Non-Metals - Polytetrafluoroethylene (Teflon, TF, KEL F-81, Halon TFE, or equivalent) are acceptable for use at moderate pressures and low flow rates with nitrogen trifluoride. TFE tape has been used extensively as a thread sealant for gaseous service.

Plastics acceptable for use with liquid nitrogen trifluoride under flow conditions are:

Carbon CJPS
polytetrafluoroethylene
PFA Teflon KEL F-81CTFE

These materials are also compatible for liquid flow system.

Glass has been used successfully in low temperature service.

B18.5.1.3 Lubricants - Lubricants suitable for use with nitrogen trifluoride at high-velocity compressed gas flow at moderate temperature below 450 K (350°F) are:

Fluorocarbon Oil (FC-75)
Petroleum Jelly (vaseline)
Light-weight Machine Oil (Walco '985)

B18.5.2 Equipment

B18.5.2.1 Containers - Liquid nitrogen trifluoride may be stored in either fixed or mobile tanks of approved design and materials. Storage and shipping containers designed for noncryogenic fluids should not be used for this service. To insure against material or fabrication defects, storage tanks should be tested in accordance with the provisions of applicable ASME, ASTM, and/or DOT specifications for pressure vessels. Materials used for pressure vessels containing liquid nitrogen trifluoride should be impact-tested in accordance with ASME Code (d), Paragraph UG-84, Section VIII (Reference 16).

Containers for shipping, storing, and transferring liquid nitrogen trifluoride should be fabricated in accordance with the physical and structural requirements dictated by the service for which it is intended. Containers for storing liquid nitrogen trifluoride may be constructed with three horizontal, concentric containers forming an outer shell, an intermediate shell, and an inner shell. As dictated by the user, the two inner shells should be constructed by either Monel or stainless steel. The inner shell contains liquid nitrogen trifluoride and the intermediate shell contains liquid nitrogen. The outer shell forms a vacuum jacket and may contain an insulating material of low thermal conductivity such as Santocel or Perlite.

The most common storage condition has involved the gas at ambient temperature and pressures up to approximately 7 MPa (1000 psig). Steel cylinders used as shipping containers also serve as ground storage vessels.

B18.5.2.2 Pumps and Hoses - Pumps for high pressure gaseous nitrogen trifluoride service should be designed, fabricated and installed in accordance with pertinent standards and codes.

Hoses for gaseous nitrogen trifluoride should conform to military or industrial specifications. Hoses may be fabricated of steel, copper, brass, polyethylene or tygon.

Hoses for liquid service should be of proper design, engineered specifically for this service. Copper and brass tubing has been employed successfully.

B18.5.2.3 Lights - Temporary, portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines. Flash lights or storage battery lamps, where permitted for use, should be of the safety types approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 17).

B18.5.2.4 Pipes and Fittings - Pipes and fittings for liquid nitrogen trifluoride should be fabricated from nickel, Monel, stainless steel (301, 321, 316, 347), copper, or aluminum. Welded and welded-flanged connections are recommended for all applications, especially for piping over 1 inch in diameter. All welds should be made using inert gas shielding; welds should be of butt type and should have full penetration. Usually, liquid nitrogen trifluoride lines will be insulated to prevent excessive heat transfer to the product. Metal tubing with compression AN type fitting of suitable material is also recommended in sizes up to and including 12.7 mm (1/2 in) O.D. Fittings should be used with soft copper or aluminum fittings rings and conical seals to prevent leakage. Swage type fittings with two piece compression swaged locking arrangement have been found suitable in sizes up to 12.7 mm (1/2 in) for use with gaseous and liquid nitrogen trifluoride cycling between temperature of the boiling point of the liquid and 300 K (80°F).

B18.5.2.5 Gaskets - Gaskets made from any of the following materials are recommended:

a. For gaseous service: aluminum, copper, lead, and polytetrafluoroethylene (TFE) "O-ring" seals have been used in high pressure gas service.

b. For liquid service: aluminum-2S, copper and nickel can be used.

B18.5.2.6 Pressure Gauges - Liquid nitrogen trifluoride equipment should be monitored through the use of approved pressure gauges as required. In order to minimize operating reading errors, all pressure gauges used for a common purpose should have identical scales.

B18.5.2.7 Valves - Materials recommended for valve fabrication are stainless steels, Monel, nickel or brass (zinc free). Forged type valves with metal bellows or diaphrams have proved to be the most reliable in both gaseous and liquid nitrogen trifluoride service and are used most often in such installations. The type of valve employed varies with application. Monel, teflon and stainless steel may be used for valve seats. They show excellent resistance to high pressure and high velocity flow conditions. Valves must be free of lubricant, grease, dirt and organic material prior to installation.

Valves designed for liquid or high pressure gaseous nitrogen trifluoride service, should allow for remote operating. Valve packing may be Teflon, copper braid with teflon or Kel-F. Valve plugs may be made using Monel.

B18.5.2.8 Venting Systems and Pressure Relief - Large storage containers should be insulated, vacuum-jacketed, nitrogen-cooled tanks. The vacuum-insulated space between the intermediate and outer shells should be equipped with either a rupture-disk or a pressure-relief device. The storage container itself should be of welded construction. The liquid nitrogen system should be equipped with an adequate vent line, a pressure-relief valve, and it should be equipped with some alarm or other means to indicate pressure build-up, but no automatic pressure-relief devices are to be installed on the inner nitrogen trifluoride product tank itself.

B15.5.2.9 Venting Systems and Pressure Relief - Safety-relief devices should be used with compressed nitrogen trifluoride gas.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-200.3°F	144.1 K
Freezing Point		-340.2°F	66.35 K
Density (Gas)		0.389 lb/ft	6.235 kg/m³
(Liquid)		96 lb/ft³ at -200°F	1.538 Mg/m³ at 144.1 K
Specific Gravity - Vapor (relative to STP air)		2.45	2.45
Critical Density		35.1 lb/ft³	562.3 kg/m³
Critical Pressure		646.7 psia	4.46 MPa
Critical Temperature		38.2°F	234 K
Vapor Pressure		14.7 psia at 880°F	101 kPa at 744 K
Coefficient of Viscosity	Kinematic		
	Absolute	0.0175 micropoise at 32°F	1.75×10^{-6} Pa·s at 273.15 K
Autoignition Temperature			N/A
Flash Point			N/A
Flammability Limits	Lower:		N/A
	Upper:		N/A

Figure B18-1 Physical Properties of Nitrogen Trifluoride (MW = 71.00 g mol)

CHAPTER B19
HYDROGEN PEROXIDE

B19.1 PROPERTIES

B19.1.1 Identification

Propellant-grade hydrogen peroxide contains from 90 (Type I) to 98 (Type II) percent hydrogen peroxide (H_2O_2), the remainder being water. Military Specification MIL-P-16005 "Propellant Hydrogen Peroxide" covers this grade (Reference 1). Hydrogen peroxide of reduced strength, 70 to 90 percent, is covered in MIL-H-22868 "Hydrogen Peroxide E-Stabilized" (Reference 2). The molecular weight of pure hydrogen peroxide is 34.01 g mol.

B19.1.2 General Appearance and Common Uses

High strength hydrogen peroxide is a clear, colorless liquid, slightly more viscous than water. Hydrogen peroxide is used as a rocket propellant oxidizer and as a monpropellant, as gas generator.

B19.1.3 Physical and Chemical Properties (Reference 3)

The physical properties of hydrogen peroxide at 70 to 90 percent, and 98 percent composition by weight are given in Figures B19-1, B19-2, and B19-3.

B19.1.3.1 Solubility - Hydrogen peroxide is miscible with water, and thus, reactivity can be quickly reduced by dilution with water. It is also miscible with a large number of water-soluble organic liquids, such as alcohols, glycols, acetates, acids, and ketones. It is nearly insoluble in petroleum ether, toluene, styrene, carbon tetrachloride, chloroform, kerosene, fuel oil, and gasoline.

B19.1.3.2 Stability - Pure hydrogen peroxide stored in proper containers decomposes at a very slow rate. In a 113.6 l (30 gal) aluminum shipping drum stored at about 294.3 K (70°F) less than 1 percent of the hydrogen peroxide decomposes in a year; in large tanks, the loss is as low as 0.1 percent per year. If stored in containers of unsuitable material or if contaminated, hydrogen peroxide can decompose very rapidly, releasing large amounts of heat and gas which may rupture the container.

The decomposition rate of hydrogen peroxide increases with temperature, approximately doubling for each temperature rise of 8.33 K (15°F). If the heat that is released cannot be dissipated, it may cause runaway decomposition. The effect of temperature on the rate at which hydrogen peroxide decomposes is shown in Table B19-1.

When a hydrogen peroxide solution is chilled, it is supercooled far below its true freezing point. As the solution freezes, a slush forms. Final solidification takes place at 218 K (-67°F). Solutions of more than 65 percent hydrogen peroxide (H_2O_2) contract on freezing and will not burst containers.

Table B19-1 Effect of Temperature on Decomposition of Hydrogen Peroxide

TEMPERATURE K (°F)	APPROXIMATE DECOMPOSITION RATE (at 1 atm pressure)
303 (85)	1% per year
339 (150)	Less than 1% per week
373 (212)	Less than 2% per day
414 (285)	Rapid with boiling (resulting in explosion if confined)

B19.1.3.2.1 Chemical Stabilizers - Certain organic and inorganic compounds, when added in small quantities to hydrogen peroxide, act as stabilizers. Some protection against minor contamination (of the order of 1 or 2 parts per million of contaminant) can be secured by the addition of stabilizers, but no additive will entirely counteract rapid decomposition if there is gross contamination. The use of excessive quantities of stabilizer should be avoided.

B19.1.3.2.2 Emergency Stabilization - The decomposition of hydrogen peroxide, at an accelerating rate, as evidenced by increasing temperature and gas evolution, may be brought under control by the following emergency stabilization procedure: Add 453.6 g (1 lb) of 85 percent phosphoric acid solution (in water) for each 378.5 l (100 gal) of hydrogen peroxide solution. Mixing is not necessary because natural turbulence will disperse the stabilizer.

If decomposition is controlled by phosphoric acid, the stabilized hydrogen peroxide solution may be stored in aluminum containers until consumed or otherwise disposed of. This solution must not be used in applications involving catalytic decomposition chambers, because the stabilizer will poison the catalyst.

B19.1.3.3 Reactivity - Hydrogen peroxide, a monpropellant and an active oxidizing agent, is an energy-rich material which can decompose, yielding water (steam), oxygen, and heat. It does not burn but vigorously supports combustion because of the oxygen it liberates upon decomposition. Because of its strong oxidizing nature, it can initiate the combustion of many organic materials such as wood, cotton, and paper. High strength hydrogen peroxide reacts under certain conditions with many organic compounds and may form detonable mixtures. It is hypergolic with many fuels. It decomposes rapidly on contact with many inorganic compounds, such as potassium permanganate and iron oxide. When decomposed by catalysts, it generates heat rapidly. If completely decomposed, solutions stronger than 67 percent hydrogen peroxide by weight generate enough heat to raise the temperature of the solution to its boiling point and to convert all the decomposition products into vapor. Stronger solutions, therefore, produce superheated vapors whose temperature depends upon the concentration of the peroxide and the characteristics of the confining equipment. Hydrogen peroxide does not concentrate as a result of adiabatic decomposition. There is a possibility that contaminated hydrogen peroxide will explode under normal operating conditions. Mixtures of hydrogen peroxide and organic materials (sugar, strach, alcohols, petroleum products) in sufficient concentration form combinations that are sensitive, and can explode from shock or application of heat. In many cases such oxidizable materials show no evidence of reaction on mixing with hydrogen peroxide. The mixture may not appear dangerous, but the hazard is still present.

The contamination of hydrogen peroxide changes it decomposition rate. If it is contaminated with small quantities of active decomposition promoting catalysts or with larger quantities of less active catalysts, runaway decomposition may follow, possibly rupturing the container. Alkaline contamination promotes active decomposition. Most of the heavy metals are active decomposition catalysts. Decomposition also can be initiated by dirt, cigarette ashes, rust, etc.

B19.1.3.4 Environmental Fate - Hydrogen peroxide is a strong oxidizing agent and will react with many different compounds, including organics, nitrogen and sulfur compounds and metals. Hydrogen peroxide oxidizes a number of functional groups of organic material in surface waters. Further oxidation of short chain hydrocarbons to carbon dioxide and water is possible. The major transportation mode of hydrogen peroxide in the environment would be by water, although in water, in the presence of impurities, it quickly decomposes into non-toxic residuals.

B19.2 HAZARDS

B19.2.1 Health Hazards

B19.2.1.1 Toxicity - Contact with hydrogen peroxide liquid, mist, or vapor, can cause irritation, prolonged skin contact causes bleaching and burns. The eyes must be promptly flushed with water after contact or severe delayed damage can occur. Inhaling the vapor can cause respiratory tract irritation.

B19.2.1.2 Exposure Limits

B19.2.1.2.1 Threshold Limit Values-Time Weighted Average (TLV-TWA) (References 4, 5 and 6) - The TLV-TWA for hydrogen peroxide is given by the ACGIH as 1 ppm (1.5 mg/m^3) in air. The Threshold Limit Value-Short Term Exposure Limit (TLV-STEL) is 20 ppm (3 mg/m^3). The IDLH is 75 ppm (113 mg/m^3).

B19.2.2 Fire and Combustion Product Hazards

Hydrogen peroxide is not flammable by itself but it is a powerful oxidizing agent. It may react with flammable materials and generate enough heat to cause ignition. It actively supports combustion resulting in a "flare" fire that may terminate in an explosion. Liquid spillage of hydrogen peroxide on ordinary clothing or shoes (particularly in the presence of dirt or grease) can cause a fire.

B19.2.3 Explosion Hazards

An explosion hazard exists whenever hydrogen peroxide is contaminated, especially when it is mixed with fuels or organic solvents. Small amounts of materials containing catalysts (silver, lead, copper, chromium, mercury, and iron oxide rust) cause immediate decomposition. If hydrogen peroxide is stored in a container made of improper materials, or if it is accidentally contaminated with rust or another catalyst, it decomposes rapidly, releasing large amounts of heat and gas. Contaminated hydrogen peroxide can decompose so rapidly that a vented container may rupture. This situation is generally preceded by a progressive rise in temperature.

At atmospheric pressure, vapor concentrations of hydrogen peroxide above 26 percent by volume (40 percent by weight) become explosive and can be ignited by a spark in a temperature range below the boiling point of the liquid. The explosive region of the vapor occurs above a liquid concentration of 74 percent of greater.

If confined, liquid hydrogen peroxide may propagate an explosion. Accidental mixing of liquid hydrogen peroxide and liquid fuel will result in an explosion unless diluted with water to less than 30 to 35 percent concentration.

B19.2.4 Environmental Effects

A large spill of hydrogen peroxide would have serious consequences in a surface water environment, as large concentrations would tend to have biocidial properties. Small spills would oxidize organics, precipitate metals and so on. Generally, the hydrogen peroxide would dissipate in water into harmless by-products.

B19.3 RESERVED

B19.4 RESERVED

B19.5 MATERIALS AND EQUIPMENT COMPATIBILITY

Hydrogen peroxide decomposes slowly at ordinary temperatures and builds up pressure if the container is closed. The rate of decomposition doubles for each 10 K (18°F) rise in temperature and becomes self-sustaining at 413.7 K (285°F). Hydrogen peroxide in concentrations up to about 90 percent does not readily detonate. Higher concentrations or elevated temperatures may facilitate detonation (Reference 7).

Contact with most organic or readily oxidizable materials and combustibles causes fires and explosions. Contact with iron, copper, brass, bronze, chromium, zinc, lead, manganese, silver, and other catalytic metals (or their salts) causes rapid decomposition with evolution of oxygen gas and heat which may increase container pressure. It should be noted that if yellow or red phosphorous is incompletely immersed while undergoing oxidation in hydrogen peroxide solutions, heating at the air-solution interface can ignite the phosphorous and lead to a violent reaction.

Hydrogen peroxide must be stored in drums or tanks which are manufactured from properly selected materials and are of approved design and construction. Storage, transfer, and test areas must be kept neat and clean and absolutely free from combustibles. All leaks and spills must be flushed away at once with large amounts of water. These areas must be inspected frequently; safety regulations must be strictly enforced.

B19.5.1 Materials

In selecting materials for fabricating equipment, the effect the hydrogen peroxide will have on the material is of much less importance than the effect of the material on the hydrogen peroxide. As a general rule, the only materials for construction or for surface treatments that should be used are those which have been found to be compatible. Emphasis should be placed on compatibility testing under actual conditions of use and hydrogen peroxide concentration. The following tables give the relative compatibility of various materials with 90 percent concentration. The tables are not complete, but are specific in order to point out gross differences in compatibility due to slight changes in material. In the tables, materials for service with hydrogen peroxide are grouped in four general classes according to their compatibility and use, as follows:

a. Class 1 - These materials show no effect upon the hydrogen peroxide and the material itself is not affected.

   Uses - Bulk storage tanks, rail and truck tanks, and shipping drums.

b. Class 2 - Materials that have a minor effect, such as increasing the decomposition rate of the hydrogen peroxide or staining of the material. Typical of these materials are the 300-series stainless steel alloys. Contact time allowed for such materials is determined by material finish: the smoother the finish, the better the compatibility. A contact exposure guide is 4 hours at 294.8 K (160°F) and 1 week to 4 weeks at 294.3-295.4 K (70-72°F). Longer contact periods are allowable when specially stabilized hydrogen peroxide is being used.

   Uses - Typical examples are rocket engine chambers, high pressure test tanks, high pressure feed lines, solenoid valves, and other components.

c. Class 3 - Materials that upon direct contact, initiate a strong reaction, or decomposition effect upon the hydrogen peroxide or vice versa. Maximum contact time should be 1 minute at 344.3 K (160°F) or 1 hour at 294.3 K (70°F) prior to immediate use.

   Uses - Rocket engine chambers and gas generator and thrust motor housings.

d. Class 4 - Materials such as copper, lead, 400-series stainless steel, and other alloys, plastics, and lubricants should never be used in hydrogen peroxide systems because of the violence of the effect they produce.

B19.5.1.1 **Metals** - Tables B19-2, B19-3, and B19-4 list the relative compatibility of pure metals (alloys with controlled impurities), aluminum alloys, and stainless steel alloys with concentrated hydrogen peroxide.

Table B19-2 Compatibility of Pure Metals with H_2O_2 (90 Wt %)

MATERIAL	CLASS
Aluminum	1
Beryllium	4
Cadmium	4
Chromium	4
Copper	4
Iron	4
Lead	4
Magnesium	4
Mercury	4
Molybdenum	4
Nickel	4
Platinum	4
Silver	4
Tantalum	1
Tin	2
Titanium	4
Tungsten	4
Zinc	4
Zirconium	1

Table B19-3 Compatibility of Pure Aluminum Alloys with H_2O_2 (90 Wt %)

MATERIAL	CLASS
1060	1
1100	2
1260	1
2017 chromic acid anodized	4
2017 sulfuric acid anodized	3
2024 chronic acid anodized	4
2024 sulfuric acid anodized	3
3003	2
4043	2
5052	2
5054	2
5056	2
5254	1
5652	1
6061	2
6063	2
40E	3
43	2
214B	2
218	4
355F	4
B356	1

Table B19-4 Compatibility of Stainless Steel Alloys with H_2O_2 (90 Wt %)

MATERIAL	CLASS
302	2
304	2
310	2
316	2
316 porous	4
329	3
347	2
AM355	3
410	4
416	4
420	4
440	4
440C	3
17-7PH, 37-43Rc	2
Durimet 20	3

B19.5.1.2 Non-Metals - Table B19-5 lists the relative compatibility of selected plastic materials with concentrated hydrogen peroxide.

Table B19-5 Compatibility of Plastic with H_2O_2 (90 Wt %)

MATERIAL	CLASS
Aclar	1
Acrylon rubber BA-12	4
Bisolin No. 50	2
Buna N	4
Fluorel 2141	3
Fluorosilicone LS-53	2
GE 12601	4
GE 12650	2
Geon 404 (yellow)	3
Halgene	2
Hypalon S-2	4
Kel-F	1
Kel-F800	1
Kel-F820	2
50% Kel-F 5500/50% Kel-F 800	1
Koroseal 116, or equivalent	3
Koroseal 700, or equivalent	2
Mylar A	1
Nylon	4
Phenol-formaldehyde	4
Plexiglas	4
Polethylene	2
Polystyrene	2
Saran	2
Silastic 152	3
Silastic 9711	2
Silicone 407-B-217-1	3
Silicone Y-1749	2
Polytetrafluoroethylene, TFE	1
Tygon 360A	2
Vicone	2
Vinyl 29139	2
Vinylite VU1940	3
Viton A (271-7) (770545)	2
Viton A (V271-7)	4
Viton B 805	3

B19.5.1.3 Lubricants - Table B19-6 shows the relative compatibility of various lubricants with concentrated hydrogen peroxide.

Table B19-6 Compatibility of Lubricants with H_2O_2 (90 Wt %)

MATERIAL	CLASS
Arochlo 1221	4
Bardahl	4
Ceresin Wax	4
Fluorolube FS	2
Fluorolube Heavy Grease 1021-4	2
Halocarbon Light Oil 11-14	2
Halocarbon Medium Oil 11-21	2
Kel-F Alkane	2
Kel-F Light Oil No. 1	2
Kel-F No. 90 Grease	2
Mineral oil	4
Lubriseal	4
Petrolatum	4
Silicone Oil DC-7	4
Tributyl Phosphate	4

B19.5.1.4 Miscellaneous Materials - Table B19-7 shows the compatibility of various ceramic materials with concentrated hydrogen peroxide.

Table B19-7 Compatibility of Miscellaneous Materials with H_2O_2 (90 Wt %)

MATERIAL	CLASS
Al-Si-Mag Porcelain	2
Aluminum silicate	2
Alundum La 116	2
Boron Nitride	4
Carbaloy	4
Ceramic AB-2	2
Ceramic AL-200	2
Graphite	4
Karbate	4
KT Silicon Carbide	2
Norbide	2
Pyroceram	1
Synthetic sapphire (polished)	1
Zirconium silicate	2

B19.5.2 Equipment

B19.5.2.1 Containers - Local requirements shall determine the size and number of storage containers.

The horizontal-type atmospheric-pressure tanks for hydrogen peroxide storage are normally used, but other well designed and well built types of tanks are also suitable (Reference 13). Care should be taken during fabrication to prevent occulsions of iron and other impurities which may cause catalytic decomposition of the stored hydrogen peroxide.

B19.5.2.2 Pumps and Hoses - Pumps manufactured from wrought or forged 300-series stainless steel (304, 316, 321, and 347), or aluminum alloys B356, 356, or 43 should be used for pumping hydrogen peroxide. Cast stainless steel should be avoided because it is subject to chromium leaching, which seriously contaminates the propellant and hastens decomposition. Self-priming pumps should be used for transferring hydrogen peroxide from tank cars or other tanks with top outlets. Stainless steel mechanical seals with glass-filled PTFE plastic and ceramic faces are recommended. Where used, packing should be lubricated with a fluorinated hydrocarbon. Pump packing should contain no unsuitable materials. (See Tables B19-3 through B19-5 for compatible materials.)

Hoses, if used, should be constructed of approved Class 1 or 2 materials listed in Section B19.5.1.

B19.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines.

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 14).

B19.5.2.4 Pipes and Fittings - The most compatible piping in hydrogen peroxide service is 1060 aluminum, and this is recommended, particularly if the liquid remains in static contact with it. Where greater strength is required, other than Class 1 or Class 2, aluminums may be used if the classification permits. Piping of 300-series stainless steel may also be used in certain applications. Tubing and fittings of 300-series stainless steel are used almost exclusively for test systems. Welded and flanged construction is recommended. Threaded fittings and connections should be avoided, but if they must be used, the male threads should be wrapped with polytetrafluoroethylene tape except for the first three threads. Hydrogen peroxide piping should be clearly identified and tested at one and one half times working pressure (References 13 and 15).

B19.5.2.5 Gaskets - Gaskets may be made of any of the following materials: silicon rubbers such as DC 9711, polytetrafluoroethylene (PTFE), polytrifluorochloroethylene, and a combination of stainless steel and polytetrafluoroethylene plastic (Flexitallic) the last generally being used in pressure systems. Whenever possible, contact between dissimilar metals should be avoided to prevent electrolytic action.

B19.5.2.6 Pressure Gauges - Gauges should be approved oxygen-clean types with a stainless steel bourdon tube. A "gas-leg" or a diaphragm protector is recommended to prevent direct exposure of gauges to propellant grade hydrogen peroxide. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales. Blowout protected gauges should be specified.

B19.5.2.7 Valves - Valves should be designed so that hydrogen peroxide is not trapped in any part of the valve. Self-venting ball valves are recommended. Gate and plug valves may be used but must be modified to insure venting. Globe valves are approved for use with hydrogen peroxide, as well as Y-type valves. A PTFE plastic must be used to prevent metal-to-metal contact of plug and seat. Materials approved for gaskets may be used as a valve packing.

B19.5.2.8 Venting Systems and Pressure Relief - Safety-relief devices for tanks are discussed in Section B19.6, CPIA 394. All storage tanks must be vented and the vent should be located so as to adequately protect personnel and equipment. A vent filter is required to prevent dirt from entering the tank.

B19.5.2.9 Grounding - All tanks should be properly grounded.

LIST OF REFERENCES

1. MILITARY SPECIFICATION, PROPELLANT HYDROGEN PEROXIDE, MIL-P-16005E, Department of Defense, 31 January 1968.

2. MILITARY SPECIFICATION, HYDROGEN PEROXIDE E-STABILIZED, MIL-H-22868, Department of Defense, 21 March 1961.

3. LIQUID PROPELLANT MANUAL, Chemical Propulsion Information Agency, Laurel, MD, CPIA/M4.

4. THRESHOLD LIMIT VALUES FOR CHEMICAL SUBSTANCES AND PHYSICAL AGENTS IN THE WORK ENVIRONMENT, American Conference of Governmental Industrial Hygienists, Cincinnati, OH, 45211, 1983-84. (Published Annually)

5. Sittig, Martin, HANDBOOK OF TOXIC AND HAZARDOUS CHEMICALS, 729 pp., Noyes Publications, Park Ridge, NJ, 1981.

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8. MILITARY SPECIFICATION, GLOVES, PROTECTIVE, FUEL AND OXIDIZER, Department of Defense, MIL-G-43196 (3), 5 July 1966.

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13. ASME BOILER AND PRESSURE VESSEL CODE, Section VIII, "Unfired Pressure Vessels," American Society of Mechanical Engineers, New York, 1965.

14. NATIONAL FIRE CODES, Vol. 6, "National Electrical Code," NFPA 70-1981, National Fire Protection Association, Quincy, MA, 1983.

15. MILITARY STANDARD, COLOR CODE FOR PIPELINES AND FOR COMPRESSED GAS CYLINDERS, MIL-STD-101A, 13 August 1961 or latest revision.

16. CODE OF FEDERAL REGULATIONS, Title 49, TRANSPORTATION, Parts 178 to 199, 1982.

17. CODE OF FEDERAL REGULATIONS, Title 46, SHIPPING, 1982.

18. CODE OF FEDERAL REGULATIONS, Title 40, PROTECTION OF ENVIRONMENT, Parts 260 to 265, Vol. 45, 1982.

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		257.9°F	398.65 K
Freezing Point		-40.5°F	232.9 K
Density (Gas)			
(Liquid)		10.8 lb/gal at 68°F	1.29 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)			
Critical Density			
Critical Pressure			
Critical Temperature			
Vapor Pressure		0.11 psia at 68°F	0.75 kPa at 293 K
		0.37 psia at 104°F	2.55 kPa at 313 K
		1.05 psia at 140°F	7.24 kPa at 333 K
		2.62 psia at 176°F	18.07 kPa at 353 K
Coefficient of Viscosity	Kinematic	0.95 centistokes at 68°F	9.5×10^{-7} m^2/s at 293 K
	Absolute	1.23 centipoises at 68°F	1.23×10^{-3} Pa·s at 293 K
Explosive Range			
Autoignition Temperature			
Flash Point			
Flammability Limit	Lower:	26 volume %hydrogen peroxide in air	
	Upper:	100 volume %hydrogen peroxide in air	

Figure B19-1 Physical Properties of Hydrogen Peroxide 70% Wt

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		286.1°F	414.3 K
Freezing Point		11.3°F	261.65 K
Density (Gas)			
(Liquid)		11.6 lb/gal at 68°F	1.390 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)			
Critical Density			
Critical Pressure			
Critical Temperature			
Vapor Pressure		0.05 psia at 68°F	0.345 kPa at 293 K
		0.17 psia at 104°F	1.17 kPa at 313 K
		0.52 psia at 140°F	3.59 kPa at 333 K
		1.38 psia at 176°F	9.52 kPa at 353 K
Coefficient of Viscosity	Kinematic	0.905 centistokes at 68°F	9.05×10^{-7} m^2/s at 293 K
	Absolute	1.26 centipoises at 68°F	1.26×10^{-3} Pa·s at 293 K
Explosive Range		26 - 100%by volume in air	
Autoignition Temperature			
Flash Point			
Flammability Limit	Lower: Upper:	26 volume %hydrogen peroxide in air 100 volume %hydrogen peroxide in air	

Figure B19-2 Physical Properties of Hydrogen Peroxide 90% Wt Type I

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		302.4°F	423.4 K
Freezing Point		31.2°F	272.7 K
Density (Gas)			
(Liquid)		12.1 lb/gal at 68°F	1.448 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)			
Critical Density			
Critical Pressure		3130 psia	21.58 MPa
Critical Temperature		855°F	730.4 K
Vapor Pressure		0.03 psia at 68°F	0.207 kPa at 293 K
		0.10 psia at 104°F	0.69 kPa at 313 K
		0.33 psia at 140°F	2.28 kPa at 333 K
		0.93 psia at 176°F	6.41 kPa at 353 K
Coefficient of Viscosity	Kinematic	0.863 centistokes at 68°F	8.63×10^{-7} m^2/s at 293 K
	Absolute	1.25 centipoises at 68°F	1.25×10^{-3} Pa·s at 293 K
Explosive Range		26%- 100%by volume in air	
Autoignition Temperature			
Flash Point			
Flammability Limit	Lower: Upper:	26 volume %hydrogen peroxide in air 100 volume %hydrogen peroxide in air	

Figure B19-3 Physical Properties of Hydrogen Peroxide 98% Wt Type II

CHAPTER B20
NITROGEN

B20.1 PROPERTIES

B20.1.1 Identification

Liquid and gaseous nitrogen (N_2) are used in their elemental forms. Nitrogen has a minimum purity of 99.5 percent, oxygen being the major impurity. The applicable specifications are in the MIL-P-27401B and MIL-V-2/11B (References 1 and 2). The molecular weight of nitrogen is 28.014 g mol.

B20.1.2 General Appearance and Common Uses

High purity liquid nitrogen is a colorless transparent liquid, slightly lighter than water. Because of its low boiling point, it is usually boiling vigorously. Uninsulated containers are generally frosted on the outside. Gaseous nitrogen is colorless, odorless and tasteless. Gaseous nitrogen is used as an inert atmosphere in various process controls. Liquid nitrogen is used as an inert cooling agent.

B20.1.2 Physical and Chemical Properties

Refer to Figure B20-1 for the physical properties of liquid and gaseous nitrogen.

B20.1.3.1 Solubility - Most common solvents are solid at liquid nitrogen temperatures. Liquid nitrogen is miscible with liquid oxygen and liquid fluorine at all temperatures. Gaseous nitrogen is only slightly soluble in water.

B20.1.3.2 Stability - Liquid and gaseous nitrogen are stable to shock, heat and electrical spark. At ordinary temperatures in properly designed containers, the evaporation rate of liquid nitrogen is approximately as follows:

Tank Volume (Gallons)	Loss in 24 Hours (Percentage)
50 (0.189 m^3)	3.0
150 (0.568 m^3)	2.1
500 (1.89 m^3)	1.0
2000 (7.57 m^3)	0.5

B20.1.3.3 Reactivity - Nitrogen in either gaseous or liquid form is inert. It is noncorrosive and will undergo chemical reactions only at very high temperatures. It is nonflammable and does not prevent a fire hazard.

B20.1.3.4 Environmental Fate - Nitrogen becomes part of the atmosphere.

B20.2 HAZARDS

B20.2.1 Health Hazards

B20.2.1.1 Toxicity - Nitrogen is nontoxic at atmospheric pressure. Under increased pressure it has been reported to cause "nitrogen narcosis" and to be involved in the formation of gas bubbles causing "bends" when decompression is inadequate.

Nitrogen leaks in confined areas pose a threat to personnel because it is an odorless, tasteless, asphyxiant. The victim is not aware that his oxygen supply is insufficient to sustain life. Entry into a suspected enriched nitrogen atmosphere is not permitted without positive pressure self-contained breathing apparatus until an oxygen analysis shows at least 16-17 percent by volume oxygen.

Frostbite can result from liquid nitrogen contacts with the skin.

B20.2.1.2 Exposure Limits

B20.2.1.2.1 Threshold Limit Values-Time Weighted Average (TLV-TWA) - None established. With respect to oxygen displacement, (Oxygen at 16-17 percent by volume at 760 mm Hg. total pressure is the generally accepted minimum for a work environment).

B20.2.1.3 Special Medical Information - For first aid information, see CPIA 394, Appendix E20.

B20.2.2 Fire and Combustion Product Hazards

Gaseous and (pure) liquid nitrogen present no fire hazard. However, it is critical to avoid contamination with combustible materials or oxidizers if the nitrogen is to be used for pressurizing propellant systems. In emergency situations, liquid nitrogen may be used as a fire-extinguishing agent, since it acts to exclude air or oxygen by forming an inert gas blanket.

B20.2.3 Explosion Hazards

Gaseous nitrogen will not explode and may even be used to dilute an explosive atmosphere to below the lower explosive limit. If exposed to external heating, cylinders containing gaseous nitrogen may pressure rupture. Therefore, DOT specification cylinders are fitted with safety devices.

Pure liquid nitrogen presents no explosion hazard. Undetected contamination with combustibles or oxygen could result in a serious explosion if the nitrogen is introduced into a closed system containing substances with which the contaminants react. In transfer operations, liquid nitrogen should not be exposed to air, because oxygen from the atmosphere will condense in the liquid nitrogen.

Pressure rupture may occur when liquid nitrogen is trapped in a closed system and refrigeration is not maintained. Nitrogen cannot be kept as a liquid if its temperature rises above 126.2 K (-232.6°F), regardless of confining pressure. Liquid nitrogen trapped between valves can cause the pipe or tube to rupture violently. Loss of refrigeration can cause a storage tank to rupture if the pressure is not relieved by suitable devices.

B20.2.4 Environmental Effects

None.

B20.3 RESERVED

B20.4 RESERVED

B20.5 MATERIALS AND EQUIPMENT COMPATIBILITY

When handling or storing liquid nitrogen, two important factors must be taken into account: (a) Liquid nitrogen is extremely cold, and (b) it is an asphyxiant. It must, therefore, be stored in fixed or mobile containers of approved design, materials and construction and suitably housed.

The installation requirements for pressurized-gas systems are very stringent. Particularly important are the selection of materials and equipment, the use of adequate fabrication procedures and proper maintenance. Personnel assigned to operate these systems must be qualified and familiar with the equipment and proper operating procedures.

B20.5.1 Materials

When subjected to the temperature extremes of liquid nitrogen service, many materials undergo a marked physical change. The extent of the change for a given material should be known before it is specified for low temperature use. Metals used with nitrogen should be able to withstand impact shock at low temperatures, as well as the stresses produced by the extremely low temperature of liquid nitrogen. Their selection will be based primarily on the intended uses, since neither corrosivity nor reactivity are factors for consideration.

The prime consideration in selecting materials to be used with high-pressure gases is their strength. Reactivity or corrosivity are not problems with propellant grades of nitrogen, but are considerations in the maintenance of purity and cleanliness for the end use of the nitrogen.

B20.5.1.1 Metals - Ordinary carbon steels and most ferritic and martensitic alloy steels are unsuitable for liquid nitrogen service because they lack ductility at low temperatures below 244 K (-20°F). The following metals are satisfactory for this service:

Austenitic chrome-nickel steels	nickel steel (9%)
stainless steel, 18-8 series	copper-silicon alloys
copper	Monel
brass	aluminum
bronze	shredded lead
	titanium

These metals and carbon steels may be used with gaseous nitrogen near ambient temperatures.

B20.5.1.2 Non-Metals - Non-metal materials for liquid nitrogen service must also be selected to withstand low temperatures. The following nonmetals are suitable for this service and may be used singly or in any combination:

a. Tetrafluoroethylene (TFE Teflon, TFE Halon, or equivalent)

b. Polychlorotrifluoroethylene (Kel-F, Halon CTF, or equivalent)

c. Selected types of graphite

The following non-metals are suitable for use with gaseous nitrogen:

synthetic and natural rubbers
Teflon
Kel-F
DC-55
Fluorolube (LG-160)

B20.5.1.3 Lubricants - Materials used in handling nitrogen must be free of grease, oil and other combustible materials. Use special lubricants, such as the fluorolubes or perfluorocarbons, unless aluminum is present.

B20.5.2 Equipment

B20.5.2.1 Containers - Liquid nitrogen may be stored in stationary or mobile tanks of approved materials and construction. Gaseous nitrogen may be stored and shipped in cylinders conforming to DOT specifications. Storage tanks and cylinders should be tested as required by applicable provisions of the ASME Unfired Pressure Vessel Code (Reference 3) or DOT specification to ensure against defects in material or fabrication. Materials used for pressure vessels used at temperatures lower than 244 K (-20°F) should be impact-tested in accordance with Paragraph UG-84, Section VIII, of Reference 3.

Containers for shipping, storing and transferring liquid nitrogen may be fabricated in accordance with any standard which meets the structural requirements for that container. Vacuum-insulated tanks should be used for storage. The insulated area between the inner and outer shells should be maintained under vacuum and should have a pressure-relief device or a rupture disc. The storage tank itself should be of welded construction. It should be equipped with a rupture disc and a pressure-relief valve that has an adequate vent line. The vents should discharge to the atmosphere. Bottom outlets are recommended for storage tanks, as they materially simplify the transfer-system design and the selection of pumping equipment.

Storage containers for nitrogen gas should comply with one of the following:

a. Designed, constructed, and tested in accordance with appropriate requirements of the ASME Boiler and Pressure Vessel Code, Section VIII, Unfired Pressure Vessels (Reference 3).

b. Designed, constructed, tested, and maintained in accordance with DOT regulations and specifications.

B20.5.2.2 Pumps and Hose - Pumps for high-pressure gaseous nitrogen service should be designed, fabricated, and installed in accordance with the pertinent standards and codes.

Since the liquid nitrogen storage tanks may be designed with bottom outlets, reciprocating-type or flooded-suction centrifugal pumps may be used when gravity-flow or pressure transfer is not applicable. A graphite-braided Teflon packing is recommended as a pump-shaft seal.

For gaseous nitrogen, flexible hose should conform to military or industrial specifications. Unless otherwise specified in the governing code, a minimum safety factor of 4 to burst is required. Flexible hose should only be employed when the use of rigid tubing is impractical. The hose should be of minimum length. When long runs are unavoidable, hose reels and other protective devices, including hold-down weights and chain ties at end connections should be incorporated.

Hoses for liquid nitrogen use should be of proper design, engineered specifically for this service.

B20.5.2.3 Lights - Any lighting which meets local electrical codes will be adequate in areas where nitrogen is stored or transferred (Reference 4).

B20.5.2.4 Pipes and Fittings - The pipes and fittings should be of approved materials and construction and should be hydrostatically tested at specified pressures. The use of welded and flanged connections is recommended whenever possible. Threaded connections are permissible when other methods are not feasible. TFE tape thread sealant is recommended.

Tubing may also be used for high-pressure gaseous nitrogen service. Tubing connections will be made with standard AN or industrial flared fittings of like materials. Stainless steel fittings should be used with stainless steel tubing to prevent galvanic action of dissimilar metals. The fabrication of pipe or tube anchors, hangers, supports and tie-downs should be in accordance with the American Society of Mechanical Engineers requirements for pressure piping. The fabrication techniques and selection of materials for the installation and testing of piping systems shall be in accordance with the ASME codes. All high-pressure systems should be properly marked and identified.

B20.5.2.5 Gaskets - Gaskets may be made of the materials listed in Section B20.5.1. In high-pressure systems, "O" rings may also be used. Gaskets may be flanged, serrated, laminated, corrugated or ring-joint types.

B20.5.2.6 Pressure Gauges - Liquid nitrogen equipment should be monitored with pressure gauges of approved type as required. Standard-type pressure gauges incorporating compatible materials will be used in gaseous nitrogen systems. In addition, gauges used in gaseous nitrogen systems should have plastic faces and blow-out discs. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B20.5.2.7 Valves - Extended stem gate, globe and ball valves are satisfactory for liquid nitrogen service. All valve packing and gaskets should be of approved materials.

For gaseous nitrogen all gate and globe valves 6.35 cm (2.5 in) in size and larger should be made out of the outside-screw, rising-stem type, whenever this is practicable; valves under 6.35 cm (2.5 in) may be of the inside-screw, rising-stem type. Plug and valves may be used instead of gate or globe types. Plug valves for high-temperature service should be designed to prevent galling by making the plug and the valve body of different materials, by treating the plug to give it different physical properties, by welding hardface overlays to the surface, or by using special mechanical designs. Every valve should be plainly marked to identify the gas and valve function.

B20.5.2.8 Venting Systems and Pressure Relief - When the maximum allowable inlet pressure to one or more of the pressure-reducing devices is greater than the piping systems maximum allowable operating pressure, one or more pressure-relief or other safety devices should be included, e.g., automatic shut-off valves. Suitable protective measures should be taken to prevent injury or damage resulting from the discharge of gases from those safety devices.

The relief valves should have a combined discharge capacity ensuring that the pressure rating of the lower pressure piping system will not be exceeded as the result of an equipment failure, for instance, the pressure-reducing valves. Pressure-reducing and relief devices should conform to the valve requirements for the specified service conditions.

All lines and vessels in which liquid nitrogen may be trapped between closed valves should be equipped with pressure relief valves. Associated vessels and lines must also be equipped with rupture discs in parallel with the relief valve.

<u>LIST OF REFERENCES</u>

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2. MILITARY SPECIFICATION SHEET, VALVE, CYLINDER, GAS: ARGON, HELIUM, NITROGEN, NEON AND XENON, (INERT-OIL FREE), Outlet 581, MIL-V-2/11B, Department of Defense, 9 September 1970.

3. ASME BOILER AND PRESSURE VESSEL CODE, Section VIII, "Unfired Pressure Vessels," American Society of Mechanical Engineers, New York, 1965.

4. NATIONAL FIRE CODES, Vol. 6, "National Electrical Code," NFPA 70-1981, National Fire Protection Association, Quincy, MA, 1983.

5. CODE OF FEDERAL REGULATIONS, Title 49, TRANSPORTATION, Parts 178 to 199, 1982.

6. CODE OF FEDERAL REGULATIONS, Title 46, SHIPPING, 1982.

7. CODE OF FEDERAL REGULATIONS, Title 40, PROTECTION OF ENVIRONMENT, Parts 260-265, Vol. 45, 1982.

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-320.4°F	77.37 K
Freezing Point		-346°F	63.15 K
Density (Gas)		0.0712 lb/ft^3 at 70°F	1.14 kg/m^3 at 294.15 K
(1 atm) (Liquid)		50.46 lb/ft^3 at -320°F	0.808 Mg/m^3 at 77.35 K
Specific Gravity - Vapor (relative to STP air)		0.967	0.967
Critical Density		2.62 lb/gal	0.313 Mg/m^3
Critical Pressure		493 psia	3.40 MPa
Critical Temperature		-232.5°F	126.2 K
Vapor Pressure		20 psia at -316°F	138 kPa at 79.8 K
		52 psia at -298°F	358 kPa at 89.8 K
		113 psia at -280°F	779 kPa at 99.8 K
		214 psia at -262°F	1.48 MPa at 109.8 K
Coefficient of Viscosity	Kinematic (liquid)	2 centistokes at 14.7 psia	2 x 10^{-6} m^2/s at 101 kPa
	Absolute (gas)	0.01778 centipoises at 14.7 psia	1.778 x 10^{-5} Pa·s at 101 kPa
		0.01809 centipoises at 294 psia	1.809 x 10^{-5} Pa·s at 2.027 MPa
		0.01916 centipoises at 1029 psia	1.916 x 10^{-5} Pa·s at 7.095 MPa
Autoignition Temperature		Non flammable	
Flash Point		Non flammable	
Flammability		Non flammable	

Figure B20-1 Physical Properties of Nitrogen (MW = 28.014)

CHAPTER B21
NOBLE GASES (HELIUM)

B21.1 PROPERTIES

B21.1.1 Identification

The noble gases are comprised of the following elemental gases: helium, neon, argon, krypton and xenon. These elemental gases are positioned to the far right on the periodic table (References 1 through 9).

Argon, helium and liquid neon are covered by the following military specifications:

Argon - MIL-P-27415
Helium - MIL-P-27407
Liquid Neon - MIL-P-87932

B21.1.2 General Appearance and Common Uses

Helium, neon, argon, krypton and xenon are all colorless gases which are (with the exception of xenon) inert. These elemental gases are used most commonly as pressurizing agents.

B21.1.3 Physical and Chemical Properties

Refer to Figure B21-1 for the physical properties of helium.

B21.1.3.1 Solubility - Helium, neon, argon, krypton and xenon are slightly soluble in water. Neon is also soluble in liquid oxygen. Helium is absorbed by platinum.

B21.1.3.2 Stability - The elemental gases (helium, neon, argon, krypton and xenon) are inert and chemically stable.

B21.1.3.3 Reactivity - The noble gases (helium, neon, argon, krypton and xenon) are inert, non-flammable and will not support combustion. Helium, neon, argon and krypton will not combine with other materials. However, it is reported that xenon does form compounds such as xenon hydrate, sodium perxenate and xenon diflouride.

B21.1.3.4 Environmental Fate - The noble gases are dissipated into the atmosphere.

B21.2.1 Health Hazards

B21.2.1.1 Toxicity - Noble gases are nontoxic but may cause oxygen deficient atmospheres, i.e., simple asphyxiants (Reference 10).

B21.2.1.2 Threshold Limit Values-Time Weighted Average (TLV-TWA) - None established. (Regarding displacement of oxygen, oxygen at 16-17 percent by volume at 101 kPa (760 mm Hg) total pressure is the generally accepted minimum for a work environment.)

B21.2.1.3 Special Medical Information - For first aid information, see CPIA 394, Appendix E21.

B21.2.2 Fire and Combustion Product Hazards

The noble gases are inert. They are nonflammable.

B21.2.3 Explosion Hazards

Noble gases will not explode and may even be used to dilute an explosive atmosphere. If exposed to external heating, cylinders containing noble gases may pressure rupture.

B21.2.4 Environmental Effects

The noble gases are inert. Therefore, there are no environmental effects to be considered.

B21.3 RESERVED

B21.4 RESERVED

B21.5 MATERIALS AND EQUIPMENT COMPATIBILITY

The installation requirements for pressurized-gas systems are very stringent. Particularly important are the selection of materials and equipment, the use of adequate fabrication procedures and proper maintenance. Personnel assigned to operate these systems must be qualified and familiar with the equipment and proper operating procedures.

B21.5.1 Materials

The prime consideration in selecting materials to be used with high-pressure gases is their strength. Reactivity or corrosivity should not be problems with propellant grades of noble gases, but are a consideration in the maintenance of purity and cleanliness with regard to the end use.

B21.5.1.1 Metals - The following metals are suitable for use with noble gases:

Austenitic chrome-nickel steels
stainless steel, series 18-8
copper
brass
bronze
copper silicon alloys
Monel
aluminum
shredded lead
carbon steel

B21.5.1.2 Non-Metals - The following non-metals are suitable for use with noble gases:

synthetic and natural rubbers
Teflon
Kel-F
DC-55
Fluorolube (LG-160)

B21.5.1.3 Lubricants - Because noble gases are often used to pressurize oxidizer systems, the use of petroleum-base lubricants with this gas is prohibited. Lubricants such as the fluorolubes or the perfluorocarbons are recommended. The fluorolubes should not be used with aluminum.

B21.5.2 Equipment

B21.5.2.1 Containers - Storage containers should comply with one of the following:

a. designed, constructed, and tested in accordance with appropriate requirements of the ASME Boiler and Pressure Vessel Code, Section VIII, Unfired Pressure Vessels (Reference 11)

b. designed, constructed, tested, and maintained in accordance with DOT regulations and specifications

B21.5.2.2 Pumps and Hoses - Flexible hose generally conforms to military or industrial specifications. Unless otherwise specified in the governing code, a minimum safety factor of 4 to burst is required Flexible hose should only be employed when the use of rigid tubing is impractical. The hose should be of minimum length. When long runs are unavoidable, hose reels and other protective devices, including hold-down weights and chain ties at end connections, should be incorporated.

Pumps for high-pressure service should be designed, fabricated and installed in accordance with pertinent standards and codes.

B21.5.2.3 Lights - Any lighting which meets local electrical codes will be adequate in areas where noble gases are stored or transferred (Reference 12).

B21.5.2.4 Pipes and Fittings - Either pipe or tubing may be used for high-pressure service. Connections for tubing may be flange, thread, socket weld or butt weld. Tubing connections will be made with standard AN or industrial flared fittings of like materials. Stainless steel fittings should be used with stainless steel tubing to prevent galvanic action of dissimilar metals. Welded or brazed joints are preferred, since with flanged or threaded joints there is a greater possibility of leakage. The fabrication of pipe or tube anchors, hangers, supports and tie-downs should be in accordance with the American Society of Mechanical Engineers requirements for pressure piping (Reference 13).

The fabrication techniques and selection of materials for the installation and testing of piping systems should be in accordance with the ASME codes. High-pressure systems should be properly marked and identified.

B21.5.2.5 Gaskets - In high-pressure systems, "O" rings may be used. Gaskets may be flange, serrated, laminated, corrugated, or ring-joint types. The materials listed in Sections B21.5.1.1 and B21.5.1.2 may be used in the fabrication of gaskets and seals.

B21.5.2.6 Pressure Gauges - Standard-type pressure gauges incorporating compatible materials will be used in gaseous helium systems. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales. In addition, all gauges should have plastic faces and blow-out discs.

B21.5.2.7 Valves - Wherever practicable, all gate and globe valves should be of the outside-screw, rising-stem type. See Reference 19.

Plug and valves may be used instead of gate or globe types. Plug valves for high-temperature service should be designed to prevent galling by making the plug and the valve body of different materials, by treating the plug to give it different physical properties, by welding hard-face overlays to the surface, or by using special mechanical designs. Every valve should be plainly marked to identify the gas and valve function.

B21.5.2.8 Venting Systems and Pressure Relief - When the maximum allowable inlet pressure to one or more of the pressure-reducing devices is greater than the piping system's maximum allowable operating pressure, one or more pressure-relief or other safety devices should be included, e.g., automatic shut-off valves. Suitable protective measures should be taken to prevent injury or damage resulting from the discharge of gases from those safety devices.

The relief valves should have a combined discharge capacity ensuring that the pressure rating of the lower pressure piping system will not be exceeded as the result of an equipment failure, for instance, the pressure-reducing valves. The relief or safety valve should be located next to the pressure-reducing valve. Pressure-reducing and relief devices should conform to the valve requirements for the specified service conditions.

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PROPERTY	ENGLISH UNITS	SI UNITS
Boiling Point	-452°F	4.224 K
Freezing Point	He freezes at 1.0 atm pressure and has no triple point.	
Density (Gas)	0.078 lb/ft^3 at -452°F	16.89 kg/m^3 at 4.224 K
(1 atm)	0.011 lb/ft^3 at 32°F	0.176 kg/m^3 at 273.15 K and 101 kPa
(Liquid)	7.798 lb/ft^3 at -452°F	0.125 Mg/m^3 at 4.224 K
Specific Gravity - Vapor (relative to STP air)	0.137	0.137
Critical Density	4.35 lb/ft^3	69.64 kg/m^3
Critical Pressure	33.2 psia	229 kPa
Critical Temperature	-450.3°F	5.2 K
Vapor Pressure	14.696 psia at -452°F	101 kPa at 4.224 K
Coefficient of Absolute Viscosity (1 atm)	0.1864 centipoise at 32°F, and 14.7 psia	1.864 x 10^{-4} Pa·s at 273.15 K and 101 kPa

Figure B21-1 Physical Properties of Helium (MW = 4.0026 g mol)

CHAPTER B22
THE ALCOHOLS

B22.1 PROPERTIES

B22.1.1 Identification

See Table B22-1. Their molecular weight are also listed in Table B22-1.

B22.1.2 General Appearance and Common Uses

Methyl, ethyl and isopropyl alcohols are clear, water-white, mobile liquids. Furfuryl alcohol is a clear, amber-colored, mobile liquid. All of the alcohols have characteristic odors. Alcohols are widely used in various industries for a variety of purposes.

B22.1.3 Physical and Chemical Properties

Major physical properties of the alcohols are provided in CPIA 394.

B22.1.3.1 Solubility - All of these alcohols are miscible with water in all proportions. They are soluble in most of the common organic solvents such as acetone, ether, carbon tetrachloride, benzene, kerosene and gasoline.

B22.1.3.2 Stability - The only alcohol showing a tendency toward instability is furfuryl alcohol. Furfuryl alcohol forms water insoluble products with darkening of the solution upon exposure to heat, atmospheric oxygen, acidic atmospheres, or upon standing for long periods of time.

B22.1.3.3 Reactivity - All of the alcohols are excellent solvents. In addition, furfuryl alcohol may undergo polymerization. They are combustible liquids and will react vigorously with strong mineral acids or strong organic acids. The alkyl alcohols are not hypergolic with nonfluorinated oxidizers. Furfuryl alcohol, however, is hypergolic with fuming itric acid.

B22.1.3.4 Environmental Fate - The alcohols are biodegradable when diluted in water.

B22.2 HAZARDS

B22.2.1 Health Hazards

B22.2.1.1 Toxicity - The alcohols used in propulsive units have several properties that contribute to toxic hazards:

a. They are irritant to sensitive tissues.

b. They are fat solvents.

c. When absorbed into the body, they depress the higher brain centers.

Table B22-1 Identification of the Alcohols

NAME (MW in g mol)	CHEMICAL NAME	FORMULA
Methyl Alcohol (32.04)	Methanol (Wood Alcohol)	CH_3OH
Ethyl Alcohol (46.07)	Ethanol (Grain Alcohol)	C_2H_5OH
Isopropyl Alcohol (60.11)	Isopropanol	C_3H_7OH
Furfuryl Alcohol (98.10	Furfurol	$C_9H_8O_2OH$

The greatest danger from alcohols is the likelihood of their consumption as a beverage by uninformed persons. The occasional consumption of diluted but otherwise pure ethyl alcohol will not cause other than temporary intoxication, but because propellant alcohol is denatured with poisonous substances such as methyl alcohol, benzene (drying agent) and other denaturants, its consumption must be prohibited.

The defatting action of the alcohol is probably responsible, at least in part, for their ability to produce a mild irritation of the skin if allowed to remain on the skin or clothing. In either liquid or vapor form, these alcohols are irritating to the eyes, the mucous membranes and the lungs.

B22.2.1.1.1 Ethyl Alcohol - Ethyl alcohol in liquid form can irritate the eyes and, to a lesser degree, the skin. In high vapor concentrations the eyes and respiratory passages may be irritated (References 1, 2, and 3). If swallowed, the well-known alcoholic intoxication ensues. This is the result of a depression of the higher brain centers, erroneously interpreted by many as a stimulant effect. The consumption of propellant alcohol must be prohibited because of the poisonous compounds used to denature it.

B22.2.1.1.2 Furfuryl Alcohol - Like the other alcohols, furfuryl alcohol has a depressant action on the central nervous system. It appears to be more irritant to the respiratory tract than the other alcohols. Small doses stimulate respiration. Larger doses depress respiration, reduce body temperature, produce nausea, salivation, diarrhea, dizziness and diureses.

B22.2.1.1.3 Isopropyl Alcohol - Isopropyl alcohol has greater narcotic action than ethyl alcohol. Ingestion of 10 milliliters or more, or inhalation of large quantities of the vapor may progressively cause flushing, headache, dizziness, mental depression, nausea, vomiting, narcosis, anesthesia, coma and death.

B22.2.1.1.4 Methyl Alcohol - Methyl alcohol is so volatile that toxic concentrations of vapor can readily build up in enclosed spaces. Moderate exposure to the vapor causes irritation of the eyes, nose, throat, and lungs. Severe exposures cause stupor, dizziness, depression of the central nervous system, and gastrointestinal symptoms.

Consumption of methyl alcohol is dangerous. In addition to excruciating abdominal pain with nausea and vomiting progressing to convulsive spasms, some people may suffer blindness after drinking as little as 10 milliliters.

Methyl alcohol has a specific effect on the optic nerve. Optic nerve damage from inhalation exposure to methyl alcohol is not encountered in concentrations which are bearable (References 1, 4 and 5). Individual susceptibility varies. Severe poisoning may cause a coma lasting for several days terminated by death.

B22.2.1.2 Exposure Limits

B22.2.1.2.1 Threshold Limit Values-Time Weighted Average (TLV-TWA) (See References 6, 7 and 8)

Ethyl Alcohol: The TLV-TWA is 1000 ppm (1900 mg/m^3)
(1000 ppm is about the odor threshold)

Furfuryl Alcohol: The TLV-TWA is 10 ppm (40 mg/m^3)
Skin warning. Skin exposure can contribute to inhalation type health effects. The TLV-STEL is 15 ppm (60 mg/m^3) and the IDLH is 250 ppm (1 g/m^3)

Isopropyl Alcohol: The TLV-TWA is 400 ppm (980 mg/m^3)
The TLV-STEL is 500 ppm (1225 mg/m^3) and the IDLH is 20,000 ppm (49 g/m^3)

Methyl Alcohol: The TLV-TWA is 200 ppm (260 mg/m^3)
Skin warning. Skin exposure can contribute to inhalation type health effects. The TLV-STEL is 250 ppm (310 mg/m^3) and the IDLH is 25,000 ppm (31 g/m^3)

B22.2.1.3 Special Medical Information - For first aid information, see CPIA 394, Appendix E22.

B22.2.2 Fire and Combustion Product Hazards

Combustion of the alcohols differs from the combustion of the hydrocarbons in that the alcohol flames are difficult to see in daylight: there are no luminous carbon particles (smoke) to make them visible. For this reason, care must be taken in combatting open-spill fires or fires involving spills in equipment, since fire back-flashes may occur and be unperceived by the fire fighting personnel. Such an occurrence could result in injury. The alcohols are moderately flammable and water soluble. Methyl and ethyl alcohols will support combustion, once initiated, even though diluted to less than 50% concentration in water.

B22.2.3 Explosion Hazards

Vapors of methyl, ethyl and isopropyl alcohol readily form explosive mixtures with air. The vapor pressure of furfuryl alcohol is lower than those of methyl, ethyl or isopropyl alcohol, it is less hazardous under normal conditions.

B22.3 RESERVED

B22.4 RESERVED

B22.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B22.5.1 Materials (Reference 9)

Contact with oxidising materials may cause a vigorous reaction for each of these alcohols.

Methyl alcohol becomes unstable with the addition of heat and may react with metallic aluminum at high temperatures. Contact with strong oxidizers may cause fires and explosions.

B22.5.1.1 Metals - Steel is very commonly used for the construction of drums, containers, main storage tanks, and permanent storage facilities for alcohol, although stainless steel, high-tensile steel, Monel metal, aluminum and aluminum alloys are also used.

Steels authorized for use with the alcohols include mild steel (hot-rolled, cold-rolled), low-carbon steel, open-hearth steel, and electric steel. Standard commercial quality is acceptable.

Several aluminum alloys are also authorized for use with the alcohols discussed herein.

B22.5.1.2 Non-Metals - Materials acceptable for alcohol service are:

Tetrafluoroethylene (Teflon TFE), Halon TFE, or equivalent
Chlorotrifluoroethylene (Kel-F), Halon CTF, or equivalent
Polyethylene
Polyvinyl Chloride
Fluorosilicone (FSI)
Neoprene, Chloroprene
Ethylene Propylene (PM, EDPM, EPT)
Silicone (SI)
Styrene Butadiene (SBR, BUNAS, GR)

Neoprene and rubber may be used with all alcohols except furfuryl alcohol.

B22.5.1.3 Lubricants - Because all of the alcohols are excellent solvents, specialized lubricants such as fluorinated hydrocarbons, molybdenum disulfide and graphite-based lubricants must be used wherever the alcohol can come in contact with the lubricant. Thread lubricants and sealants include:

Permatex No. 2
Litharge and Glycerin
Perfluoroethylene Tape

B22.5.2 Equipment

B22.5.2.1 Containers - The alcohols are shipped in drums, tank trucks, and tank cars, and may be stored in drums or main storage tanks. See Section B22.5.1.1 for material data.

B22.5.2.2 Pumps and Hose - Flooded-suction, centrifugal pumps may be used with alcohol storage tanks designed with bottom outlets. Centrifugal pumps equipped with a liquid reservoir to make them self-priming may be used with alcohol storage tanks not equipped with bottom outlets. If limited space will not accommodate a self-priming centrifugal pump, a positive-displacement pump is recommended.

Hoses used in alcohol service may be constructed of materials as listed in Section B22.5.1.2.

B22.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Bureau of Mines. Flash lights or storage battery lamps, where permitted for use, should be of the safety types approved by the Bureau of Mines, or as accepted by the current edition of the National Electrical Code (Reference 10).

B22.5.2.4 Pipes and Fittings - Pipes and fittings must be of approved materials (See Section B22.5.1), and must be tested for a design working pressure of at least 1.034 MPa (150 psig). Welded or flanged connections are recommended; threaded connections with the proper thread sealing compound (See Section B22.5.1.3) are permissible.

B22.5.2.5 Gaskets - Gaskets may be made of the material listed in Section B22.5.1.2.

B22.5.2.6 Pressure Gauges - Standard-pressure gauges will be used in alcohol service and storage systems. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B22.5.2.7 Valves - It is the practice of industry to use gate or globe type valves; however, plug and ball type valves are also acceptable.

B22.5.2.8 Venting Systems and Pressure Relief - Vent openings in the storage system should terminate outdoors and should be protected by approved flame arresters. Vents should be sized according to the specifications given in Reference 12.

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CHAPTER B23
HALOCARBONS (FREONS)

B23.1 PROPERTIES

B23.1.1 Identification

A list of the halocarbons, divided into three sections, is presented in the tables below. Table B23-1 lists the Halon fire extinguishing agents (Reference 1). Table B23-2 lists the Freon refrigerants. Table B23-3 lists halocarbon solvents (References 2, 3). Molecular weights of all halocarbons are included in these tables.

B23.1.2 General Appearance and Common Uses

The halocarbons discussed in this chapter are used as fire extinguishing agents, aerosol propellants, refrigerants, and solvents.

Most Freon halocarbons appear as colorless, almost odorless liquids and vapors.

The solvents, carbon tetrachloride, perchloroethylene, trichloroethane, and chloroform are colorless liquids with ethereal odor. Methylene chloride is a colorless, volatile liquid.

B23.1.3 Physical and Chemical Properties (References 1, 2, 3 and 4)

The specific chemical properties of all the halocarbons are provided in CPIA 394. Figures B23-1 through B23-5 list the properties of the Freon refrigerants.

B23.1.3.1 Environmental Fate (References 5 and 6) - Halocarbons in the atmosphere apparently have no tropospheric sink and diffuse into the stratosphere where they are disassociated by short wave radiation to yield atomic chlorine. The atomic chlorine then reacts with ozone and cataiyzes its decomposition. The reactivity of the halocarbons determines its susceptibility to photo-oxidation. Experiments by Cox, et. al., (Reference 5) have shown that some of the halocarbons are susceptible to photo-oxidation, but the fully halogenated compounds such as Freon 11 and 12 and carbon tetrachloride are for all practical purposes nonreactive toward the hydroxyl radical and thus resistant to photo-oxidation. The tropospheric lifetimes for some of the halocarbons are give below (Reference 5):

HALOCARBON	LIFETIME, YR.
Chloroform	0.19
Methylene Chloride	0.3
Trichloroethylene	1.1
Freon 12	330
Carbon Tetrachloride	< 330
Freon 11	> 1000

Table B23-1 Halon Fire Extinguishing Agents

SHORT TITLE	MIL SPECS	CHEMICAL NAME	CHEMICAL FORMULA (MW in g mol)
Halon 1202	MIL-D-4540	Dibromodifluoromethane	CF_2Br_2 (209.83)
Halon 1211	MIL-B-38741	Bromochlorodifluoromethane	CF_2ClBr (165.37)
Halon 1301	MIL-M-12218	Bromotrifluoromethane	CF_3Br (148.92)
Halon 2402	No Spec	Dibromotetrafluoroethane	$C_2F_4Br_2$ (259.83)

Table B23-2 Halocarbon Refrigerants

SHORT TITLE	MIL SPECS	CHEMICAL NAME	CHEMICAL FORMULA (MW in g mol)
Halon 1130 (Freon 11)	BB-F-1421	Trichlorofluoromethane	$CFCl_3$ (137.37)
Halon 1220 (Freon 12)	BB-F-1421	Dichlorodifluoromethane	CF_2Cl_2 (120.91)
Halon 1120 (Freon 21)	BB-F-1421	Dichloromonofluoromethane	$HCFCl_2$ (102.92)
Halon 1210 (Freon 22)	BB-F-1421	Difluorochloromethane	HCF_2Cl (86.47)
Halon 2330 (Freon 113)	as refrigerant BB-F-1421 as solvent MIL-C-81302	Trichlorotrifluoroethane	$C_2F_3Cl_3$ (187.38)

Table B23-3 Halocarbon Solvents

SHORT TITLE	MIL SPECS	CHEMICAL NAME	CHEMICAL FORMULA (MW in g mol)
Carbon Tetrachloride	No Spec.	Tetrachloromethane	CCl_4 (153.82)
Chloroform	No Spec.	Trichloromethane	$CHCl_3$ (119.38)
Methylene Chloride	MIL-D-6998	Dichloromethane	CH_2Cl_2 (84.93)
Perchloro-ethylene	O-T-236	Tetrachloroethylene	CCl_2CCl_2 (165.83)
Trichloro-ethylene	O-T-634		CCl_2CClH (131.29)
Trichloro-ethane	O-T-620 or MIL-T-81533		$CH_2ClCHCl_2$ (133.41)

Halocarbons can find their way into water systems by atmospheric fallout, spills or direct discharge. Since these compounds are for the most part hydrophobic and of limited solubility, they are absorbed onto bottom sediments, clays, biomass and other organic particulates.

B23.1.4 Solubility

Halocarbons are generally slightly soluble or insoluble in water. Many are miscible with alcohols, benzene, chloroform, ether and carbon disulfide.

B23.1.5 Stability

Many of the halocarbons emit dangerous gases when decomposed at high temperature (e.g., carbon tetrachloride emits phosgene when heated to decomposition). Otherwise, under normal conditions, these chemicals are considered stable.

B23.1.6 Reactivity

Methylene Chloride reacts violently with Li, NaK, potassium tert-butoxide.

Chloroform reacts violently with acetone and a base Al, disilane, Li, Mg, N_2O_4, K, (perchloric acid and phosphorous pentoxide), (KOH and methanol), K-tert-butoxide, Na, (NaOH and methanol), sodium methylate, NaK.

Freon 11 reacts violently with Ba.

Perchloroethylene reacts violently with Ba, Be, Li.

Trichloroethylene reacts violently with Al, Va, N_2O_4, Li, Mg, KOH, KNO_3, Na, NaOH, Ti.

Trichloroethane reacts violently with acetone, N_2O_4, O_2, Na, NaOH, Na-K alloys.

B23.2 Hazards

B23.2.1 Health Hazards

B23.2.1.1 Toxicity - Most of the halocarbons used as refrigerants, fire extinguishing agents, and solvents are toxic in some degree. Since they are generally present in sealed systems, those in the first two categories present a lower risk than the chlorinated solvents. All are readily absorbed into the blood stream. Refrigerant gases may cause respiratory irritation, dizziness, unconsciousness and death. Heat arrhythmias may also be produced. The fire extinguishing agent (Halons) can be lethal if the concentration in air is above 10 percent. For safety consideration extinguisher vapor concentration should be limited to a maximum of 7 percent. The halocarbon solvents cause various degrees of liver damage; in this respect carbon tetrachloride is extremely hazardous and should never be used for cleaning purposes or for fire extinguishment. In a fire, the chlorinated solvents and refrigerants may produce toxic products. The defatting action of halocarbons on the skin may cause dry skin, dermatitis, blisters and local infection. The refrigerants may cause frostbite. Specific comments are provided below for individual chemicals (References 7, 10, and 11):

Halon 1301 - Halon 1301 vapors are narcotic when in high concentrations. When this material decomposes, toxic gases such as hydrogen bromide, carbonyl fluoride, hydrogen fluoride, and carbon monoxide may be released. (Also see Special Medical Information, Section B23.2.1.3)

Freon 11 - Freon 11's vapors are narcotic and at high concentrations may cause asphyxia. Upon decomposition, it emits toxic gases and vapors such as hydrogen chloride, phosgene, and hydrogen fluoride. (Also see Special Medical Information)

Freon 12 - (See Special Medical Information)

Freon 21 - Freon 21 is a respiratory irritant and at very high concentrations causes asphyxia. Upon decomposition, toxic gases such as hydrogen chloride, phosgene, and hydrogen fluoride may be released. (Also see Special Medical Information)

Freon 113 - (See Special Medical Information)

Carbon Tetrachloride - Vapors of carbon tetrachloride are narcotic. They can cause severe damage to the liver and kidneys. In humans the majority of fatalities due to carbon tetrachloride have resulted from renal injury with secondary cardiac failures. Liver failure is more frequently associated with ingestion than inhalation of the vapors. Upon decomposition, toxic gases and vapors such as hydrogen chloride, chlorine, phosgene, and carbon monoxide may be released (Also see Special Medical Information). This material should be used only where substitute solvent can not be found.

Chloroform - The vapors of chloroform act as a depressant and are toxic to the liver and kidneys. The liquid has a defatting effect on the skin and may cause chronic irritation with drying and cracking. It causes an immediate burning pain and conjunctional irritation when splashed in the eyes. Upon decomposition, chloroform may release toxic gases and vapors such as hydrogen chloride, chlorine, phosgene, and carbon monoxide. It slowly reacts to form phosgene and hydrogen chloride gases when in the presence of air. (Also see Special Medical Information)

Methylene Chloride - Methylene chloride is painful and irritating when splashed in the eyes or if confined on the skin by gloves, etc. A burn can result from such skin or eye contact. Methylene chloride has produced slight narcosis in animals. Extreme exposure can be fatal. Short term exposures can result in headaches, giddiness, stupor, irritability, numbness, and tingling in limbs, fatigue, irritation of the eyes and respiratory passage, neurasthenic disorders. Long term exposures may damage the liver and kidneys. Long term exposures may cause toxic encephatoses with acoustical and optical delusions and hallucinations. Toxic effects of the methylene chloride are due in part to the body's capacity to metabolize methylene chloride to CO. Carbon monoxide and carboxyhemoglobin in the blood are produced in humans receiving exposures. These effects have been found to be augmented in an additive way to the effects of CO in the air.

PCE (Perchloroethylene) - (See Special Medical Information)

1,1,2-Trichloroethane - This material has a potent narcotic vapor and considered to be carcinogenic. It may injure the lungs, liver, and kidneys and may irritate the nose and eyes. No cases of intoxication or systemic effects have been reported for humans from industrial exposures. Upon decomposition, in a fire, toxic gases and vapors such as hydrogen chloride, phosgene, and carbon monoxide may be released. (Also see Special Medical Information)

B23.2.1.2 Exposure Limits - Exposure limit data for the halocarbons is summarized in CPIA 394 (References 7, 8, 9 and 10).

B23.2.1.3 Special Medical Information (References 10, 11 and 12) - For first aid information, see CPIA 394, Appendix E23. Special medical information for the individual halocarbons is given below:

Halon 1301 - Halon 1301 can affect the body by inhalation or contact with the skin or eyes. When breathed in large concentrations, it may cause light headedness or may cause the heart to beat irregularly or stop (cardiac arrhythmia). It is not a known eye irritant and does not have other adequate warning properties.

Freon 11 - Freon 11 can affect the body by inhalation, by contact with the skin or eyes, or by ingestion. It may cause drowsiness, unconsciousness, and death. Breathing high concentrations may cause the heart to beat irregularly or stop. Prolonged overexposure may cause skin irritation. Other symptoms include incoordination, tremors, and frostbite (upon exposure to cold material). It is considered to have poor warning properties.

Freon 12 - Freon 12 can affect the body by inhalation or by contact with the skin or eyes. It can affect the cardiovascular system and peripheral nervous system, the symptoms including dizziness, tremors, unconsciousness, cardiac arrhythmias, and cardiac arrest.

Freon 21 - Freon 21 can affect the body by inhalation, contact with skin or eyes, or ingestion. The effects of long term exposure are not known, but short term exposures may cause drowsiness, unconsciousness, and death. It may cause the heart to beat irregularly or stop and can affect the lungs and respiratory system by asphyxia. Skin contact with the liquid may cause frostbite (Reference 13).

Freon 113 - Freon 113 can affect the body by inhalation, contact with skin or eyes, and ingestion. The skin and heart can be affected, symptoms including irritated throat, drowsiness, and dermatitis.

Carbon Tetrachloride - This material can affect the body if inhaled or by skin or eye contact with the liquid. Long term exposure may cause liver and kidney damage. Long term contact of the liquid with the skin may cause skin irritation. OSHA/NIOSH considers this material to be carcinogenic. Short term exposures may cause drowsiness, dizziness, incoordination and unconsciousness. Delayed effects include heart, liver and kidney damage. Liver damage has symptoms including yellow jaundice and dark urine. Eye contact with the liquid causes burning and intense irritation (Reference 13).

Chloroform - This material can affect the body if it is inhaled, contacts the skin or eyes, or is ingested. Long term exposure may cause liver or kidney damage. Long term, repeated skin contact with the liquid may cause skin irritation. Short term exposure to chloroform vapors may cause headache, drowsiness, vomiting, dizziness, unconsciousness, mental dullness, irregular heart beat and death. Liver and kidney damage also may result. Chloroform splashed in the eyes causes pain and irritation. Swallowing chloroform results in immediate severe burning of the mouth and throat with abdominal and chest pain and vomiting. Loss of consciousness and liver damage may also result from ingestion of the liquid.

Methylene Chloride - This material can affect the body by inhalation, contact with skin or eyes and ingestion. Symptoms include fatigue, weakness, sleepiness, lightheadedness, numbness or tingle in limbs, nausea, eye irritation, skin irritation, vertigo and worsen angina.

PCE (Perchloroethylene) - This material can affect the body by inhalation, contact with the skin or eyes and ingestion. It can effect the central nervous system, mucous membranes, eyes, lungs, liver, kidney, heart and skin. Symptoms include irritated/burning eyes, nose and throat, congestion, nausea, flushed face and neck, vertigo, dizziness, unconsciousness, light narcosis, difficulty in motor coordinating, incoordination, headache, erythemia and liver damage. It is a possible carcinogen. When given as a hookworm anthelmintic, it produced narcotic effects, exhilaration, and inebriation.

1,1,2-Trichloroethane - This material can affect the body by inhalation, absorption through the skin, skin or eye contact, and ingestion. Exposures may also cause kidney or liver damage, eye irritation, nose irritation, drowsiness, depression, incoordination, unconsciousness, and death.

B23.2.2 Fire and Combustion Product Hazards

Upon decomposition, toxic gases and vapors are produced by the halocarbons. The specific toxic products for the individual chemicals are delineated under Toxicity (Section B23.2.1.1). The Freons, carbon tetrachloride, and chloroform, are not combustible. Methylene chloride is not combustible based on standard tests but will burn under extreme conditions. Trichloroethane is flammable and has flammability limits in air of 6 and 15.5 percent by volume.

B23.2.3 Explosion Hazards

Generally no explosion hazards were cited for these materials. Incompatibilities that could possibly lead to fire or explosion are listed below:

Halon 1301 - Reacts with chemically active metals, calcium, or with powdered aluminum, zinc, and magnesium (Reference 13).

Freon 11, 12, and 21 - Reacts with chemically active metals such as sodium, potassium, calcium, powdered aluminum, zinc, and magnesium (Reference 13).

Freon 113 - Reacts with chemically active metals, calcium, powdered aluminum, zinc, magnesium, and beryllium. It decomposes in contact with alloys containing greater than 2 percent magnesium.

Carbon Tetrachloride - This material will react with chemically active metals such as sodium, potassium, and magnesium.

Chloroform - Chloroform reacts with strong caustics and chemically active metals such as aluminum, magnesium powder, sodium, and potassium.

Methylene Chloride - Methylene chloride has flammability limits between 12 and 19 percent. This material will react with strong oxidizers, strong caustics, chemically active metals such as aluminum, magnesium powders, sodium, and potassium.

PCE (Perchloroethylene) - PCE reacts with strong oxidizers, chemically active metals such as barium, lithium, and beryllium.

Trichloroethane - This material reacts on contact with strong oxidizers, strong caustics, and chemically active metals such as aluminum and magnesium powders, sodium, and potassium. Contact may result in fire or explosion.

B23.2.4 Environmental Effects

The persistent nature of the fully halogenated methane and the lack of tropospheric sinks indicate that these compounds will terminate in the stratosphere. Perchloroethylene and trichloroethylene are reactive in the troposphere and their release can lead to the formation of phosgene which may have an adverse effect. The release of atomic chlorine in the stratosphere and its subsequent reaction with ozone could have deleterious effects if significant quantities over time are released. Most of these compounds are insoluble in water or slightly miscible. Their major transportation mode in the environment is by air. In aqueous systems, however, the halocarbons will absorb onto sediments and persist. Since these compounds are hydrophobic, one would expect that their transport through land into groundwater reservoirs would be limited. However, recent studies have demonstrated alarming concentrations of halocarbons, including trichloroethylene, tetrachloroethylene, carbon tetrachloride, 1,1,1-trichloroethane, and methylene chlorine in ground water drinking water supplies. Many of these compounds are suspected carcinogens, and possible health risks associated with consumption of water with high concentrations of these have prompted the EPA to require treatment of such water supplies.

B23.3 RESERVED

B23.4 RESERVED

B23.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B23.5.1 Materials

The halogenated hydrocarbons may be divided into several subgroups: the Halons, Freons, and the solvents.

Halons: halon 1301, 1211, 2401 are classified as a liquified compressed gas, for transfer storage purposes. Cylinders of approved design, construction and properly selected materials in accordance with 49 CFR 170-190 (Reference 16) should be used. Excessive heat may contribute to instability causing containers to burst. Toxic gases and vapors (such as hydrogen bromide, carbonyl fluoride, hydrogen fluoride, and carbon monoxide) may be released when trifluoromonobromomethane decomposes. Halons react with chemically active metals, calcium, or with powdered aluminum, zinc, and magnesium. A special precaution is the possibility of liquid halons attacking some forms of plastics, rubber and coatings.

Freon 11, 21, 22, 113 are all stable liquids or gases under normal temperature and pressure conditions. These substances, however, become unstable when heated. In general, Freons react with chemically active metals such as sodium, potassium, calcium, powdered aluminum, zinc, and magnesium. Liquid Freons, also in general, attack some forms of plastics, rubber, and coatings.

The solvents include carbon tetrachloride, chloroform, methylene chloride (MC), perchloroethylene (PCE), trichloroethylene (TCE), and trichloroethane.

Carbon tetrachloride is not combustible but does react with chemically active metals such a sodium, potassium, and magnesium.

Chloroform (or trichloromethane) is slightly flammable when exposed to high heat but otherwise practically nonflammable. Chloroform reacts energetically with strong caustics and, like carbon tetrachloride, chemically active metals.

Methylene chloride is practically nonflammable, normally stable, but unstable at elevated temperature and pressure.

All of these solvents will attack some forms of plastics, rubber, and coatings.

The selection of materials must take into account the compatibility of the pure solvent with the metals or nonmetals. In addition, the selection of materials must consider the overall compatibility of the solvent and contaminants acquired during cleaning process.

The presence of water or minute quantities of hydrochloric acid in the solvents will have a corrosive action on most metals. Continuous exposure of zinc, aluminum, magnesium, copper, and copper alloys to solvents containing excessive moisture under elevated temperatures should not be allowed. These metals, however, can be cleaned in the solvent. The following metals are suitable for solvent service.

a. When moisture content or acidity is equal to or less than specified in the solvent Military Specification.

   Carbon steels

b. Under wet conditions

   Stainless steels (300 series)
   Nickel
   Monel
   Inconel

B23.5.1.2 Non-Metals - The effects of solvents on nonmetal (plastics or elastomers) are difficult to predict or to make any generalized statement. As an example, the effect of a solvent on elastomers will depend on the plasticizers, curing conditions, nature of the polymer and other variables. Synthetic rubbers produced by some manufacturers may contain plasticizers which can be extracted by the solvent. Therefore, tests should be conducted to determine overall material suitability to the application involved. The effect of solvents on selected plastics and elastomers are given in CPIA 394. The following non-metals are recommended for solvent handling equipment:

a. Pump Packings; Graphited asbestos

b. Compressor Packing; Square plaited graphite asbestos

c. Mechanical Shaft Seals; Tetrafluoroethylene polymer, high density

d. Gaskets; Compressed asbestos, paper stock with phenolformaldehyde resin binder (for service below 366 K [200°F]), paper fiber and cork impregnated with solvent insoluble varnish or dope, lead gaskets, and fluorocarbon polymer reinforced with metal mesh.

e. Lubricants; Solvent-insoluble lubricants such as calcium soap or soap of a stearate and fluorolubes.

B23.5.2 Equipment

B23.5.2.1 Tanks - Storage tanks must be liquid and vapor tight. Either horizontal or vertical tanks are suitable and should bo so constructed to facilitate removal of sludge by flushing tank bottoms. Underground storage tanks are not recommended. Warm storage locations increase solvent evaporation losses and should be avoided.

B23.5.2.2 Pipings, Fittings and Valves - Piping can be welded, flanged or threaded. Standard weight piping is satisfactory. Fittings and valves can be of steel, wrought iron or cast iron. Gate or iron plug-cock valves will give good service. Piping should be clearly identified (Reference 17).

B23.5.2.3 [illegible] - Whenever possible, a centrifugal pump should be used. A self-priming positive displacement pump or gravity transfer should be used if the storage system is located below the pump.

B23.5.2.4 Vents - All storage tanks must be provided with vents so air can enter or leave the tank as solvent is added or withdrawn. A vent dryer containing silica gel is recommended to remove moisture present in the air.

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		75°F	297 K
Freezing Point		-168°F	162 K
Density (Saturated vapor at b.p.)		0.366 lb/ft³	5.86 kg/m³
Liquid		12.24 lb/gal (91.58 lb/ft³)	1.487 Mg/m³
Specific Gravity - Vapor (relative to STP air)		4.72	4.72
Critical Density		34.6 lb/ft³	0.554 Mg/m³
Critical Pressure		639.5 psia	4.41 MPa
Critical Temperature		388.4°F	471 K
Vapor Pressure		13.3 psia	92 kPa at 93 K
Coefficient of Viscosity	Kinematic		
	Absolute (Gas)	0.0122 centipoises at 68°F	1.22×10^{-5} Pa·s at 293 K
	(Liquid)	1.4 centipoises at 68°F	1.4×10^{-3} Pa·s at 293 K

Figure B23-1 Physical Properties of Halon 1130 (Freon 11)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-22°F	243 K
Freezing Point		-252°F	115.4 K
Density (Saturated vapor at b.p.)		0.395 lb/ft^3	6.33 kg/m^3
(Liquid)		10.94 lb/gal (81.84 lb/ft^3)	1.311 Mg/m
Specific Gravity - Vapor (relative to STP air)		4.17	4.17
Critical Density		34.84 lb/ft^3	0.558 Mg/m^3
Critical Pressure		596.9 psia	4.116 MPa
Critical Temperature		233.6°F	385 K
Vapor Pressure		83.8 psia	578 kPa
		at 68°F	at 293 K
Coefficient of Viscosity	Kinematic		
	Absolute (Gas)	0.0122 centipoises at 68°F	1.22 x 10^{-6} Pa·s at 293 K
	(Liquid)	1.4 centipoises at 68°F	1.4 x 10^{-3} Pa·s at 293 K

Figure B23-2 Physical Properties of Halon 1220 (Freon 12)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		113 to 140°F	317.6 to 333.2 K
Freezing Point		-56 to 115°F	224.3 to 319.3 K
Density (Gas)		0.285 lb/ft^3	4.57 kg/m^3
(Liquid)		11.40 lb/gal (85.28 lb/ft^3)	1.366 Mg/m^3
Specific Gravity - Vapor (relative to STP air)		3.55	3.55
Critical Density		4.17 lb/gal (32.57 lb/ft^3)	0.522 Mg/m^3
Critical Pressure		750 psia	5.172 MPa
Critical Temperature		353.3°F	451.7 K
Vapor Pressure		7.7 x 10^{-2} psia at 68°F	0.533 kPa at 293 K
Coefficient of Viscosity	Kinematic		
	Absolute (Gas)	0.011 centipoises at 77°F	1.1 x 10^{-5} Pa·s at 298 K
	(Liquid)	0.34 centipoises at 77°F	3.4 x 10^{-4} Pa·s at 298 K
Autoignition Temperature			
Flash Point		36 - 39°F	275-277 K
Flammability Limits	Lower:	9.7 percent by volume in air	
	Upper:	12.8 percent by volume in air	

Figure B23-3 Physical Properties of Halon 1120 (Freon 21)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		-41°F	232.6 K
Freezing Point		-256°F	113.2 K
Density (Saturated vapor at b.p.)		0.295 lb/ft^3	4.72 kg/m^3
(Liquid)		9.66 lb/gal (74.5 lb/ft^3)	1.194 Mg/m^3
Specific Gravity - Vapor (relative to STP air)		3.0	3.0
Critical Density		4.38 lb/gal (32.79 lb/ft^3)	0.525 Mg/m^3
Critical Pressure		721.9 psia	4.98 MPa
Critical Temperature		204.8°F	369.2 K
Vapor Pressure			
Coefficient of Viscosity	Kinematic		
	Absolute (Gas)	0.0122 centipoise at 68°F	1.22 x 10^{-5} Pa·s at 298 K
	(Liquid)	1.4 centipoise at 68°F	1.4 x 10^{-3} Pa·s at 298 K

Figure B23-4 Physical Properties of Halon 1210 (Freon 22)

PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		118°F	320.9 K
Freezing Point		-31°F	238.2 K
Density (Saturated vapor at b.p.)		0.46 lb/ft^3	7.38 kg/m^3
(Liquid)		13.1 lb/gal (97.7 lb/ft^3)	1.585 Mg/m^3
Specific Gravity - Vapor (relative to STP air)		6.45	6.45
Critical Density		4.95 lb/gal (37.04 lb/ft^3)	0.593 Mg/m^3
Critical Pressure		498.9 psia	3.44 MPa
Critical Temperature		417.4°F	487.3 K
Vapor Pressure		5.49 psia at 68°F	37.9 kPa at 293 K
Coefficient of Viscosity	Kinematic		
	Absolute (Gas)	0.0122 centipoise at 68°F	1.22 x 10^{-6} Pa·s at 298 K (101 kPa)
	(Liquid)	1.4 centipoise at 68°F	1.4 x 10^{-3} Pa·s at 298 K

Figure B23-5 Physical Properties of Halon 2330 (Freon 112)

CHAPTER B24
HYDRAULIC FLUIDS

B24.1 PROPERTIES

B24.1.1 Identification

The hydraulic fluids discussed in this chapter are listed in Table B24-1, with the pertinent military specifications. These hydraulic fluids are comprised of a petroleum base and an anti-wear agent (triorthocresyl phosphate) in addition to other additives (References 1, 2, 3, 4, and 5).

Table B24-1 Hydraulic Fluids

DESIGNATION	MIL SPECIFICATION
C-635 (NATO Symbol)	MIL-H-6083D
H-515	MIL-H-5606E
H-537	MIL-H-83282B
H-573	None
Triorthocresyl phosphate	Chemical formula $(CH_3C_6H_4O)_3$

B24.1.2 General Appearance and Common Uses

The fluids discussed above are used as hydraulic fluids and heat exchange mediums. Triorthocresyl phosphate is a colorless, odorless liquid. The hydraulic fluids covered by military specifications generally contain dye for coloration, and the petroleum base gives them an oil appearance.

B24.1.3 Physical and Chemical Properties

The chemical and physical properties for the hydraulic fluids are provided in CPIA 394.

B24.1.3.1 Solubility - (Insufficient information other than to treat it as a petroleum base product.)

B24.1.3.2 Stability - Triorthocresyl phosphate is stable.

B24.1.3.3 Reactivity - Triorthocresyl phosphate will attack some forms of plastics, rubber, and coatings. (Insufficient information on other hydraulic fluids.)

B24.1.3.4 Environmental Fate - Hydraulic fluids would most likely find their way into the environment through spills. Phosphate esters can be hydrolyzed in water to phosphoric acids. These acids form complexes in surface and soil water with metals, of which the calcium salt is soluble in water. Phosphates occur naturally; however, the effect on biostimulation of the aquatic life has caused restrictions on the quantities of phosphates allowed to be released in natural water systems. Typical sources of phosphate in natural and soil water systems are from point discharges, particularly municipal sewage treatment facilities, and through fertilization. Phosphates are required by living systems for metabolism. Orthophosphate is generally considered the most bioavailable form.

B24.2 HAZARDS

B24.2.1 Health Hazards

B24.2.1.1 Toxicity - Triorthocresyl phosphate, a constituent compound in hydraulic fluid, is known to be toxic. Some hydraulic fluids contain polychlorobiphenyl and nitrosamine which are also considered toxic.

B24.2.2 Fire and Combustion Product Hazards

Petroleum-based hydraulic fluids are highly combustible. A typical fluid may have a flash point of 422 K (300°F) and an autoignition temperature of 533.2 K (500°F). A fine mist is produced when oil is discharged under hydraulic pressure, and these burn rapidly generating large quantities of heat. The Btu content of hydraulic oil is 37,630 MJ/m^3 (135,000 Btu/gal) or 41.9 MJ/kg (18,000 Btu/lb) (Reference 6).

B24.2.3 Explosion Hazards

There are no explosion hazards.

B24.2.4 Environmental Effects

Phosphate esters in soil and water systems would cause a stimulation of aquatic life, or a fertilization effect. The impact would be dependent on how easily the ester could be converted to a useable form by biota. Phosphates would be carried as complexes in surface waters. Some would be sorbed into sediments and taken up by aquatic life but the majority would be transported until a termination point such as the sea or a lake is reached. Phosphates in significant concentrations are directly responsible for lake eutrophication through biostimulation of aquatic life.

Synthetic hydraulic fluids are insoluble in water, and in surface water systems would be expected to collect in an upper phase. In surface water, the fluids would sorb on plant life and have harmful effects. Eventually, they may be slowly biodegraded.

B24.3 RESERVED

B24.4 RESERVED

B24.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B24.5.1 Materials

The prime consideration in choosing materials for use with hydraulic fluids is related to their solvent action on most organic matter. Corrosion associated with hydraulic fluids at ambient temperatures is negligible for most metals. Limitations which should be observed are outlined in this section.

B24.5.1.1 Metals - Common ferrous and non-ferrous alloys are suitable for the fabrication of containers (fixed or mobile drums and tanks), associated piping and fittings, pumping equipment, valves, and other metal parts. Long term storage of hydraulic fluids may require special consideration of compatible metal containers and associated equipment or parts.

Preferred metals for usage with hydraulic fluids include:

Aluminum alloys - 1000, 3000, 5000, 6000 series
Stainless steel - 300 series

B24.5.1.2 Non-Metals - Listed below are some of the recommended and prohibited non-metals:

RECOMMENDED	PROHIBITED
Cork, or paper gasket materials designed for this service	Acrylics
Buna N	Polyisobutylenes
Fluorocarbons (Teflon, Kel-F Halon, TFE)	Natural Rubber
Polyamides	
Polyethylene	
Neoprene	
Vinyls	

B24.5.1.3 Lubricants - Graphite-base, molybdenumdisulfide, some silicone and fluorocarbon lubricants may be used with some hydraulic fluids.

Since hydraulic fluids may be excellent solvents for most organic matter, petroleum lubricants are not recommended.

Recommended sealants with hydraulic fluids are:

MIL-S-8802
Babbit No. 2
Permatex 1 and 2
X-Pando
Seal-Rite No. 5
Q-Seal, Teflon

B24.5.2 Equipment

B24.5.2.1 Containers - Hydraulic fluids are drummed or stored in tanks of approved design and construction which may be either permanent or transportable. See DOT regulations for container requirements. Portable containers should be gas-right (See Venting, Section B24.5.2.7).

B24.5.2.2 Pumps - Permanently installed pumps in main storage systems may also be equipped with a liquid reservoir to serve as a primer for the pump used to empty tank cars, trucks, and drums not equipped with bottom outlets.

B24.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Mine Safety and Health Administration (MSHA).

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the MSHA or as accepted by the current edition of the National Electrical Code (Reference 7).

B24.5.2.4 Pipes and Fittings - Pipes and fittings for liquid hydraulic fluid should be made of approved materials (See Section B24.5.1), and should be hydrostatically tested. Threaded connections with a recommended sealing compound are permitted for this service; flanged or welded connections are preferred. Top loading lines should extend into the bottom of the tank.

B24.5.2.5 Hoses and Gaskets - Hoses used in liquid hydraulic fluid service may be constructed of materials as listed in Section B24.5.1.2. Gaskets may be made of any of the following materials:

- Commercial asbestos, cork or paper gasket material
- Fluorocarbons (Teflon TFE, Kel-F, Halon, TFE or equivalent)
- Neoprene
- Polyethylene

Hoses for cryogenic fluids should be designed and engineered for this service.

B24.5.2.6 Valves - A steel plug valve is recommended as the most suitable for hydraulic fluids.

B24.5.2.7 Venting Systems and Pressure Relief - All openings in the storage system should terminate outdoors and should be protected by approved flame arresters. Vents should be sized according to the specifications given in the National Fire Codes. All vents and pressure-relief systems will terminate at a height and location that will give adequate protection for personnel and buildings. Vents on atmospheric tanks should be of the pressure-vacuum type to avoid collapse of tanks when withdrawing fuel and to relieve pressure when filling.

B24.5.2.8 Grounding - Since hydraulic fluids are flammable and nonconductive, all stationary or mobile tanks should be bonded and grounded to prevent fuel charging static. The ground resistance should be monitored regularly and not exceed 25 ohms.

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CHAPTER B25
MERCURY

B25.1 PROPERTIES

B25.1.1 Identification

Mercury is identified by the chemical formula Hg. Its molecular weight is 200.59 g mol. It is also known as quicksilver (References 1 and 2).

B25.1.2 General Appearance and Common Uses

Mercury is a silvery, mobile, and odorless liquid. It has many uses in the manufacture of chemicals, batteries, lamps, gauges, dental fillings, and power tubes.

B25.1.3 Physical and Chemical Properties (Reference 3)

The specific chemical properties for mercury are listed in Figure B25-1.

B25.1.4 Solubility

The solubility of mercury in water is .002g/100g water at 293 K (68°F).

B25.1.5 Stability

Mercury is stable.

B25.1.6 Reactivity

Contact of mercury with acetylene, acetylene products, or ammonia gases may form solid products that are shock sensitive. Mercury attacks most transition metals and their alloys. An exception is iron which is useful in containing either pure mercury or its waste product.

B25.2 HAZARDS

B25.2.1 Health Hazards

B25.2.1.1 Toxicity - Metallic mercury is a mobile silvery slightly volatile liquid. Although the vapor pressure is low, dangerous concentrations can accumulate in confined spaces. Inhalation of mercury vapor can cause headaches, cough, chest pains and tightness, and difficulty in breathing. It may also cause pneumonitis, soreness of the mouth, loss of teeth, nausea, and diarrhea. Liquid mercury may irritate the skin. Repeated or prolonged exposure may cause, in addition to the above, shaking of the hands, eyelids, lips, tongue or jaw, allergic skin rash, sores in the mouth, insomnia, excess salivation, loss of memory, personality change, irritability, indecision, and intellectual deterioration. Kidney damage has been observed. In well kept and well-ventilated areas, exposure to mercury vapors should present no problem under ordinary conditions. When mercury is spilled, however, the liquid is broken into many fine droplets, which tend to collect in crevices and under furniture and equipment. Unless these are sought out and collected, they represent a source of vapor contamination which will remain for many months, and can substantially increase the mercury concentration, especially in poorly ventilated areas. Substantially increased concentrations of mercury can be produced when mercury is heated, as in a fire, or by contact with heated equipment. etc. Application of finely divided sulfur or the commercially available mercury decontaminant, HgX, will immobilise the mercury by formation of Mercury Sulfide, a non-volatile and insoluble solid (Reference 4).

Personal hygiene is of upmost importance in working with and handling mercury. Since mercury can be absorbed through the skin, extreme care should be taken not to handle mercury or mercury contaminated components directly. Exposed skin should be thoroughly washed with soap and water or suitable hand cleaner following clean-up procedures. Persons coming in contact with mercury should not smoke until hands are washed. Clothing on which mercury has been spilled should be cleaned of visible mercury, then removed, double bagged, and disposed of as mercury waste.

B25.2.1.2 Threshold Limit Value-Time Weighted Average (TLV-TWA) - The TLV-TWA is 0.05 mg/m^3 (Reference 5).

B25.2.1.3 Exposure Limits - The OSHA permissible exposure limit (PEL) for mercury is 0.1 mg/m^3. NIOSH has recommended that the PEL be changed to 0.05 mg/m^3 averaged over an eight hour shift (Reference 1).

The IDLH value for mercury is 28 mg/m^3. The IDLH value is "immediately dangerous to life or health," meaning that it is the maximum level from which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects (Reference 6).

B25.2.1.4 Special Medical Information - Mercury can enter the body by inhalation, skin absorption, ingestion and contact with skin or eyes. It affects the skin, respiratory system, central nervous system, kidneys and eyes. Symptoms include cough, chest pain, dyspnea, bronchitis, pneumonia, tremor, insomnia, irritation, indecision, headache, fatigue, weakness, stomatities, salivation, gastrointestinal problems, anorexia, low weight, protonuria, irritated eyes, and skin irritation (Reference 1).

For first aid information, see CPIA 394, Appendix E25.

B25.2.2 Fire and Combustion Product Hazards

Mercury itself is not combustible and does not have hazardous decomposition products. However, "contact with acetylene, acetylene products, or ammonia gases may form solid products that are sensitive to shock and which can initiate fires of combustible materials" (Reference 1).

B25.2.3 Explosion Hazards

None.

B25.2.4 Environmental Effects

Mercury and its inorganic compounds have little effect on environment being intrinsic visitors within the earth's crust. However, introduction of metallic mercury into receiving waters either by spill or dissolution by acid rain subjects mercury to conversion to methyl mercury by microorganisms in the bottom of sediment of lakes and rivers. This transformation greatly enhances both bioavailability and toxicity. The brain is the target organ for methyl mercury, while the kidney is the target organ for inorganic mercury. Care, therefore, should be taken to preclude release of free mercury or its soluble compounds into the environment (Reference 7).

B25.3 RESERVED

B25.4 RESERVED

B25.5 MATERIALS AND EQUIPMENT COMPATIBILITY

Mercury is corrosive to most non-ferrous metals and their alloys and should be excluded from spaces where micro control components for missiles and satellites are exposed.

Mercury forms explosive compounds with acetylene and ammonia and sensitizes such compounds as chlorine dioxide, methyl azide and sodium carbide into explosive mixtures. Mercury salts form explosive compounds upon contact with nitromethane (Reference 9).

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PROPERTY		ENGLISH UNITS	SI UNITS
Boiling Point		674°F	630 K
Freezing Point		-38°F	234 K
Density (Gas)			
(Liquid)		846 lb/ft^3 at 68°F	13.55 Mg/m^3 at 293 K
Specific Gravity - Vapor (relative to STP air)		6.93	6.93
Critical Density			
Critical Pressure			
Critical Temperature			
Vapor Pressure		2.4 x 10^{-6} psia at 68°F	1.67 x 10^{-4} kPa at 293 K
Coefficient of Viscosity	Kinematic		
	Absolute	1.55 Centipoises at 68°F	1.55 x 10^{-3} Pa·s at 293 K
		0.99 Centipoises at 523°F	9.9 x 10^{-4} Pa·s at 546 K

Figure B25-1 Physical Properties of Mercury (MW = 200.59 g mol)

CHAPTER B26
COMPRESSED AND LIQUID AIR

B26.1 PROPERTIES

B26.1.1 Identification

Air is a mixture, the composition of which varies with the altitude at which the sample is taken. Air is packaged commercially in various types of cylinders at various pressures. The specification of the average composition of dry air at surface altitudes is shown in Table B26-1. It will condense into a bluish, mobile liquid at 84 K (-308°F) (References 1, 2).

Table B26-1 Composition of Dry Air at Surface Altitudes

MOLECULAR WEIGHT	GAS	MOLE%
28.97	Nitrogen	78.084
	Oxygen	20.946
	Argon	.943
	Carbon dioxide	.033
	Rare Gases (Neon, Helium, Krypton Hydrogen, Xenon, Radon)	.003

B26.1.2 General Appearance and Common Uses

Air, a mixture of gases, is colorless and odorless. It is a source of oxygen, nitrogen, and rare gases. Liquid air is a bluish, mobile liquid. Atmospheric air is used in air conditioning systems and cooling of hot fluids with heat exchangers. Air is the source of oxygen for burning, respiration of plants and animals, decay, and industrial oxidations. Liquid air is used in cryogenic applications.

B26.1.3 Physical and Chemical Properties

The specific chemical properties of air are listed in CPIA 394.

B26.1.4 Solubility

Air is only slightly soluble in water.

B26.1.5 Stability

Air is stable.

B26.1.6 Reactivity

The reactivity of air is due to its content of oxygen. Many metals are converted to oxides when they are heated in air. Flammable materials and organic matter which have been in contact with liquid air may explode easily.

B26.2 HAZARDS

B26.2.1 Health Hazards

The major hazard to personnel from liquid air is frostbite due to skin contact.

B26.2.1.1 Toxicity - (Not Applicable)

B26.2.1.2 Exposure Limits - (Not Applicable)

B26.2.1.3 Specific Medical Information - For first aid information, see CPIA 394, Appendix E26.

B26.2.2 Fire and Combustion Product Hazards

Flammable mixtures are produced with fuels.

B26.2.3 Explosion Hazards

Mixing with fuels causes a dangerous explosion hazard. Mixtures of frozen fuel and liquid air may be shock-sensitive.

B26.2.4 Environmental Effects

Not applicable.

B26.3 RESERVED

B26.4 RESERVED

B26.4 MATERIALS AND EQUIPMENT COMPATIBILITY

Liquid air is classified as a liquified compressed gas, for transfer and storage purposes, and must be handled and stored in fixed or mobile containers of approved design, materials, and construction.

B26.5.1 Materials

When selecting materials for liquid service, consideration should be given to physical properties at low temperature, and the reactivity of the material with liquid air. The ability to withstand stress concentrations, particularly those resulting from sudden temperature changes, is important. Fourteen percent oxygen in air may create a dangerous fire hazard if it escapes or leaks into combustible materials. Compressed oxygen in compressed air in the presence of oils and greases may cause fire (Reference 3). Contact with oxidizing materials may cause liquid oxygen in liquid air to explode (Reference 4).

B26.5.1.1 Metals - Metals to be used in liquid air equipment should possess satisfactory physical properties at extremely low operating temperatures. The following metals are recommended for service with liquid:

a. Aluminum and aluminum alloy types

1000	3000	5083	5454	6062
2014	5050	5085	5456	6063
2024	5052	5154	6061	7075

b. Stainless steel types

304	316	304L
310	321	304ELC

c. 9 percent nickel steel alloy

d. Copper and copper alloys

Copper	Aluminum bronze
Naval brass	Cupro-nickel

e. Nickel and nickel alloys

Nickel	Inconel-X
Rene 41	Hastelloy B
K-Monel	

B26.5.1.2 Non-Metals - The number of acceptable non-metals is small due to the extremely low temperatures encountered. The following list contains non-metals known to be acceptable:

Tetrafluoroethylene polymer (TFE, Halon TFE, Teflon, or equivalent)
Unplasticized chlorotrifluoroethylene polymer (Kel-F, Halon CTF, or equivalent)
Asbestos
Special silicon rubbers

B26.5.1.3 Lubricants - Some petroleum based lubricants could be used. Special lubricants such as the fluorolubes or the perfluorocarbons are recommended.

B26.5.2 Equipment

B26.5.2.1 Containers - Liquid air should be stored in stationary or mobile tanks of approved materials and construction. Storage and shipping drums used for other propellants are not to be used in this service. To insure against defects in materials or fabrication, the storage tanks should be tested as required by the provisions of applicable ASME or DOT specifications for unfired pressure vessels (Reference 5). Materials used for pressure vessels operating at temperatures less than 244 K (-20°F) should be impact-tested in accordance with Paragraph UG-84, Section VIII, of the ASME Boiler and Pressure Vessel Code (Reference 5). Containers for the shipment, storage, and transfer of liquid air fabricated in accordance with any standard that meets pertinent structural requirements. Storage containers should be vacuum-jacketed; the vacuum space may contain reflective insulation or powders. The storage tank itself should be of welded construction and should be equipped with an adequate pressure-relief system (Section B26.5.2.8). Bottom outlets on storage tanks are recommended, since they materially simplify the transfer system design and the selection of pumping equipment.

B26.5.2.2 Pumps and Hoses - Since the storage tanks may be designed with bottom outlets, flooded-suction centrifugal pumps may be used when gravity flow is not applicable. Only pumps and shaft seals designed for liquid air service should be used. Details on these pumps and hoses may be secured from manufacturers of air handling equipment. Hoses should be of proper design and engineered specifically for liquid-air service.

B26.5.2.3 Lights - Temporary portable extension cords with lights, used in inspecting the interior of containers, should be of a type approved by the Mine Safety and Health Administration (MSHA).

Flash lights or storage battery lamps, where permitted for use, should be of the safety type approved by the MSHA or as accepted by the current edition of the National Electrical Code (Reference 6).

B26.5.2.4 Pipes and Fittings - The pipes and fittings should be of approved material and construction, and should be hydrostatically tested at specified pressures. The uses of welded and flanged connections, whenever possible, is recommended. Threaded connections sealed with litharge and water are permissible, when other methods are not feasible. Threaded connections may also be sweated with soft solder. A satisfactory threaded joint seal is produced by covering pipe threads (except for first three threads) with polytetrafluoroethylene tape before making up connections.

B26.5.2.5 Gaskets - Gaskets may be made of soft metals selected from those listed in Section B26.5.1.

B26.5.2.6 Pressure Gauges - Liquid air equipment should be monitored with acceptable LOX-clean types of pressure gauges as required. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales. Gauges should be protected with blowout relief backs or plugs.

B26.5.2.7 Valves - The use of extended stem gate, globe, or ball valves provided with venting devices is recommended.

B26.5.2.8 Venting Systems and Pressure Relief - The storage container itself should be equipped with a bursting disc and a pressure-relief valve in parallel, both discharging to the outdoor atmosphere through an adequately sized vent line. The insulated area, between the inner and outer shells, should be equipped with either a rupture disc or a pressure-relief device, so that pressure cannot build up and rupture the vessel. All lines and vessels in which liquid air may be trapped between closed valves should have pressure-relief valves; if it is likely that the relief valve may freeze, rupture discs should also be provided.

B26.5.2.9 Grounding - Since air supports combustion, all stationary or mobile tanks should be properly bonded and grounded.

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CHAPTER B27
CARBON DIOXIDE

B27.1 PROPERTIES

B27.1.1 Identification

Carbon dioxide has a chemical formula of CO_2 and molecular weight of 44 g mol. It is primarily a gas but it can be a liquid or solid.

B27.1.2 General Appearance and Common Uses

Carbon dioxide is a nonflammable, colorless, odorless, slightly acid gas. In high concentrations, it has an acidic taste. It is used in the carbonation of beverages, manufacture of carbonates, in fire suppression, in refrigeration processes and as propellant in aerosols.

B27.1.3 Physical and Chemical Properties

The physical properties for carbon dioxide are listed in Figure B27-1.

B27.1.4 Solubility

Carbon dioxide is soluble in water (88 ml/100 ml H_2O) at 101 kPa (760 mm Hb) at 293 K (68°F). It is more soluble at higher pressures. It is less soluble in alcohol and other neutral organic solvents.

B27.1.5 Stability

Carbon dioxide is stable.

B27.1.6 Reactivity

Liquid or solid carbon dioxide will attack some forms of plastics, rubber and coatings. It is absorbed by alkaline solutions with the formation of carbonates.

B27.2 HAZARDS

B27.2.1 Health Hazards

B27.2.1.1 Toxicity - Carbon dioxide is an asphyxiant and can also paralyze respiratory system at high concentrations. Symptoms result when sufficiently high concentrations are present to displace oxygen in the air making it insufficient to support life. The symptoms of this (asphyxia) include headache, dizziness, shortness of breath, muscular weakness, drowsiness, and ringing in the ears. Removal from exposure results in rapid recovery (Reference 4).

B27.2.2 Exposure Limits

The threshold limit value-time weighted average (TLV-TWA) for CO_2 is 5000 ppm (9000 mg/m^3); the short term exposure limit (TLV-STEL) is 15,000 ppm (27,000 mg/m^3) (Reference 5). NIOSH suggests 10,000 ppm (18,000 mg/m^3) as the 10 hour time weighted average value (TWA), and 30,000 ppm (54,000 mg/m^3) as the 10 minute ceiling value. The IDLH is given as 50,000 ppm (5%) (90,000 mg/m^3) (Reference 1).

The IDLH value is "immediately dangerous to life or health," meaning that it is the maximum level form which one could escape in 30 minutes without escape-impairing symptoms or any irreversible health effects.

B27.2.3 Special Medical Information

Carbon dioxide can damage the body due to inhalation (asphyxiation) and contact with the solid material. Damage by inhalation is to the lungs and cardiovascular system. Contact with the skin or eyes cause frostbite. Symptoms due to inhalation include headache, dizziness, restlessness, paralysis, dyspnea, sweat, malaise, increased heart rate, elevated blood pressure, pulse pressure, coma, asphyxia, and convulsions. For first aid information, see CPIA 394, Appendix E27.

B27.2.4 Fire and Combustion Product Hazards

Carbon dioxide itself is not combustible, but contact with chemically active metals such as sodium, potassium, or hot titanium may cause fire. It also reacts vigorously with (AL + Na_2O_2), Cs_2O, $Mg(C_2H_5)$, Li, (Mg + Na_2O_2), K, KHC_2, Na, Na_2C_2, NaK alloy, and Ti (Reference 6).

B27.2.5 Explosion Hazards

Contact with reactive metals may produce a violent reaction. Solid carbon dioxide, if confined, will rapidly build up extremely high pressures as it thaws and will burst most containers.

B27.2.6 Environmental Effects

B27.3 RESERVED

B27.4 RESERVED

B27.5 MATERIALS AND EQUIPMENT COMPATIBILITY

When handling or storing liquid carbon dioxide, important factors must be taken into account:

a. Liquid or solid carbon dioxide is extremely cold, and

b. it is an asphyxiant.

It must, therefore, be stored in fixed or mobile containers of approved design, materials and construction and suitably housed.

The installation requirements for pressurized-gas systems are very stringent. Particularly important are the selection of materials and equipment, the use of adequate fabrication procedures and proper maintenance. Personnel assigned to operate these systems must be qualified and familiar with the equipment and proper operating procedures.

B27.5.1 Materials

When subjected to the temperature extremes of liquid carbon dioxide service, many materials undergo a marked physical change. The extent of the change for a given material should be known before it is specified for low temperature use. Metals used with carbon dioxide should be able to withstand impact shock at low temperatures, as well as the stresses produced by the extremely low temperature of liquid carbon dioxide.

The prime consideration in selecting materials to be used with high-pressure gases is their strength. Reactivity or corrosivity, with the exception of that due to moisture where carbonic acid may be produced, should not be a problem with carbon dioxide; but, it is a consideration in the maintenance of purity and cleanliness with regard to the end use.

B27.5.1.1 Metals - Ordinary carbon steels and most ferritic and martensitic alloy steels are unsuitable for liquid carbon dioxide service because they lack ductility at low temperatures. The following metals are satisfactory for this service:

Austenitic chrome-nickel steels	nickel steel (9%)
stainless steel, 18-8 series	copper-silicon alloys
copper	Monel
brass	aluminum
bronze	shredded lead
	titanium

These metals, along with carbon steels, may be used with gaseous carbon dioxide.

B27.5.1.2 Non-Metals - Non-metal materials for liquid carbon dioxide service must also be selected to withstand low temperatures. The following nonmetals are suitable for this service and may be used singly or in any combination:

Tetrafluoroethylene (TFE Teflon, TFE Halon, or equivalent)
Polychlorotrifluoroethylene (Kel-F, Halon CTF, or equivalent)
Selected types of graphite

The following non-metals are suitable for use with gaseous carbon dioxide:

synthetic and natural rubbers
Teflon
Kel-F
DC-55
Fluorolube (LG-160)

B27.5.1.3 Lubricants - Materials used in handling carbon dioxide must be free of grease, oil and other combustible materials. Use special lubricants, such as the fluorolubes or perfluorocarbons, unless aluminum is present.

B27.5.2 Equipment

B27.5.2.1 Containers - Liquid carbon dioxide may be stored in stationary or mobile tanks of approved materials and construction. Gaseous carbon dioxide may be stored and shipped in cylinders conforming to DOT specifications. Storage tanks and cylinders should be tested as required by applicable provisions of the ASME Unfired Pressure Vessel Code (Reference 10) or DOT specification to ensure against defects in materials or fabrication. Materials used for pressure vessels used at temperatures lower than 244 K (-20°F) should be impact-tested in accordance with Paragraph UG-84, Section VIII, of Reference 10.

Containers for shipping, storing and transferring liquid carbon dioxide may be fabricated in accordance with any standard which meets the structural requirements for that container. The storage tank itself should be of welded construction; it should be equipped with a rupture disc and a pressure-relief valve that has an adequate vent line. The vents should discharge to the atmosphere.

Storage containers for carbon dioxide gas should comply with one of the following:

a. Designed, constructed, and tested in accordance with appropriate requirements of the ASME Boiler and Pressure Vessel Code, Section VIII, Unfired Pressure Vessels (Reference 10).

b. Designed, constructed, tested, and maintained in accordance with DOT regulations and specifications.

B27.5.2.2 Pumps and Hose - Pumps for high-pressure gaseous carbon dioxide service should be designed, fabricated, and installed in accordance with the pertinent standards and codes.

For gaseous carbon dioxide, flexible hose generally conforms to military or industrial specifications. Unless otherwise specified in the governing code, a minimum safety factor of 4 to burst is required. Flexible hose should only be employed when the use of rigid tubing is impractical. The hose should be of minimum length. When long runs are unavoidable, hose reels and other protective devices, including hold-down weights and chain ties at end connections, should be incorporated.

Hoses for liquid carbon dioxide use should be of proper design, engineered specifically for this service.

B27.5.2.3 Lights - Any lighting which meets local electrical codes will be adequate in areas where carbon dioxide is stored or transferred (Reference 11).

B27.5.2.4 Pipes and Fittings - The pipes and fittings should be of approved materials and construction and should be hydrostatically tested at specified pressures. The use of welded and flanged connections is recommended whenever possible. Threaded connections are permissible when other methods are not feasible. TFE tape thread sealant is recommended.

Tubing may also be used for high-pressure gaseous carbon dioxide service. Tubing connections will be made with standard AN or industrial flared fittings of like materials. Stainless steel fittings should be used with stainless steel tubing to prevent galvanic action of dissimilar metals. The fabrication of pipe or tube anchors, hangers, supports and tie-downs should be in accordance with the American Society of Mechanical Engineers requirements for pressure piping. The fabrication techniques and selection of materials for the installation and testing of piping systems should be in accordance with the ASME codes. All high-pressure systems should be properly marked and identified.

B27.5.2.5 Gaskets - Gaskets may be made of the materials listed in Section B20.5.1. In high-pressure systems, "O" rings may also be used. Gaskets may be flanged, serrated, laminated, corrugated, or ring-joint types.

B27.5.2.6 Pressure Gauges - Liquid carbon dioxide equipment should be monitored with pressure gauges of approved type as required. Standard-type pressure gauges incorporating compatible materials will be used in gaseous carbon dioxide systems. In addition, gauges used in gaseous carbon dioxide systems should have plastic faces and blow-out discs. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales.

B27.5.2.7 Valves - Extended stem gate, globe and ball valve are satisfactory for liquid carbon dioxide service. All valve packing and gaskets should be of approved materials.

For gaseous carbon dioxide all gate and globe valves 63.5 (2.5 in) in size and larger should be of the outside-screw, rising stem type, whenever this is practicable; valves under 63.5 mm (2.5 in) may be of the inside-screw, rising-stem type. Plug valves may be used instead of gate or globe types. Plug valves for high-temperature service should be designed to prevent galling by making the plug and the valve body of different materials, by treating the plug to give it different physical properties, by welding hard-face overlays to the surface, or by using special mechanical designs. Every valve should be plainly marked to identify the gas and valve function.

B27.5.2.8 Venting Systems and Pressure Relief - When the maximum allowable inlet pressure to one or more of the pressure-reducing devices is greater than the piping systems maximum allowable operating pressure, one or more pressure-relief or other safety devices should be included, e.g., automatic shut-off valves. Suitable protective measures should be taken to prevent injury or damage resulting from the discharge of gases from those safety devices.

The relief valves should have a combined discharge capacity ensuring that the pressure rating of the lower pressure piping system will not be exceeded as the result of an equipment failure, for instance, the pressure-reducing valves. The relief or safety valve should be locked next to the pressure-reducing valve. Pressure-reducing and relief devices should conform to the valve requirements for the specified service conditions.

All lines and vessels in which liquid carbon dioxide may be trapped between closed valves should be equipped with pressure relief valves. Associated vessels and lines must also be equipped with rupture discs in parallel with the relief valve.

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PROPERTY		ENGLISH UNITS	SI UNITS
Sublimation Point		-109.3°F	194.65 K
Freezing Point		-69.8°F	216.58 K
Density (Gas)		0.176 lb/ft^3 at -109.3°F and 14.7 psia	2.814 kg/m^3 at 194.65 K and 101 kPa
(Solid		97.51 lb/ft^3 at -109.3°F and 14.7 psia	1.562 Mg/m^3 at 194.65 K and 101 kPa
Specific Gravity - Vapor (relative to STP air)		1.52	1.52
Critical Density		28.97 lb/ft^3	0.464 Mg/m^3
Critical Pressure		1071 psia	7.383 MPa (absolute)
Critical Temperature		87.8°F	304.2 K
Vapor Pressure		14.696 psia at -109.3°F	101 kPa at 194.65 K
Coefficient of Viscosity	Kinematic		
	Absolute	0.0148 centipoise at 70°F and 14.7 psia	1.48×10^{-5} Pa·s at 294.3 K and 101 kPa

Figure B27-1 Physical Properties of Carbon Dioxide (MW = 44 g mol)

CHAPTER B28
NITROUS OXIDE

B28.1 PROPERTIES

B28.1.1 Identification

Nitrous oxide has a minimum purity of 98.0%, the principal impurity being air. Average analyses show a purity of 98.5%. It is also known as "laughing gas" and sometimes called dinitrous oxide monoxide. It is shipped as a liquified compressed gas under its own vapor pressure of about 5.14 MPa (745 psig) at 294 K (70°F) (Reference 1). The molecular weight of nitrous oxide is 44.1 g mol.

B28.1.2 General Appearance and Common Uses (References 1 and 2)

Nitrous oxide is a colorless, nonflammable, nontoxic gas with a slightly sweet taste and odor. It is used chiefly in rocket fuel formulations with carbon disulfide.

B28.1.3 Physical and Chemical Properties (References 1 and 2)

Nitrous oxide is nonflammable but will support combustion. It will not combine with other materials at ordinary temperatures. Dissociation begins above 573 K (572°F) when the gas becomes a strong oxidizing agent. CPIA 394 describes the physical properties of nitrous oxide.

B28.1.3.1 Solubility (Reference 1) - Gaseous nitrous oxide is somewhat soluble in water and more soluble in alcohol.

B28.1.3.2 Stability (Reference 2) - Gaseous nitrous oxide is chemically stable at room temperature.

B28.1.3.3 Reactivity - At ordinary temperature nitrous oxide is stable and inreactive. At elevated temperature it decomposes into nitrogen and oxygen and becomes an oxidizer and supports combustion. Above 838 K (1049°F), the rate of decomposition becomes appreciable.

B28.2 HAZARDS

B28.2.1 Health Hazards

Nitrous oxide presents no significant health hazard.

B28.2.1.1 Toxicity - Nitrous oxide is nontoxic and nonirritating and is extensively used a an anesthetic in medicine and dentistry. It is a rather weak anesthetic and must be inhaled in high concentrations, mixed with air or oxygen. When inhaled without oxygen, it is a simple asphyxiant. Inhalation of small amounts often produces a type of hysteria; hence its common name, "laughing gas" (Reference 1).

B28.2.1.2 Threshold Limit Value-Time Weighted Average (TLV-TWA) - None established. (Oxygen at 16-17 percent by volume at 101 kPa (760 mm Hg) total pressure is the generally accepted minimum for a work environment.)

B28.2.1.3 Emergency Exposure Limits - None established.

B28.2.1.4 Special Medical Information - For first aid information, see CPIA 394, Appendix E29. Rapid application of oxygen may reverse the effects of severe anoxia. Nitrous oxide is a simple asphyxiant, and consequently symptoms are derived from anoxia (i.e., air, hunger, nausea, vomiting, bewilderment, loss of conscicousness, etc.).

B28.2.2 Explosion Hazards

None established. Nitrous oxide is a strong oxidizing agent. Therefore, it will form explosive or combustible mixtures with any fuel (Reference 3). If exposed to external heating, cylinders containing liquid nitrous oxide may undergo hydrostatic pressure rupture. DOT specification cylinders are therefore fitted with safety devices.

B28.2.3 Fire Hazards

None established (See Section B28.1.3).

B28.3 RESERVED

B28.4 RESERVED

B28.5 MATERIALS AND EQUIPMENT COMPATIBILITY

B28.5.1 Materials

The prime consideration in selecting materials to be used with high-pressure gases is their strength. Reactivity or corrosivity are not problems with nitrous oxide. Maintenance of purity and cleanliness is important for end use.

B28.5.1.1 Metals - The following metals are suitable for use with gaseous nitrous oxide:

Austenitic chrome-nickel steels
stainless steel, series 18-8
copper
brass
bronze
copper silicon alloys
Monel
aluminum
shredded lead
carbon steel

B28.5.1.2 Nonmetals - The following nonmetals are suitable for use with nitrous oxide:

synthetic and natural rubbers
Teflon
Kel-F
DC-55
Fluorolube (LG-160)

B28.5.1.3 Lubricants - If nitrous oxide is used to pressurize oxidizer systems, the use of petroleum-base lubricants with these gases is prohibited. Lubricants such as the fluorolubes or the perfluorocarbons are recommended. The fluorolubes should not be used with aluminum.

B28.5.2 Equipment

B28.5.2.1 Containers - Storage containers should comply with one of the following:

a. Designed, constructed, and tested in accordance with appropriate requirements of the ASME Boiler and Pressure Vessel Code, Section VIII, Unfired Pressure Vessels (Reference 6).

b. Designed, constructed, tested, and maintained in accordance with DOT regulations and specifications.

B28.5.2.2 Pipes and Fittings - Either pipe or tubing may be used for high-pressure service. Connections for piping may be flange, thread, socket weld or butt weld. Tubing connections will be made with standard AN or industrial flared fittings of like materials. Stainless steel fittings should be used with stainless steel tubing to prevent galvanic action of dissimilar metals. Welded or brazed joints are preferred. The fabrication of pipe or tube anchors, hangers, supports and tie-downs should be in accordance with the American Society of Mechanical Engineers (ASME) requirements for pressure piping.

The fabrication techniques and selection of materials for the installation and testing of piping systems should be in accordance with the ASME codes. High-pressure systems should be properly marked and identified.

B28.5.2.3 Gaskets and Seals - In high-pressure systems, "O" rings may be used. Gaskets may be flange, serrated, laminated, corrugated, or ring-joint types. The materials listed in Sections B28.5.1.1 and B28.5.1.2 may be used in the fabrication of gaskets and seals.

B28.5.2.4 Valves - Wherever practicable, all gate and globe valves, 6.4 cm (2.5 in) in size and larger, should be of the outside-screw, rising-stem type; valves under 6.4 cm (2.5 in) may be of the inside-screw, rising-stem type.

Plug valves may be used instead of gate or globe types. Plug valves for high temperature service should be designed to prevent galling by making the plug and the valve body of different materials, by treating the plug to give it different physical properties, by welding hard-face overlays to the surface, or by using special mechanical designs. Every "valve" should be plainly marked to identify the gas and valve function.

B28.5.2.5 Pumps and Hose - Flexible hose generally conforms to military or industrial specifications. Unless otherwise specified, a minimum safety factor of 4 to burst is required. Flexible hose should only be employed when the use of rigid tubing is impractical. The hose should be of minimum length. When long runs are unavoidable, hose reels and other protective devices, including hold-down weights and chin ties at end connections, should be incorporated.

Pumps for high-pressure service should be designed, fabricated and installed in accordance with pertinent standards and codes.

B28.5.2.6 Pressure Gauges - Standard-type pressure gauges incorporating compatible materials will be used in gaseous nitrous oxide systems. In order to minimize operator reading errors, all pressure gauges used for a common purpose should have identical scales. In addition, all gauges should have plastic faces and blow-out discs.

B28.5.2.7 Venting Systems and Safety Relief - When the maximum allowable inlet pressure to one or more of the pressure-reducing devices is greater than the piping system's maximum allowable operating pressure, one or more pressure-relief or other safety devices should be included (e.g., automatic shut-off valves). Suitable protective measures should be taken to prevent injury or damage resulting from the discharge of gases from those safety devices.

The relief valves should have a combined discharge capacity ensuring that the pressure rating of the lower pressure piping system will not be exceeded as the result of an equipment failure, for instance, the pressure-reducing valves. The relief or safety valve should be located next to the pressure reducing valve. Pressure-reducing and relief devices should conform to the valve requirements for the specified service conditions.

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Appendix C
System Descriptions

APPENDIX C
SYSTEM DESCRIPTIONS

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UPPER STAGE VEHICLES

SATELLITE VEHICLES

APPENDIX C
SYSTEM DESCRIPTIONS

C.0 INTRODUCTION

Appendix C provides system descriptions of launch vehicles, upper stage vehicles and satellite vehicles currently in use. The purpose of the system description data are:

(1) To provide data on hazards in existing vehicles.

(2) Identify methods of hazard control used in existing vehicles.

(3) Provide system description data to be used in submitting new safety documentation, such as the Accident Risk Assessment Report or Missile Systems Pre-Launch Safety Package.

The mission scenarios provided for each vehicle are typical missions.

Appendix C1
Space Transportation System—Centaur

APPENDIX C1
SPACE TRANSPORTATION SYSTEM (STS)

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APPENDIX C1
SPACE TRANSPORTATION SYSTEMS (STS)

C1.0 INTRODUCTION

The following data were extracted from the national STS Program Document "Space Shuttle Data for Planetary Mission Radioisotope Thermoelectric Generator (RTG) Safety Analysis," JSC 08116, Feb 15, 1985, NASA, L.B. Johnson Space Center, Houston, Texas 77058. This description includes missions carrying Centaur upper stage.

C1.1 SPACE TRANSPORTATION SYSTEM - GENERAL DESCRIPTION

The STS consists of the Shuttle, a variety of standard payload carriers, ground integration and launch facilities, and groundbased payload operations control centers. The ground integration and launch facilities consist of the KSC and Vandenberg Air Force Base (VAFB) complexes, although the latter is not yet operational for Shuttle launches. There are three groundbased payload operations control centers located at the Jet Propulsion Laboratory (JPL), the Goddard Space Flight Center (GSFC), and the Johnson Space Center (JSC). Communications are via the world-wide Space Tracking and Data Network (STDN), the Tracking and Data Relay Satellite System (TDRSS), and the Orbiter and/or carrier command and data handling system. Direct payload communication is also possible. The Orbiter is transported between ground sites such as Edwards Air Force Base (EAFB) and KSC atop the Shuttle Carrier Aircraft (SCA), a modified Boeing 747 aircraft.

The Shuttle Vehicle consists of the Orbiter, the External Tank (ET), and two Solid Rocket Motors (SRM). Figures C1-1 through C1-4 show the Shuttle Vehicle, with side, top, front, and back views. Figure C1-5 shows the coordinate systems.

C1.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C1.2.1 Orbiter

The Orbiter contains the following subsystems: structure; thermal protection, thermal control; main propulsion; orbital maneuvering; reaction control; mechanical (including ET separation); auxiliary power units; hydraulic; electrical power; power reactant storage and distribution; environmental control and life support; purge, vent, and drain; avionics; guidance, navigation, and control; communications and tracking; and data processing and software. Figures C1-6 through C1-9 show the layout of the Orbiter and many of the parts of the subsystems.

C1.2.1.1 Orbiter Structure - Figures C1-10 through C1-13 show the Orbiter structure. Most of the Orbiter structure is of conventional aluminum construction protected by Reusable Surface Insulation (RSI).

C1.2.1.2 Thermal Protection System - The Thermal Protection System (TPS) keeps the outer skin of the Orbiter airframe from becoming too hot during both ascent and reentry. The TPS materials are attached to the outside of the primary structural shell of the Orbiter. There are several types of TPS materials; the type used depends on the thermal environment at that location.

The TPS is a passive system. It has been designed for ease of maintenance and for flexibility of ground and flight operations while satisfying its primary function of maintaining acceptable airframe outer skin temperatures. The RSI materials are of three types: (1) ceramic LI-900, LI-2200, and FRCI-12 materials used for HRSI and LRSI; (2) Nomex felt material used for FRSI; and (3) quilted fibrous silica material used for AFRSI. Figure C1-14 shows the TPS materials and their location on Orbiter 103.

C1.2.1.3 Thermal Control System - The Thermal Control System (TCS) maintains subsystems and components within specified temperature limits for all mission phases, including prelaunch, launch, Earth orbit, entry, and post-landing. The integrated thermal control management uses available heat sources and heat sinks supplemented by passive thermal control techniques. Those techniques are primarily the use of insulation, thermal control coatings, and thermal isolation to attenuate heat transfer to or from critical spacecraft compartments and hardware. Supplemental heaters are provided as required when these methods are not adequate to provide the necessary temperature control.

C1.2.1.4 Main Propulsion System - The main propulsion system of the Orbiter includes three engines, which burn for about eight minutes from just before lift-off until slightly before attaining orbit. The rated (sea-level) thrust of each engine is 375,000 pounds at sea level and 490,000 pounds in vacuum. The engines are gimballed for steering. The fuel is Liquid Hydrogen (LH_2), and the oxidizer is Liquid Oxygen (LOX). The propellants for the Main Engines (MEs) are in the ET.

Each Space Shuttle Main Engine (SSME) is a reusable, high-performance, liquid-propellant rocket engine with variable thrust. Three engines are clustered on each Orbiter vehicle. The nominal engine chamber pressure is 3000 psia. The nozzle exit diameter is 94 inches. Each engine weighs approximately 6950 pounds. The nozzle is gimballed with hydraulic actuators for steering. They are ignited on the ground prior to launch and burn for an average of eight minutes during the vehicle boost phase. Figure C1-15 shows the main engines and the propellant feedlines from the ET.

C1.2.1.5 Orbital Maneuvering System - The Orbital Maneuvering System (OMS) includes two engines. The thrust from those engines carries the Orbiter into orbit after the main engines are shut down, provides for maneuvering while in orbit, and retards the Orbiter out of orbit for reentry. The nominal engine chamber pressure is 125 lb/in^2. The rated (vacuum) thrust of each engine is 6000 pounds. The fuel is monomethylhydrazine (MMH), and the oxidizer is nitrogen tetroxide (N_2O_4).

The two OMS engines are mounted in pods at the top of the aft fuselage, one on each side of the vertical tail. The nozzles are gimballed for steering, with electromechanical actuators. The OMS propellant tanks in the two pods can carry enough propellant for a change in velocity in orbit of 1,000 ft/s when the vehicle carries a payload of 65,000 pounds. Crossfeed lines connect the propellant tanks in the two OMS pods so that propellants from either pod can run either engine. Also, the Reaction Control System thrusters can burn propellants from the OMS tanks. Figure C1-16 shows an OMS engine pod, which also contains parts of the RCS.

C1.2.1.6 Reaction Control System (RCS) - The RCS of the Orbiter includes 44 thrusters (38 primary, 6 vernier) for attitude control and three-axis translation during orbit insertion and during on-orbit and entry phases of the Orbiter flight.

The RCS consists of three propulsion units, one in the forward module and one in each of the aft propulsion pods. All modules are used for external tank separation, orbit insertion, and orbital maneuvers. Only the aft RCS modules are used for entry attitude control. The RCS employs 38 bipropellant primary thrusters and 6 vernier thrusters to provide attitude control and three-axis translation during the orbit insertion, on-orbit, and entry phases of flight.

The RCS propellants are N_2O_4 and MMH. The design mixture ratio of propellants allows the use of identically sized propellant tanks for both fuel and oxidizer. The nominal propellant capacity of the tanks in each module is 928 pounds of MMH and 1477 pounds of N_2O_4. An interconnect between the OMS and the RCS in the aft pods permits the use of OMS propellant by the RCS for on-orbit maneuvers. In addition, the interconnect can be used for crossfeeding propellants between the right- and left-hand RCS pods.

The thrust of each primary thruster is 870 pounds (in vacuum), and the thrust of each vernier thruster is 24 pounds (in vacuum). Figure C1-17 shows the RCS.

C1.2.1.7 Pressure Tanks - The Orbiter carries many pressure tanks. Table C1-1 lists them, and Figure C1-18 shows their locations. The index numbers on the table refer to the circled numbers on the figure.

Table C1-1 Pressure Tanks in Orbiter

Index Number (a)	Contents	Subsystem	Maximum Operating Pressure, lb/in^2g	Volume in^3	No. of Tanks	Tank Material
1	Air (Crew Cabin)	ECLSS	15	4,363,000	1	Aluminum Alloy
2	Ammonia	ECLSS	550	3,024		Titanium
3	Breathing Oxygen (Not on OV-099 or OV-104 Mission Kit only)	ECLSS	3300	8,181	1	Inconel with Kevlar Winding
4	Freon-21 ($CHCl_2F$)	ECLSS	230	4,337	2	Aluminum
5	Helium	FRCS	4000	3,008	2	Titanium with Kevlar Winding
		ARCS	4000	3,008	2/Pod	Titanium with Kevlar Winding
		OMS	4875	30,033	1/Pod	Titanium with Kevlar Winding
		MPS	4500	29,894	7	Titanium with Kevlar Winding
		MPS	4500	8,122	3	Titanium with Kevlar Winding
		MPS	850	500	2	Titanium

Table C1-1 Pressure Tanks in Orbiter - Continued

Index Number (a)	Contents	Subsystem	Maximum Operating Pressure, lb/in^2g	Volume in^3	No. of Tanks	Tank Material
6	Hydrazine (N_2H_4)	APU	355	11,375	3	Titanium
7	Hydraulic Oil	HYD	3000	98	3	Steel
		HYD	135	1,850	3	Aluminum
		HYD	70	98	3	Steel
8	Liquid Hydrogen (LH_2)(b)	PRSD	320	36,972	3-5	Aluminum alloy pressure vessel and out-shell
9	Liquid Oxygen (LO_2)(b)	PRSD	1035	19,418	3-5	Inconel pressure vessel and aluminum alloy outer shell
10	Monomethyl-Hydrazine - N_2H_4	FRCS	350	29,376	1	Titanium
		ARCS	350	29,376	1/Pod	Titanium
		OMS	313	155,520	1/Pod	Titanium
11	Nitrogen	ECLSS	3600	8,181	4	Titanium with Kevlar Winding
		HYD	3165	113	3	Titanium
		OMS (Eng)	3000	60	2	Titanium

Table C1-1 Pressure Tanks in Orbiter - Continued

Index Number (a)	Contents	Subsystem	Maximum Operating Pressure, lb/in^{2}g	Volume in^3	No. of Tanks	Tank Material
		OMS (Eng)	450	19	2	Titanium
12	Nitrogen Tetroxide (N_2O_4)	FRCS	350	28,512	1	Titanium
		ARCS	350	28,512	1/Pod	Titanium
		OMS	313	155,520	1/Pod	Titanium
13	Water	APU	35	2,352	1	Titanium
		APU	100	435	1	Titanium
		HYD	33	4,546	3	Aluminum
14	Water (Potable, Waste, & Heat Transfer)	ECLSS/ ATCS	20	6,600	5	Aluminum
15	Oxygen	SSME (POGO) ACCUM)				
16	Halon 1301	Fire Ext (Portable)	575	80	3	Steel
		Fire Ext (Fixed)	575	87	3	Stainless Steel
17	Water/O_2	EMU	15	277	3	Aluminum
18	O_2	EMU	1050	247	6	Stainless Steel
		EMU	7400	92	6	Inco 718

C1.2.1.8 Mechanical Systems - The mechanical subsystems of the Orbiter operate the aerodynamic control surfaces, landing/deceleration system, vent doors, payload bay doors, deployable radiators, payload retention and payload handling subsystems, and provide for disconnecting propellant feedlines between the Orbiter and the ET and for separating the Orbiter from the ET.

C1.2.1.9 Auxiliary Power Unit (APU) - The auxiliary power units drive the hydraulic pumps. The units work by decomposition of hydrazine (N_2H_4).

C1.2.1.10 Hydraulic System - The hydraulic subsystem provides hydraulic power for many of the mechanical devices in the Orbiter, for steering the main engines, and for controlling the propellant valves in the main engines.

C1.2.1.11 Electrical Power System - The electrical power system supplies electricity from hydrogen-oxygen fuel cells. The power reactant storage and distribution subsystem provides hydrogen and oxygen to the electrical fuel cells and oxygen to the Environmental Control and Life Support Subsystem (ECLSS). The ECLSS provides a habitable environment for the crew. The purge, vent, and drain subsystem provides environmental control of the Orbiter's unpressurized structural cavities, which include most of the Orbiter areas except for the crew module. The avionics subsystem includes all the electronic gear in the Orbiter. The guidance, navigation, and control subsystem provides guidance commands for control, inertial navigation, and automatic and manual control capability. The communications and tracking subsystem provides communication between the Orbiter and ground stations, the Orbiter and released payloads, and the Orbiter and the Tracking and Data Relay Satellite System. The data processing and software subsystem provides computing capability for the Orbiter.

C1.2.2 External Tank Systems

The External Tank (ET) (see Fig. C1-19) supplies the Orbiter main propulsion system with Liquid Hydrogen (LH_2) and Liquid Oxygen (LOX) at prescribed pressures, temperatures, and flow rates. Both the LH_2 and LOX tanks are equipped with a vent and relief valve to permit loading, pressurization, and pressure relief functions. Tank level sensors transmit propellant loading and shutdown signals. The ET is thermally protected with a nominal 1-inch-thick spray-on foam insulation (SOFI), with additional SOFI and a charring super-lightweight ablator (SLA 561) in places to withstand localized high heating. Since the ET is an expendable element, the ET subsystems are designed for single usage to minimize costs.

The ET consists of a forward LOX tank, an unpressurized intertank, and a LH_2 tank. The LOX tank (volume: 19,500 ft^3) is an aluminum alloy monocoque structure composed of a fusion-welded assembly of preformed, chemmilled gores, panels, machined fittings, and ring chords. The LOX tank is designed to operate at a nominal pressure range of 20 to 22 psig. The tank contains antislosh and antivortex baffles as well as an antigeyser system to control conditions. A 17-inch diameter feedline conveys LOX through the intertank and externally aft to the ET/Orbiter disconnect. The tank's double wedge nose cone reduces drag and heating and also serves as a lightning rod.

The intertank is semimonocoque cylindrical vented structure with stringer and flanges for joining the LOX and LH_2 tanks. The intertank contains the Solid Rocket Booster thrust beam and fittings that distribute SRB loads to LOX and LH_2 tanks. The intertank provides an umbilical plate that interfaces with a ground facility arm. The umbilical plate accommodates purge gas supply, hazardous gas detection, and hydrogen gas boil-off during ground operations.

The LH_2 tank (volume: 53,518 ft^3) is a semimonocoque structure composed of four fusion-welded barrel sections, five beam ring frames, and forward and aft ellipsoidal domes. The LH_2 tank is designed to operate after lift-off at a nominal pressure of 32-34 psia; the lift-off pressure is 40.9-44.1 psia. The tank contains an antivortex baffle and a siphon outlet to transmit propellant to the ET/Orbiter disconnect through a 17-inch-diameter line. The LH_2 tank has provisions for the ET/Orbiter forward attach strut, two ET/Orbiter aft attach fittings, the thrust distribution structure, and the aft SRB/ET stabilizing strut attachments.

To reduce the ET weight and thus achieve greater payload capability, certain changes have been made to the original ET described above. These changes, which have reduced the dry weight of the ET by over 6000 pounds, are incorporated in the lightweight tank (LWT) version of the ET. The LWT first flew on STS-6 and became the standard version from STS-8 onward, after STS-7 used the last of the original-version ETs. Among the changes incorporated in the LWT were the elimination of five LO_2 slosh baffle rings, deletion of the anti-geyser line, and incorporation of helium inject into the main feed line.

The ET contains the Solid Rocket Motor (SRM) thrust through its intertank and provides attach structure to the Orbiter to house the Space Shuttle Main Engines (SSME). At lift-off, the ET contains 1.55 million pounds of usable propellant. At Main Engine Cutoff (MECO), the ET is separated from the Orbiter before orbital velocity is achieved. The ET then proceeds on a ballistic reentry path for a safe impact in the ocean.

The ET is made up of two pressure tanks. The LOX tank, at the forward end of the ET, is made of aluminum alloy. Its volume is 33,696,000 in^3 (19,500 ft^3), and its maximum relief pressure is 25 psia. The larger LH_2 tank also is made of aluminum alloy. Its volume is 53,518 ft^3. Besides the two tanks, there are feedlines carrying propellants under pressure, which run on the outside of the ET on the side toward the Orbiter. Figure C1-19 shows the two tanks of the ET.

C1.2.3 Solid Rocket Motors (SRM)

Two SRMs burn in parallel with the Orbiter main propulsion system (MPS) to provide initial ascent thrust. Primary elements of the booster are the motor, including case, propellant, igniter, and nozzle; structural systems; separation, operational flight instrumentation (OFI), and recovery avionics; separation motors and pyrotechnics; and deceleration, range safety destruct, and thrust vector control (TVC) subsystems. Each SRB (steel-cased) weighs approximately 1.293 million pounds. The propellant weight for each is about 1.1 million pounds.

Figure C1-20 shows an SRM. Each SRM produces 2.9 million pounds of thrust at sea level. The engines are gimballed for steering. The SRM case is segmented. The skirts at the aft end of the SRMs sit on the mobile launch platform before lift-off.

Two lateral sway braces and a diagonal attachment at the aft frame provide the structural attachment between the SRM and the ET. The SRM forward attachment to the tank is by a single thrust attachment at the forward end of the forward skirt. The same forward skirt is used for attaching the main parachute riser attachments.

The SRMs are released from the ET by pyrotechnic devices at the forward thrust attachment and the aft sway braces. Eight separation rockets on each SRM (four aft and four forward) separate the SRM from the Orbiter and tank.

There are two versions of the SRM, and the major difference between them is the structural material of the Solid Rocket Motor (SRM) case itself. The initial version has a Steel Case (SC) SRM; the second version will have a composite Filament-Wound Case (FWC). The major advantages of the FWC-SRM is its lighter weight (about 30,000 lbs less than the SC-SRM) thus making possible a heavier STS payload. However, because of the higher cost of the FWC, it will only be used when its payload advantage is needed. The Centaur planetary missions are planned to use the steel case SRMs.

Overall, the SRM (Fig. C1-21) is 150 feet long and is 148 inches in diameter, although the FWC diameter is slightly greater because of its thicker case. The inert weight of SC-SRM is 190,000 pounds, which is 30,000 pounds greater than the FWC-SRM (Fig. C1-22). The major elements consist of the structural assemblies (aft skirt, forward skirt, and nose assembly) and the four SRM segments. The parachutes are mounted in the nose assembly, electronics in the forward skirt, and the TVC system in the aft skirt. The structural assemblies are designed for 40 uses, the SC-SRM for 20 uses, the electronics and TVC hardware for 20 uses, and the parachutes for 10 uses. The FWC-SRM is designed for single use only, although the feasibility of reusing it is under study.

The SRM are cast and delivered to the launch site in four segments. The SC segments are roll-formed D6ac steel with pinned clevis joints. Two O-ring seals in each joint provide redundancy for the maintenance of pressure integrity. The design and fabrication of the case are scaled-up versions of the Titan III motor cases. Structural design factor of safety for the SC is 1.4, typical of man-rated vehicles. Only the four straight cylindrical sections of the FWC-SRM (Fig. C1-21) are actually filament wound; the other segments are steel as before. The materials used for the FWC are Hercules AS4W-12K graphite fiber (modulus $33x10^6$ psi), and Hercules HBRF-55A resin (bisphenol A epoxy, 1, 4 butanediol epoxy, and curing agent consisting of a blend of methylene dianiline with m-phenylene dianiline). The FWC design safety factors are 1.4 on metal parts and membrane structure and 2.0 on composites at joints and discontinuities.

The composite propellant is a proven PBAN propellant used in the Minuteman and Poseidon systems. More than 200 million pounds of this propellant have been produced. The propellant is vacuum cast and case bonded.

The SRM nozzle is a 20% submerged omnidirectional movable nozzle. The throat diameter is 54 inches, and the diameter at the end of the exit cone is 148 inches. The nozzle has an aft-pivoted flexible bearing that provides omniaxial TVC deflection capability of ± 8 degrees. All metal parts of the nozzle are designed for 20 uses.

The aft skirt provides attach points to the launch support structure and support to the Space Shuttle on the Mobil Launch Platform (MLP) for all conditions prior to SRM ignition. The aft skirt provides aerodynamic protection, thermal protection, and mounting provisions for the TVC subsystem and the aft mounted separation motors. The aft skirt provides sufficient clearance for the SRM nozzle at the full gimbal angles. The aft skirt kick ring provides the necessary structural capability to absorb and transfer induced prelaunch loads.

The aft skirt structure assembly is a welded and bolted conical shape, 146 inches in diameter at the top, 212 inches at the bottom, and 90.5 inches in height. It is configured for left-hand and right-hand assemblies, is fabricated using 2219 aluminum with D6ac steel rings, and weighs approximately 12,000 pounds.

The forward skirt comprises all structure between the forward SRM segment and the ordnance ring. It includes an SRM/ET attach fitting, which transfers the thrust loads to the ET, and a forward bulkhead, which seals the forward end of the skirt. The forward skirt provides the structure to react parachute loads during deployment and descent and provides an attach point for towing the SRM during retrieval operations.

The forward skirt is 146 inches in diameter and 125 inches in height. It consists of a 2219 aluminum welded cylinder assembly made from machined and brake-formed skin panels and a welded thrust post structure. The forward skirt weighs approximately 6400 pounds.

The ordnance ring, 146 inches in diameter, provides a plane for pyrotechnically separating the frustum from the forward skirt during the parachute deployment process. The ring is machined from a 2219 aluminum ring forging and provides mounting provisions for the linear-shaped charge used in the severance function.

The frustum houses the main parachutes, provides the structural support for the forward separation motors, and incorporates flotation devices and location aids (flashing light and RF beacon) for water retrieval operations. It is fabricated using machined 2219 aluminum shear beams, ring fittings, separation motor supports, main parachute supports, and 7075 aluminum formed skins. The frustum weighs approximately 3800 pounds.

The nose cap houses both the pilot and drogue parachutes and is separated from the furstum by three pyrotechnic thrusters to initiate the parachute deployment sequence. The nose cap is an aluminum monocoque structure with a hemispherical section at the forward end. The base is 68 inches in diameter and the overall height is 35 inches. The structure is a riveted assembly of machined 2024 aluminum sheet skins, formed ring segments and cap, and a machined separation ring. Its weight is approximately 300 pounds.

The systems tunnel is located outboard on each booster and houses the electrical cables and linear-shaped charge of the range safety system. The tunnel provides lightning, thermal, and aerodynamic protection and mechanical support for the cables and destruct charge. It is manufactured from 2219 aluminum and extends from the forward skirt along the motor case to the aft skirt. The tunnel is approximately 10 inches wide and 5 inches high. Its floor plate is vulcanized and bonded to each motor segment by Thiokol, the SRM contractor. The overall weight of the systems tunnel is approximately 600 pounds.

The propellant for the SRMs is a composite-type solid propellant formulated of polybutadiene acrylic acid acrylonitrile terpolymer binder, ammonium perchlorate, and aluminum powder. A small amount of burning rate catalyst (iron oxide) is added to achieve the desired propellant burning rate. The propellant is in four segments; Figure C1-21 shows the segments and their lengths. The propellant in the forward segment weighs 299,000 pounds; in each of the two center segments, 272,000 pounds; and in the aft segment, 266,000 pounds. The total propellant weight in both boosters is 2,218,000 pounds. Each booster provides 2.9 million pounds of thrust at sea level.

The higher thrust level required during the lift-off portion of the Shuttle flight results from increased burning surface provided by the 11-point star configuration in the forward segment. After the lift-off portion of the flight, the thrust is reduced with burnout of the starred section. The thrust progress is slighter after 52 seconds. Reduced vehicle acceleration is achieved by burning out the aftmost portion of the aft segment and by the programmed burnout of slivers in all four segments.

The nozzle of the SRM is gimballed for steering with hydraulic actuators. The hydraulic power supply for the TVC subsystem consists of hydrazine-fueled auxiliary power units on each SRM. The SRM TVC subsystem works in conjunction with the TVC system for the Orbiter main engines and provides the preponderance of gimbal authority for the Shuttle during the first stage flight. Pitch, roll, and yaw commands are provided by the Orbiter flight control system.

Each SRM carries several pressure tanks, which are parts of the hydraulic power supply systems of the SRM. Two tanks on each booster hold hydrazine, the fuel for the auxiliary power units. Figure C1-23 shows a hydrazine tank on an SRM as part of the TVC subsystem, and Figure C1-21 shows the location of the TVC subsystems on an SRM.

The forward section provides room for the SRM electronics, recovery gear, range safety destruct system, and forward separation rockets. The parachute deceleration subsystem consists of a pilot parachute, a ribbon drogue parachute, and three ribbon main parachutes.

C1.2.4 SRM/ET Separation

On a nominal flight, the solid rocket motors will separate from the tank about two minutes after lift-off.

Basically, the way of separating the SRMs from the ET is to fracture four separation bolts on each booster and to propel the SRMs from the ET by eight solid rocket thrusters on each booster. Figure C1-24 shows the separation bolts and separation thrusters.

The solid propellant thrusters, four forward and four aft on each SRB, propel the boosters away from the ET. Each solid propellant thruster gives a 23,000-pound thrust.

C1.2.5 ET/Orbiter Separation

The method used to separate the Orbiter from the ET is to fracture one bolt and eight nuts in order to sever structural and umbilical connections and to use the RCS to propel the Orbiter away from the tank. Figure C1-25 shows the separation mechanism.

The forward attachment between the orbiter and the ET is a spherical bearing with an attach-bolt. Firing either of two pressure cartridges on the bolt will break it. There are two structural attachments aft, one on the left and one on the right. Firing either of two pressure cartridges on each attach-bolt will break the frangible nut and release the bolt. There are two umbilical attachments aft, one on the left and one on the right. Each attachment contains three attach-bolts, and firing either of two pressure cartridges on each bolt will break the frangible nut and release the bolt.

C1.2.6 Command Destruct System

The range safety command destruct system allows the intentional destruction of the SRMs and ET if the flight deviates too far from the nominal. Linear shaped charges (LSCs) can split the two tanks in the ET and can split the cases of the boosters on command.

The systems for the three elements (one ET and two SRMs) are connected so that, if either SRM receives the destruct signal, all three receive it. The system in each element is redundant to assure reliability.

On the ET, one LSC is on the LOX tank, and the other is on the LH_2 tank. The charges sit in the cable trays on the outside of the tanks. The charge for the LOX tank is about 8 feet long, and for the LH_2 tank, about 20 feet long, in two 10-foot sections. Figure C1-26 shows the position of the charges on the ET: the destruct charges are on the side toward the Orbiter.

On each SRM, the LSC is about 80 feet long, in six 160-inch sections. The destruct charges sit in the cable trays on the outside of the booster case on the side away from the ET. The destruct charges would split the case, allowing the chamber pressure to drop sharply to terminate thrust. Figure C1-27 shows the position of the destruct charge on an SRM.

Electronically, each of the SRMs and the ET carry a command destruct receiver which will receive the arm and destruct signal from the ground station. Electronically, the three independent systems are cross-strapped so that a signal received by one can initiate destruct action on the other two systems as long as normal separation has not occurred. The SRMs carry two receivers each, and the ET has one. After inadvertent separation of one or both of the SRMs, destruct action could still be initiated on the separated solids and the ET unless the system is damaged. All three would receive the signal and respond unless SRM separation is normal, in which case the destruct systems on the SRMs are safed prior to separation.

C1.2.7 Range Safety

Range safety at the Eastern Space and Missile Center (ESMC) is based primarily on:

1. Range Safety Manual Volume 1. AFETRM 127-1, Air Force Eastern Test Range, September 1, 1972.

   This manual defines in detail the requirements and procedures necessary for obtaining range safety approval for missile and space vehicle operations at ESMC; this range safety approval derives from satisfactory review from the combined efforts of ESMC/SEO, SEM and SEY (Missile Flight Control Division, Missile Systems Safety Division, and Missile Flight Analysis Division), under direction of the Director of Safety, ESMC/SE. In general, SEO/SEY are concerned from lift-off to orbital insertion while SEM's concerns start prior to lift-off through orbital insertion, or after inadvertent mainland impact.

2. Cone, B.E.: The United States Air Force Eastern Test Range, Range Instrumentation Handbook. AFETRM-TR-76-04, July1, 1976.

This handbook describes the major instrumentation systems now operating on the ESMC, and also briefly discusses the equipment, operation, and performance of each system, including the Range Safety System (RSS).

C1.3 MISSION SCENARIO

The basic mission of Space Shuttle is to be a space transport, carrying payloads into space and returning to Earth for another load. What happens to the payloads in space depends on the mission. On some missions the payload is released into orbit. On other missions the payload is a spacecraft and an upperstage rocket; the rocket boosts the spacecraft into another orbit or into an Earth-escape trajectory. On a third kind of mission, the payload stays in the Shuttle Orbiter and is carried back to Earth. On yet a fourth kind of mission, the Orbiter crew retrieves a satellite already in orbit and returns it to Earth.

Figure C1-28 shows the basic mission cycle. It starts with the launch of the Shuttle Vehicle (with the SRBs and the Orbiter main engines firing). After approximately two minutes, at about 26 miles altitude, the SRBs are jettisoned and fall back to Earth for recovery. After approximately eight minutes of flight, before the Orbiter attains orbit, the main engines are shut down, and the ET is separated and falls back to Earth for oceanic termination. The orbital maneuvering system engines thrust the Orbiter into orbit. The mission-dependent orbital operations then ensue and, on their completion, the Orbiter returns to Earth to be made ready for its next flight.

C1.3.1 Operation Cycle

This section describes the operation of the Shuttle Vehicle. One operation cycle extends from when the Orbiter lands to finish one mission until it lands again to finish the next one. The cycle covers ground preparation, launch, ascent, orbit, and return. Figure C1-28 shows the entire cycle.

The ground preparations proceed in five phases: landing; safing, maintenance, and checkout; premate preparation; Shuttle assembly; and prelaunch. The ultimate goal of the Shuttle program is to reduce the preparation time to 160 hours over a two-week period.

The move to the launch pad, connecting interfaces, servicing, checkout, and launch are planned to take a minimum of 24 working hours. The system is designed to be capable of launch within two hours after starting the tanking of fuels and oxidizers.

Loading the propellants into the ET will begin 8.5 hours before launch. This operation commences with line chilling and progresses to slow fill, fast fill, and topping. Because topping of the ET LH_2 tanks is essentially complete several hours before launch, it is necessary to keep replenishing these tanks until a few minutes before launch.

After a successful check, the flight crew will enter the Orbiter (approximately three hours before launch). The ground control will end 31 seconds before launch when automatic sequencing of events from the Orbiter will begin.

The launch part of the operation cycle includes five principal events: ignition of the main engines; sending the signal to ionize the SRBs; lift-off; clearing the tower; and the passage of the instantaneous impact point beyond shallow water. The main engines are started by commands from the orbiter onboard computers beginning at 4.6 seconds before lift-off, using a staggered start sequence (Engines 3, 2, and 1 started sequentially in that order at 120 millisecond intervals). When the computers in the Orbiter determine that all three main engines are operating normally with a least 90% of rated thrust, the computers send the signals to start the SRBs. These signals are sent 40 milliseconds before lift-off. Once the solids light, lift-off will occur with the solids and the main engines running. After approximately seven seconds of flight, the vehicle will clear the tower; and after approximately 34 seconds of flight, the instantaneous impact point will be over water at least 20 fathoms deep.

The ascent part of the operations cycle includes several important events for nuclear safety analysis. The dynamic pressure on the vehicle will grow quite rapidly to be about 650 psf after about 40 seconds of flight, will hold at a peak of approximately 705 psf between 50 and 60 seconds, and then will decline sharply to about 50 psf by the time the SRBs burn out (about 119 seconds into the flight). Once the solids burn out (signaled by the drop of the solid-motor chamber pressure to below 50 psf) they will be jettisoned and will fall clear of the Orbiter and ET after about 128 seconds of flight. The SRBs are lowered by parachutes into the ocean and are recovered and returned to the launch site for recycle. After the solids fall away, the Orbiter and ET will continue, propelled by the main engines.

After about 500 seconds of flight, the main engines will be shut down; soon afterwards, the ET will be released to break up and fall into the ocean. Approximately one or two minutes after main engine cutoff the OMS engines in the Orbiter will be started for a short burst to propel the vehicle to orbit altitude. After the Orbiter has coasted to the desired altitude, it will be leveled using the RCS, and the OMS engines will be fired again to circularize the orbit.

The orbit part of the operation cycle may differ considerably from one mission to another. For some missions, the crew will deploy or retrieve a satellite and return to Earth in one revolution. For other missions, the Orbiter will stay in space as long as 30 days. For the Galileo and Ulysses missions, the orbital part will include releasing the spacecraft and the Centaur upper stage, moving some distance away, staying in orbit for several hours while the crew eats and sleeps, and then preparing to return to Earth. The typical time profile for the Centaur planetary missions from lift-off to cargo deployment is shown in Table C1-2.

Table C1-2 Typical Centaur/Planetary Mission Event Timeline

Event	Time (Hr:Min:Sec)
SSME Ignition	-0:00:06.6
SRB Ignition Command	0:00:00
First Motion/Lift-Off	0:00:00.24
Begin Roll Program (V=125 fps)	0:00:07
End Roll Program (V=300 fps)	0:00:15
Begin First Throttle Down (V=428 fps)	0:00:20
Begin Second Throttle Down (V=716 fps) (Optional)	0:00:31
Begin Throttle Up (V=1599 fps)	0:01:08
SRB Separation	0:02:05
3G Throttling Begins	0:07:39
MECO Command	0:08:34
Zero Thrust	0:08:40
ET Separation	0:08:52
OMS-1 TIG	0:10:34
OMS-1 Cutoff	0:13:22
OMS-2 TIG	0:46:10
OMS-2 Cutoff	0:48:29
Open Payload Bay Doors	1:15:00
IMU Realinement	3:00:00
Meal	4:00:00
Centaur/Spacecraft Checkout	5:00:00
Centaur Deploy	6:40:00
Centaur Engine Ignition	7:20:00

The return part of the operation cycle includes firing the OMS engines to retard the Orbiter out of orbit to glide down for a landing.

Following the completion of orbital operations, the Orbiter is oriented to a tail-first attitude. After the OMS provides the deceleration thrust necessary for deorbiting, the Orbiter is reoriented nose-forward to the proper attitude for entry. The orientation of the Orbiter is established and maintained by Reaction Control System (RCS) so that the atmospheric density is sufficient for the pitch and roll aerodynamic control surfaces to be effective (about 250,000 feet altitude and 26,000 ft/s velocity). The yaw RCS remains active until the vehicle reaches an angle of attack of about 10 degrees (about 80,000 feet altitude).

The Orbiter entry trajectory provides lateral flight range to the landing site and energy management for an unpowered landing. The trajectory, lateral range, and heating are controlled through the attitude of the vehicle by angle of attack and bank angle. The angle of attack is established at 38 degrees for the theoretical entry interface of 400,000 feet altitude. The entry flightpath angle is -1.19 degrees. The 38 degree attitude is held until the speed is reduced to 21,200 ft/s (about 220,000 feet altitude), is then reduced gradually to 28 degrees at 17,200 ft/s (about 190,000 feet altitude); it is held at 28 degrees until speed is reduced to 8500 ft/s (about 150,000 feet altitude) and then reduced gradually to 6 degrees where the speed is about 1500 ft/s (about 70,000 feet altitude) at the beginning of Terminal Area Energy Management (TAEM).

During the final phases of descent, flightpath control is maintained by using the aerodynamic surfaces. TAEM is initiated to provide the proper vehicle approach to the runway with respect to position, energy, and heading. Final touchdown occurs at an angle of attack of about 16 degrees. The maximum landing speed for a 32,000-pound payload, including dispersions for hot day effects and tailwinds, is about 207 knots.

The Shuttle will not be launched if electrical fields at the launch site exceed one kilovolt per meter or if its flightpath will take it into dangerous weather. On the launch pad, the Shuttle is within the "cone of protection" of the lightning protection system. This system consists of a fiberglass mast with cable threaded through the top to ground points 1,000 feet away on either side.

Lightning protection is provided by the launch site until lift-off. Thereafter, the bare 20-inch-long, 20 degree nose cone at the tip of the ET nose cap serves as a lightning rod. The ET design incorporates features to protect the structure and subsystems from the direct and indirect effects of triggered atmospheric electrical discharges during flight operations. The ET is designed to function after an initial strike of 200,000 amperes peak at the ET lightning rod and a second lightning strike of 50,000 ampere peak across the ET body while it is in motion.

C1.3.2 Orbiter Safety

This section describes plans to maintain Orbiter and cargo safety. The Orbiter has been designed to recover safely from many failures. The crew can fly the Orbiter back to Earth after certain types of failures which would not be possible for a conventional expendable launch vehicle. A basic requirement for the Shuttle has been for it to withstand a failure and yet be just as operational as if the failure had not occurred. It should also be able to tolerate a second failure in a flight-critical system and still return safely to Earth.

The Space Shuttle Vehicle is required to have an intact abort capability for contingencies arising from certain specific failures which might occur during the powered ascent flight phase (lift-off to post-MECO OMS insertion). Intact abort is defined as safely returning the Orbiter, crew, and cargo to a suitable landing site. The specific failures which will result in an intact abort are the following:

a. Complete or partial loss of thrust from one SSME;

b. Loss of thrust from one OMS engine;

c. Loss of two electrical power buses;

d. Failure of two auxiliary power units;

e. Failure of some life support equipment.

Four basic abort modes have been developed to provide continuous intact abort capability for Centaur missions during the ascent phase: Return-To-Launch-Site (RTLS), Transatlantic Abort Landing (TAL), Abort-Once-Around (AOA), and Abort-To-Orbit (ATO). The four modes are available during different segments of the ascent flight to provide intact abort capability. Two modes, AOA and ATO, are available after MECO in the event of an OMS engine failure. Figure C1-29 is an altitude/range profile showing the relationship between RTLS, TAL, and AOA aborts. Figures C1-30 and C1-31 show the flight profiles of an AOA and an ATO, respectively. It should be noted that for purposes of RTG concerns, both the LOX and LH_2 dump system (valves, lines, etc) must fail to dump during any abort before an explosive environment can exist with the Centaur. In other words, a failure which prevents the successful dumping of only LOX or LH_2 is in itself not a threat to the RTGs.

The return-to-launch-site (RTLS) abort mode will be used in event of an SSME failure occurring between lift-off and the end of the predetermined RTLS interval. However, when an overlap occurs between successive abort mode intervals, either abort mode could be selected depending upon abort failure conditions. The basic RTLS abort flight mode may be stated as follows. The RTLS abort mode is selected after separation from the SRBs (second stage flight) even though the SSME failure might have occurred in first stage flight; the vehicle will continue accelerating downrange with the two remaining SSMEs until the MPS propellant equals the amount required to reverse the direction of flight and propel the vehicle to the RTLS MECO target. The RTLS MECO target is selected to provide acceptable Orbiter/ET separation conditions, acceptable ET impact location, and acceptable range to permit gliding safely back to the selected landing site, whether it be at the launch site or at the weather alternate site. Typical earliest (lift-off) and latest RTLS abort sequence of events are defined in Table C1-3.

In the event of an SSME failure on a non-Centaur mission, the Trans-Atlantic-landing (TAL) abort mode will be used to fill the abort gap that normally exists between the "last RTLS" and the earliest Press-To-MECO. However, on aborted Shuttle/Centaur missions, the Centaur propellant dump significantly reduces the payload weight, improving the Shuttle vehicle's thrust-to-weight performance significantly--enough to allow an overlap of the RTLS and PTM window. Therefore, the TAL abort mode is not required for an SSME failure and would only be used for a systems failure requiring the Orbiter either to land in the shortest possible time (for example, crew cabin gas leak) or to avoid orbit insertion because of a loss of deorbit capability.

The basic TAL abort flight mode may be stated as follows: After selection of the TAL abort mode (TAL abort mode is acquired pre-MECO by selecting an AOA on the abort select switch), the vehicle will accelerate downrange to the TAL MECO targets. At abort initiation the OMS propellant and Centaur propellant dumps are initiated and run to completion, if time permits, to achieve the correct landing weight and center-of-gravity for entry. At 90 seconds before MECO the vehicle rolls to the head-up ET separation attitude. After ET separation and after a 30-second MPS dump interval, the onboard computers are loaded with the entry operational flight software. The Orbiter then glides to the primary landing site or the weather alternate landing site. Landing sites change as a function of intended orbit inclination. For example, for orbit inclinations near 28 degrees, Dakar, Senegal (on the west coast of Africa) is the primary landing site. For inclinations near 57 degrees, Zaragoza, Spain is currently the primary landing site. The weather alternate landing site for both of these inclinations is Moron, Spain. A typical TAL abort sequence of events is defined in Table C1-4.

Table C1-3 RTLS Abort Sequence of Events for an Earliest and Latest SSME Failure

Mission Elapsed Time (Sec)		Event
Earliest	Latest	
0	0	Lift-off of the Space Shuttle Vehicle
1	240	SSME Failure, Engine Out Pitch Biasing Guidance Engaged
125	125	SRB Separation
150	255	RTLS Abort Selected
150	255	Centaur Propellant Dump Initiated
424	255	Power Pitch Around Initiated
424	255	OMS Propellant Dump Initiated
400	505	Centaur Propellant Dump Complete
586	425	OMS Propellant Dump Completed
704	579	Power Pitch Down
721	594	MECO
734	608	ET Separation
745	620	MPS Dump Initiation
765	640	Aft RCS Propellant Dump Initiation
900	775	Aft RCS Propellant Dump Completed
865	740	MPS Dump Completed
940	800	Mach 3.5
1400	1260	Landing

Table C1-4 TAL Abort Sequence of Events for an Earliest and Latest "Systems TAL" Abort

Mission Elapsed Time (Sec)		Event
Earliest	Latest*	
0	0	Lift-off of the Space Shuttle Vehicle
125	125	SRB Separation
240	355	"Systems TAL" Abort Selected
240	355	OMS Propellant Dump Initiated
240	355	Centaur Propellant Dump Initiated
480	415	Vehicle Roll-To-Heads Up Attitude
435	510	OMS Propellant Dump Completed
490	510	Centaur Propellant Dump Completed
570	520	MECO
588	538	ET Separation
600	550	+X ARCS Thruster Firing Initiated (Manual)
605	555	+X ARCS Stop, MPS Dump Initiated (Manual)
635	585	MPS Dump Stop (Auto)
685	610	Entry Flight Software Memory Load
1650	1600	Mach 3.5
2100	2050	Landing

*The latest TAL (MET=355 seconds) is defines as one which permits Centaur residuals of 10,000 lbs at MECO (LH_2 = 2800 lbs, LOX = 7220 lbs). There are also 2500 lbs of OMS propellant remaining. The last TAL abort opportunity is at 425 seconds. However, excessive Centaur and OMS residuals remain at MECO (LH_2 = 5980 lbs, LOX = 27,650; OMS = 13,000 lbs). A post-MECO Centaur and OMS propellant dump is mandatory to eliminate the hydrogen (LH_2) residuals prior to landing.

Press-To-MECO (PTM) is not a specific flight mode but rather a decision to continue to the nominal guided Main Engine Cut-Off (MECO) target point even though there has been a SSME failure. When the abort modes were originally conceived, an AOA was used between the last RTLS capability and the earliest ATO. The ATO was used between the last AOA and the earliest availability of mission continuation with only two SSMEs operating. Ascent performance analyses results have shown that if the ascent performance dispersions were in a favorable direction (a "good day"), then an ATO abort selected pre-MECO, based on the defined abort mode boundaries, could have been a mission completion had the ATO abort not been selected. Similarly, some AOAs could have been ATOs. Consequently, AOA and ATO aborts are now selected post-MECO only, based on the state vector achieved compared with the MECO target line.

However, for Shuttle/Centaur missions, if an SSME fails between the last RTLS mode boundary and the Centaur mission completion boundary, an ATO abort will be selected pre-MECO. ATO selection provides for auto initiation of both the OMS propellant dump and Centaur propellant dump modes. The vehicle thrust-to-weight increase that ensues enables avoidance of a TAL abort. Post-MECO evaluation of the vehicle state vector determines whether an AOA or ATO will be selected. If an AOA is selected, the Orbiter is oriented to the required attitude, and an OMS burn propels the Orbiter to the OMS-1 apogee target. An OMS-2 deorbit burn will enable the orbiter to achieve the entry interface target that is acceptable for the Orbiter glide to the landing site.

The Space Shuttle Vehicle is not required to have "contingency abort" capability. The goal of a contingency abort is to save the crew of the Orbiter. The safe return of the Orbiter or its cargo would be unlikely since it would usually be ditched in the ocean or crash landed. The selected failures which will result in a contingency abort are the following: loss of thrust from two or three SSMEs, failure of the TVC of a main engine or an SRB, premature separation of the Orbiter from the ET, or failure to separate an SRB from the ET.

Specific flight procedures have been developed to maximize the crew survival chances for multiple SSME failures from lift-off to MECO or single SSME completion boundaries. However, a contingency abort during a mission with a Centaur stage in the cargo bay will probably rupture the Centaur tanks at ditching or crash landing and therefore can cause an explosion.

This paragraph lists certain types of failures that would preclude either an intact or a contingency abort. The types of failures that lead to a loss of critical function are aft compartment explosion, rupture or explosion of the ET, burn-through of an SRB, failure of major structure, complete loss of guidance or control, failure of one SRB to ignite, loss of thrust from either SRB, hardover condition of a main engine or an SRB, failure to separate the Orbiter from the ET, failure of the nozzle of an SRB, or premature separation of an SRB from the ET.

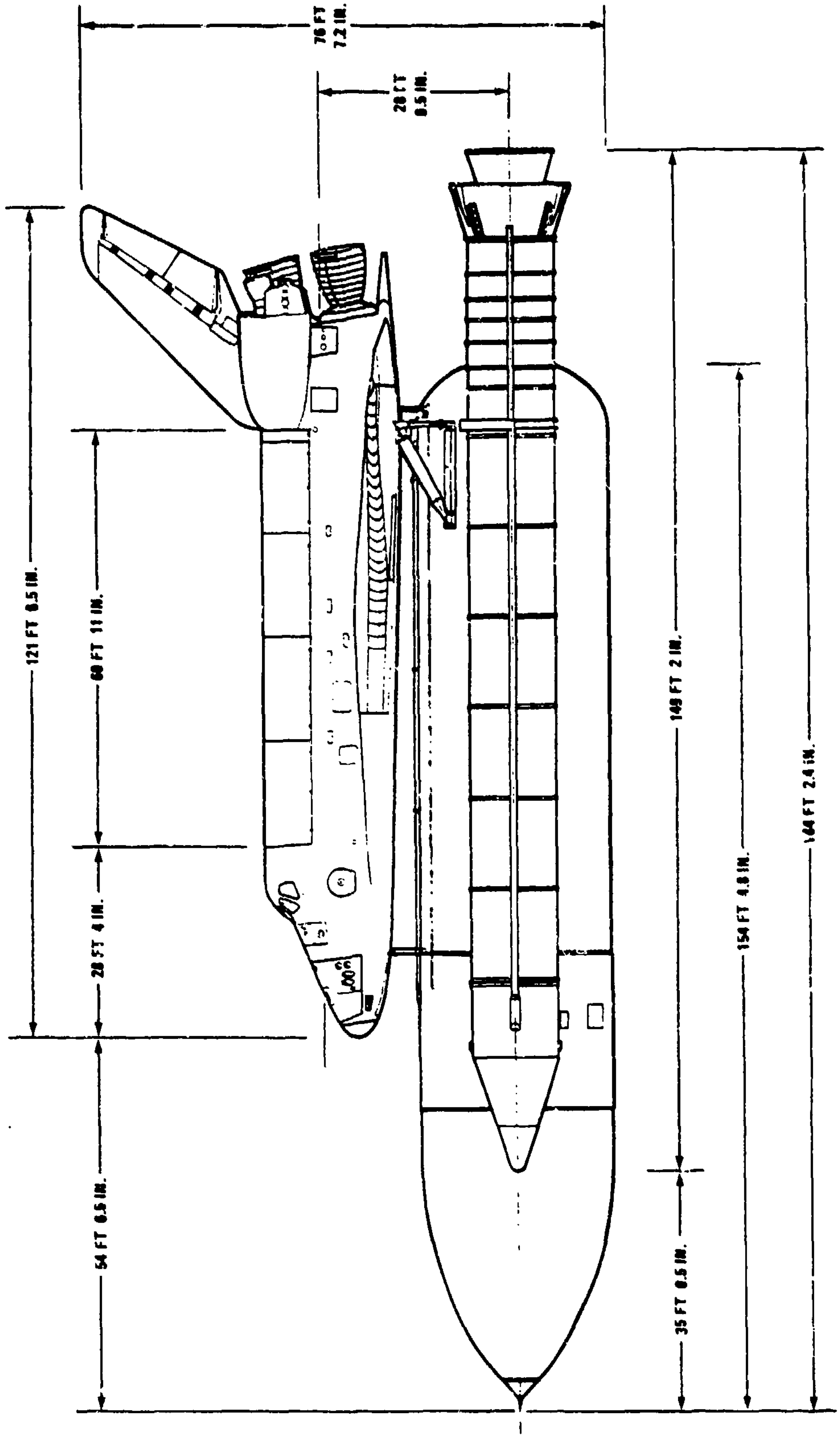

Figure C1-1 Shuttle Vehicle, Side View

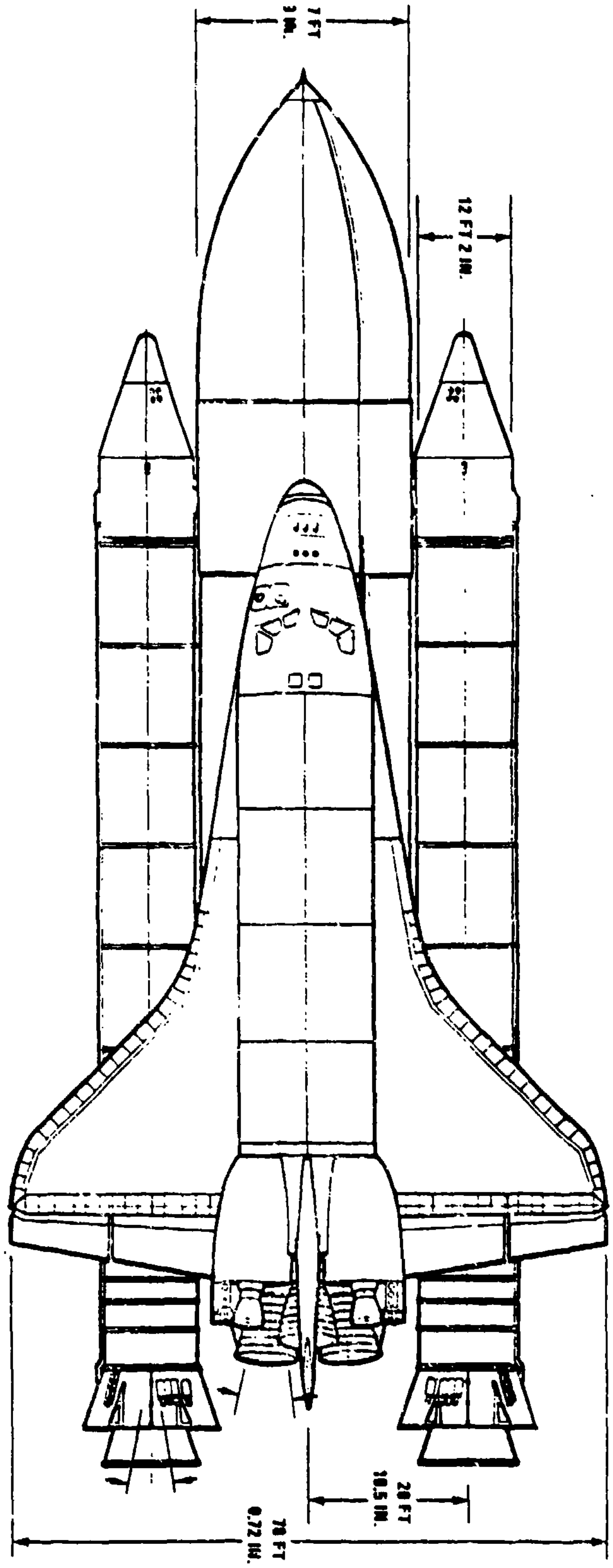

Figure C1-2 Shuttle Vehicle, Top View

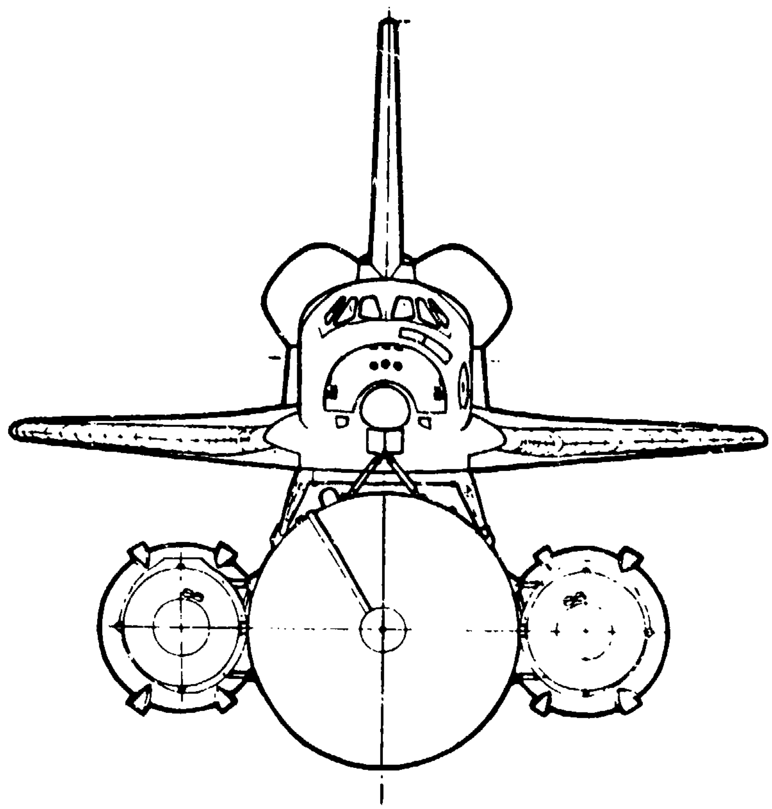

Figure C1-3 Shuttle Vehicle, Front View

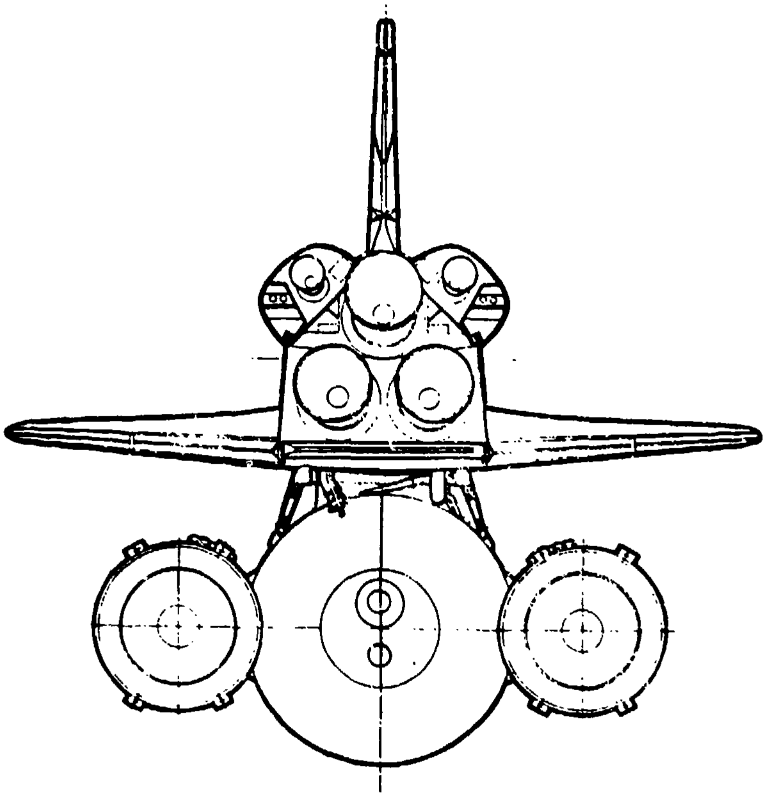

Figure C1-4 Shuttle Vehicle, Back View

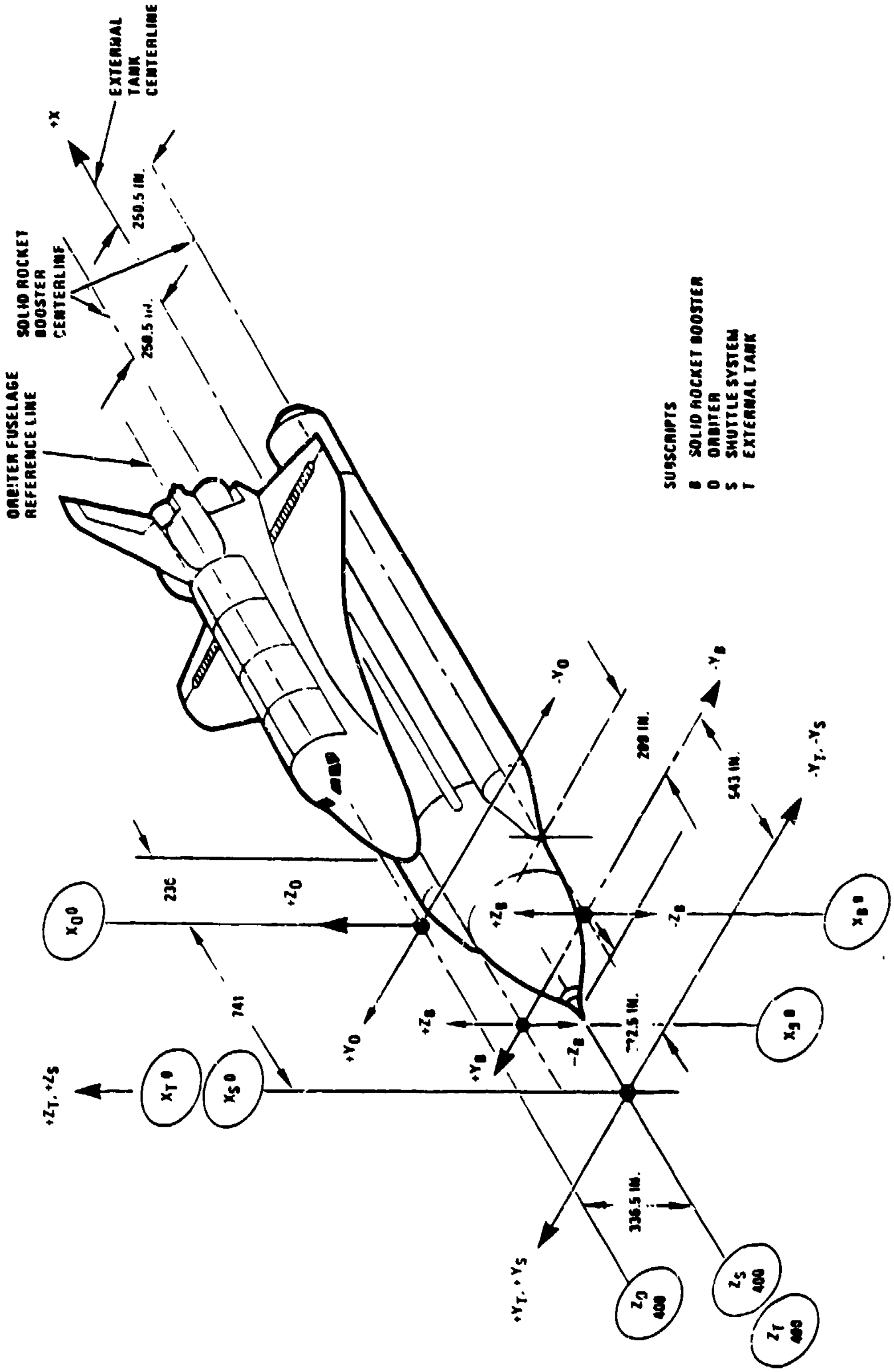

Figure C1-5 Shuttle Coordinate Systems

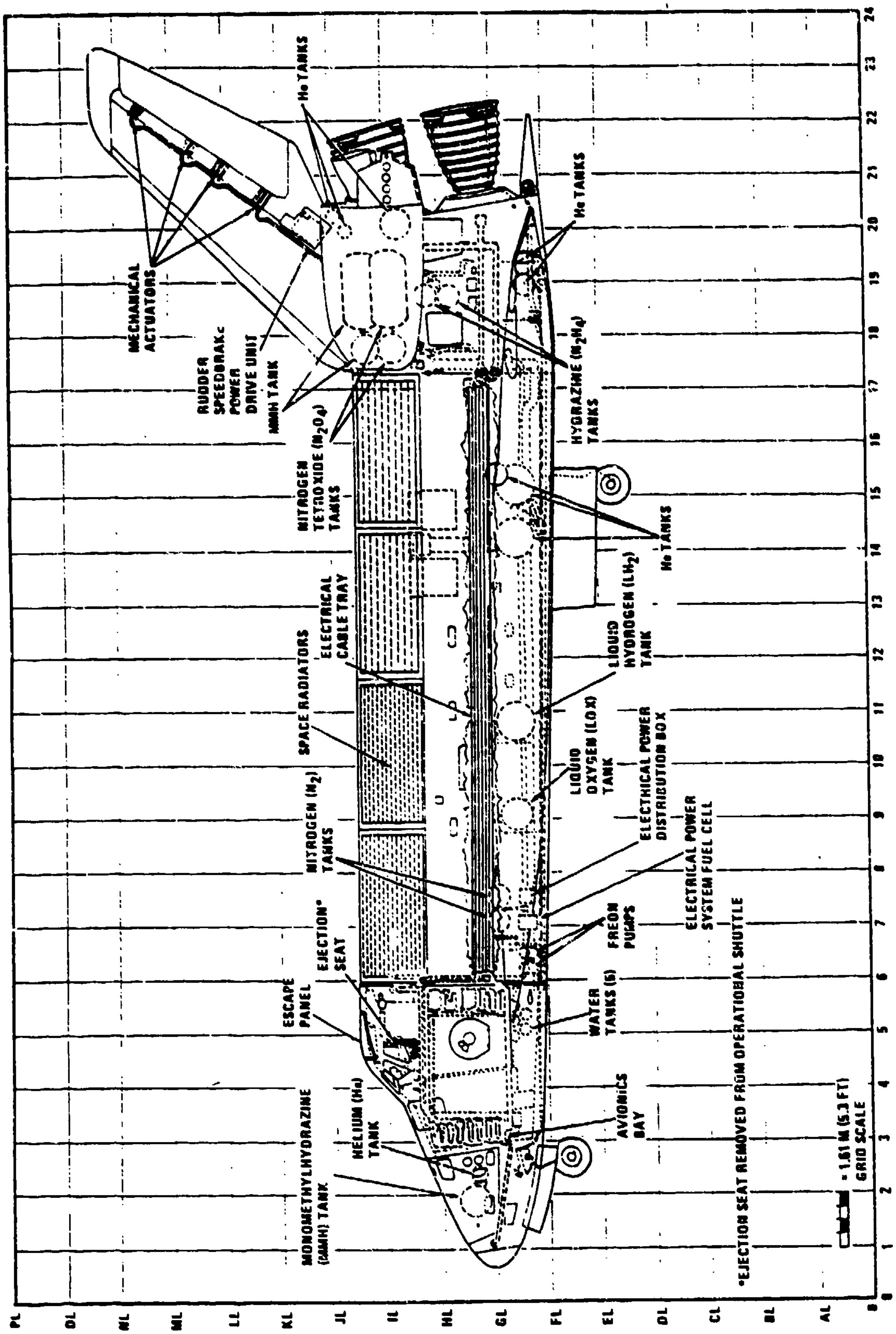

Figure C1-6 Layout of Orbiter, Left-Hand View

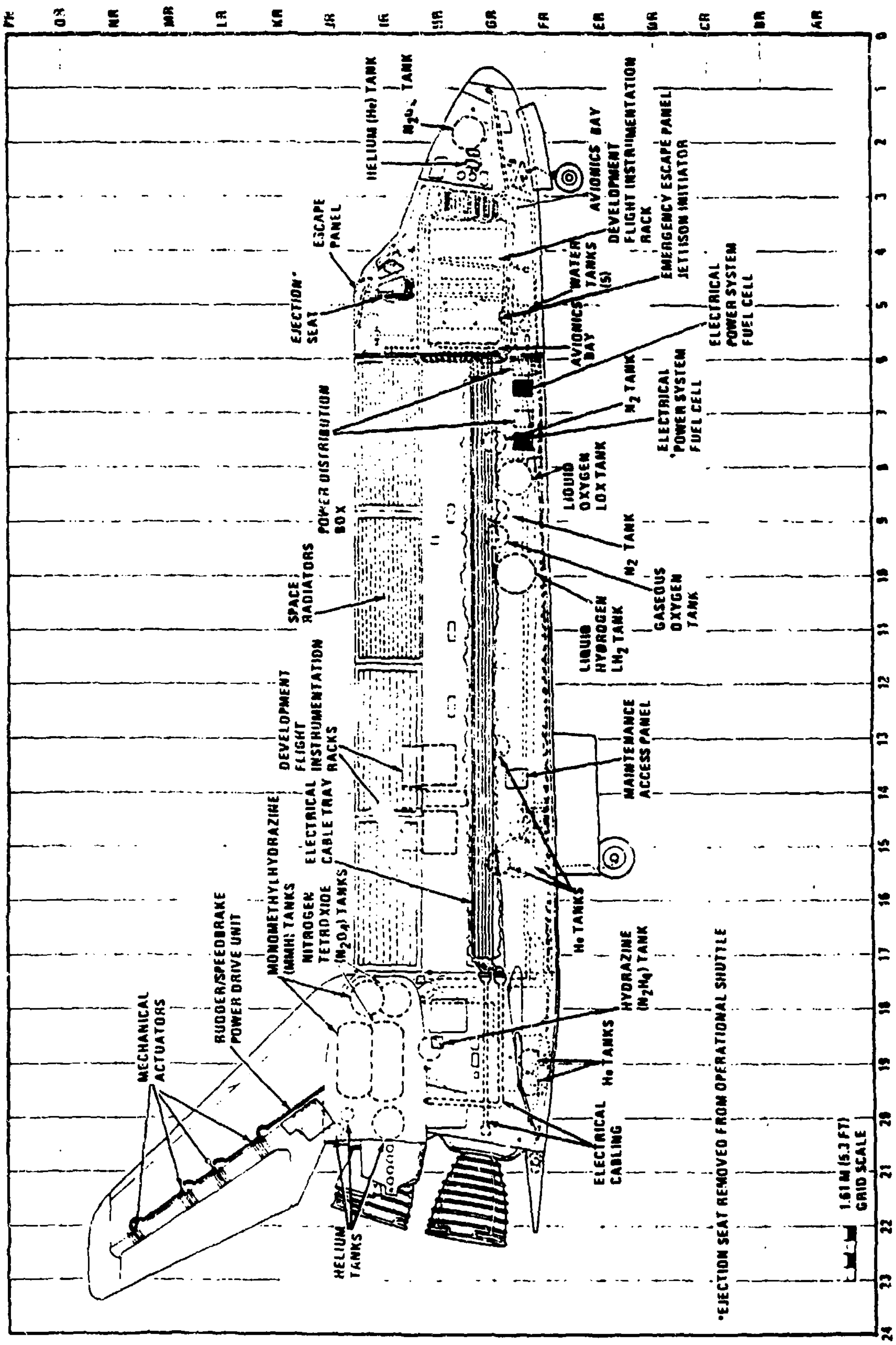

Figure C1-7 Layout of Orbiter, Right-Hand View

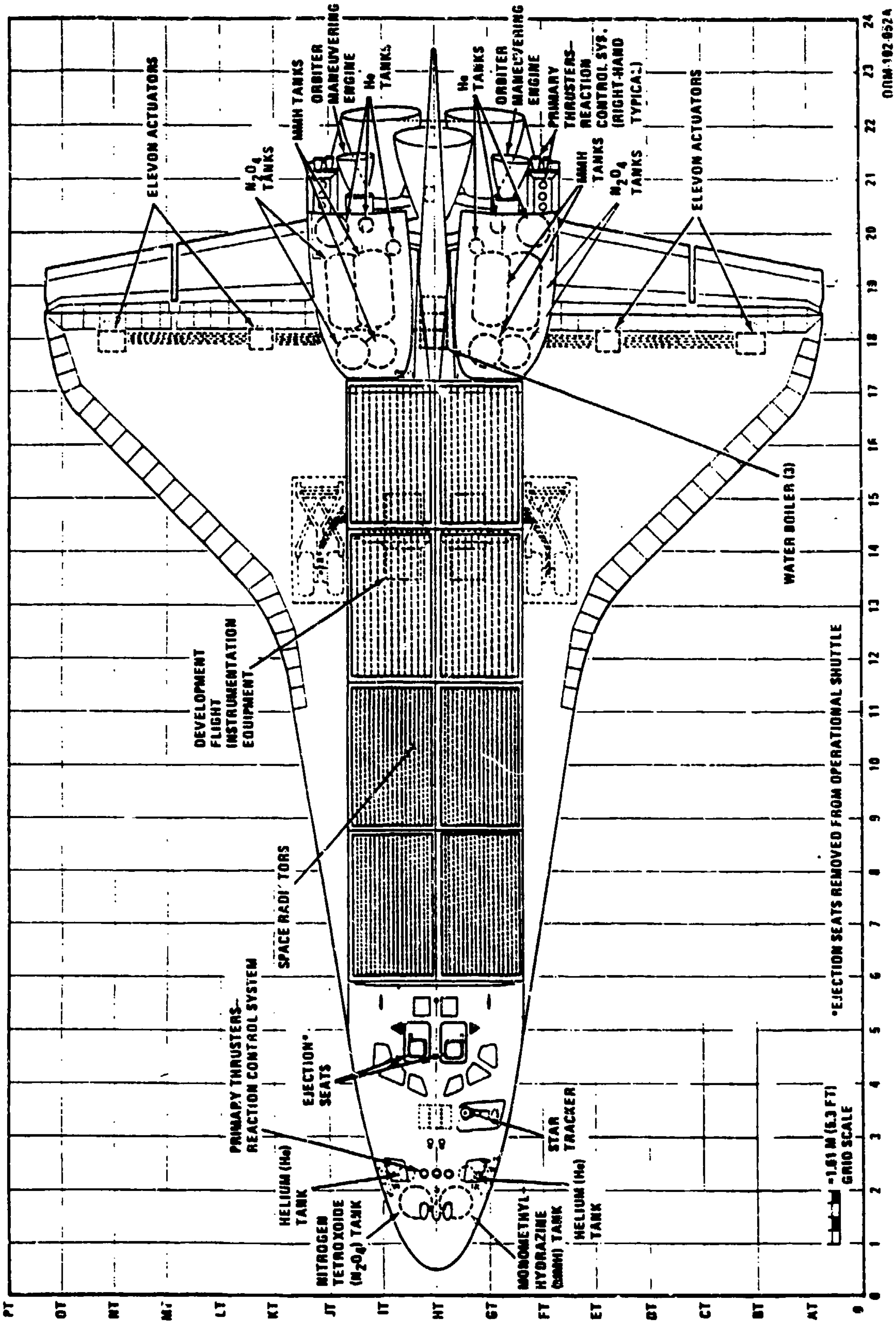

Figure C1-8 Layout of Orbiter, Top View

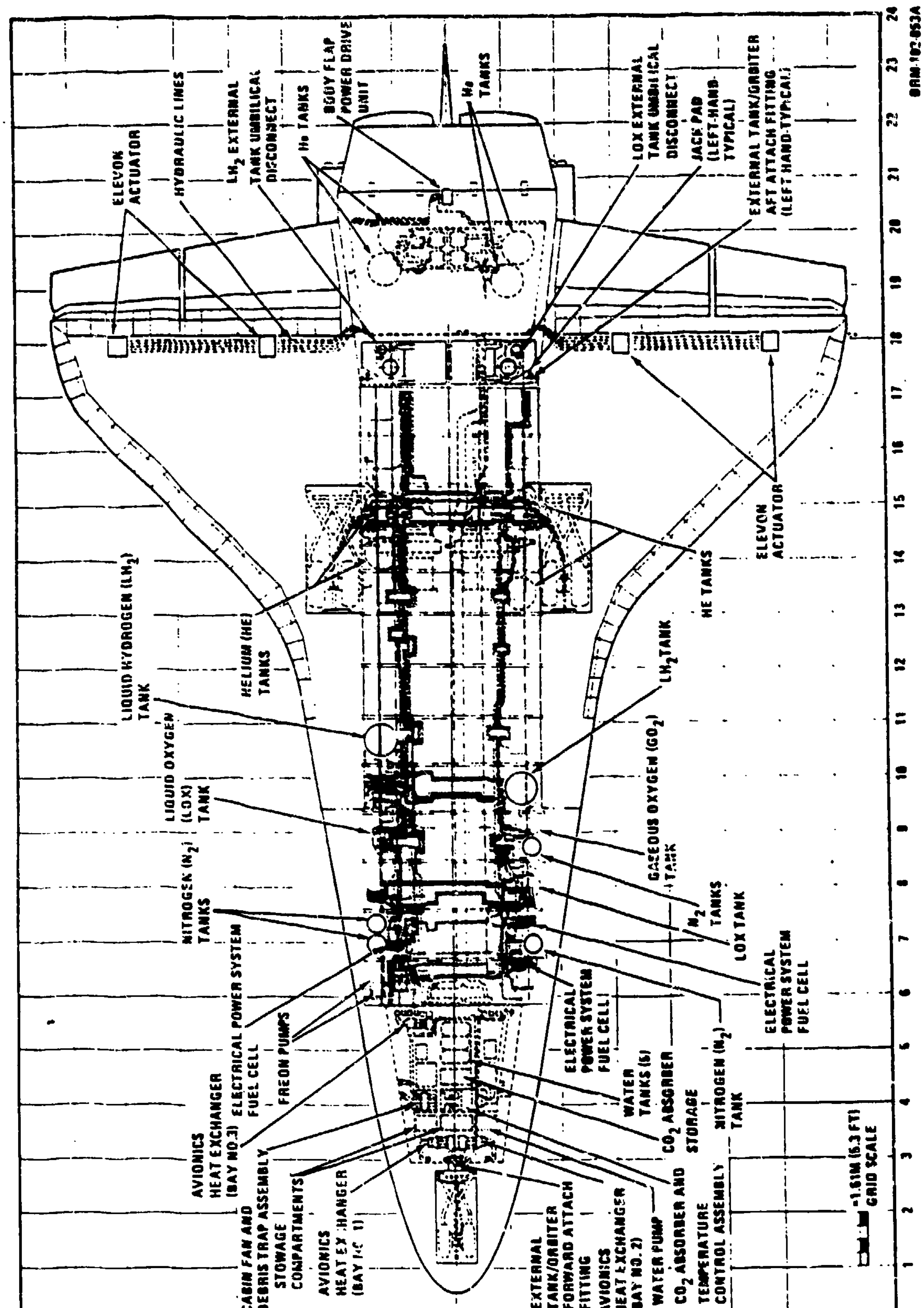

Figure C1-9 Layout of Orbiter, Bottom View

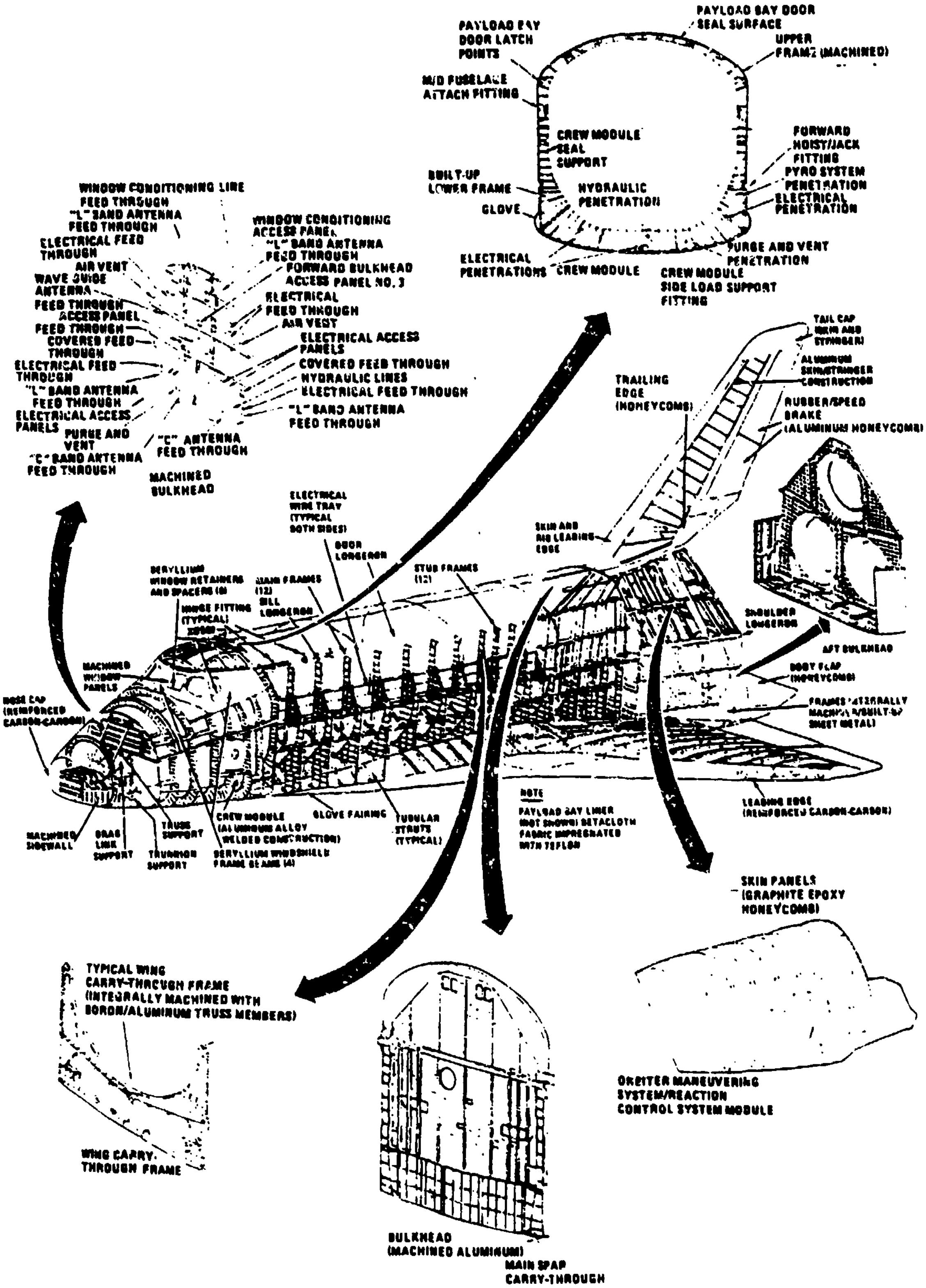

Figure C1-10 Orbiter Structure, Perspective View

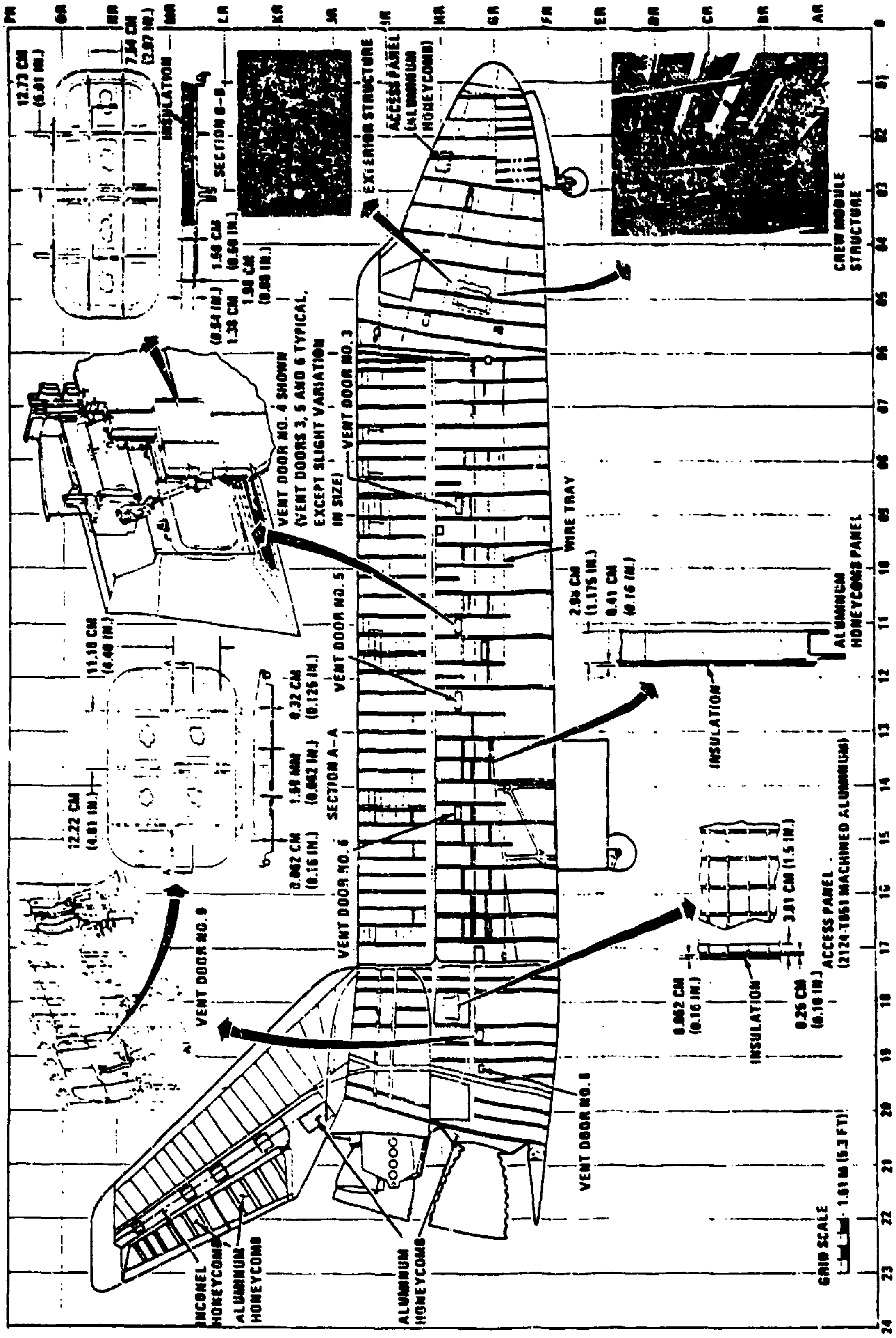

Figure C1-11 Orbiter Structure, Side View

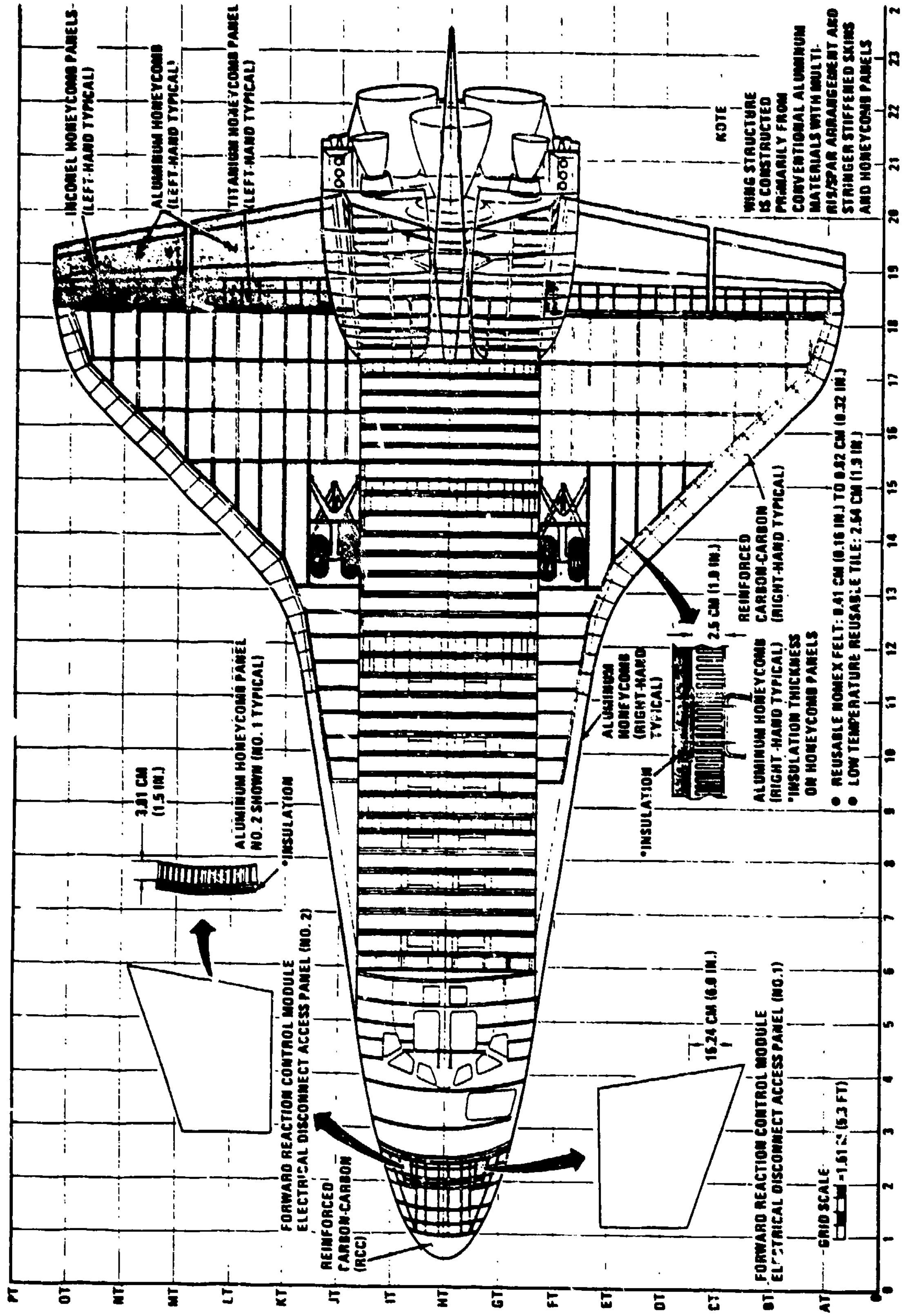

Figure C1-12 Orbiter Structure, Top View

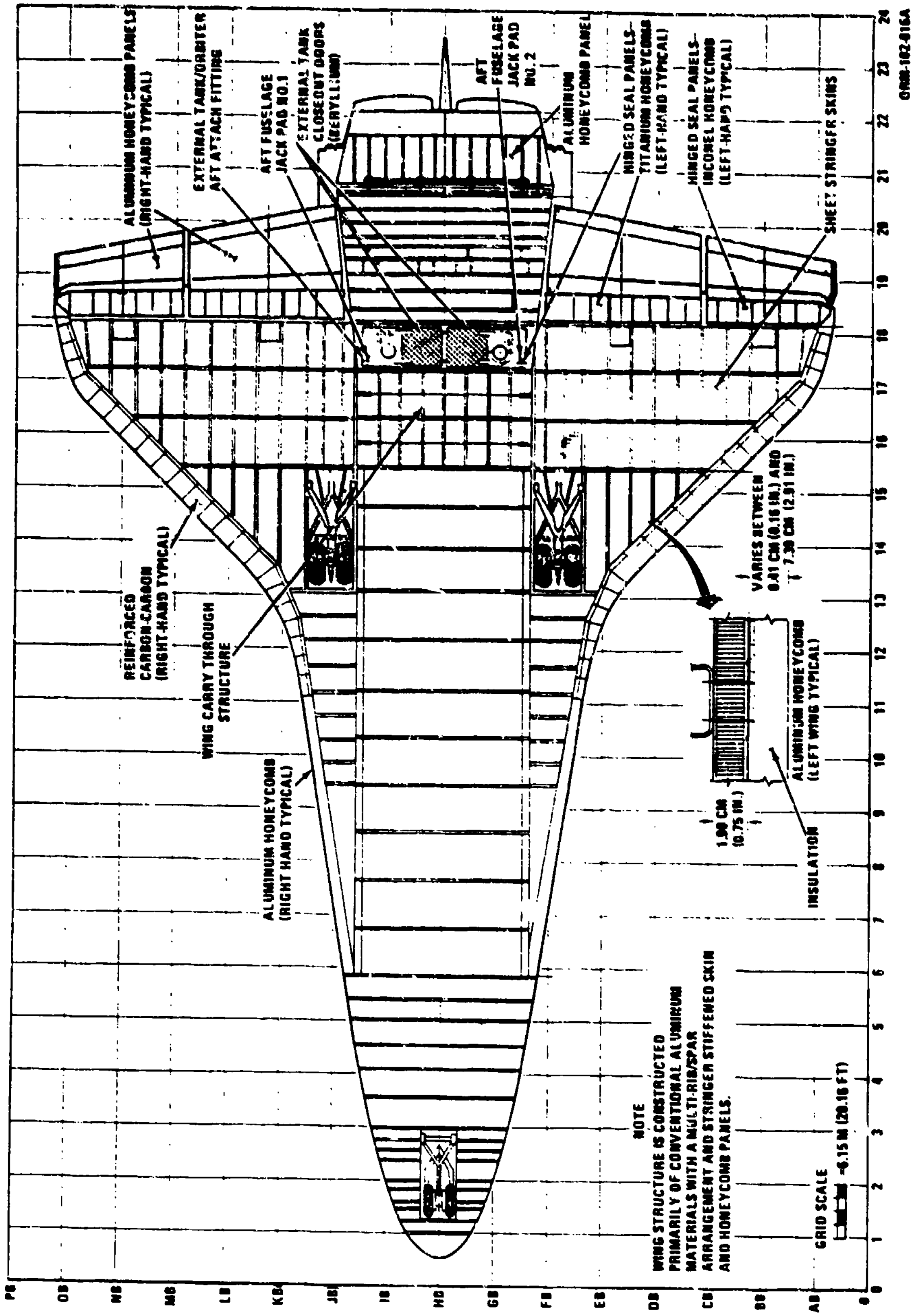

Figure C1-13 Orbiter Structure, Bottom View

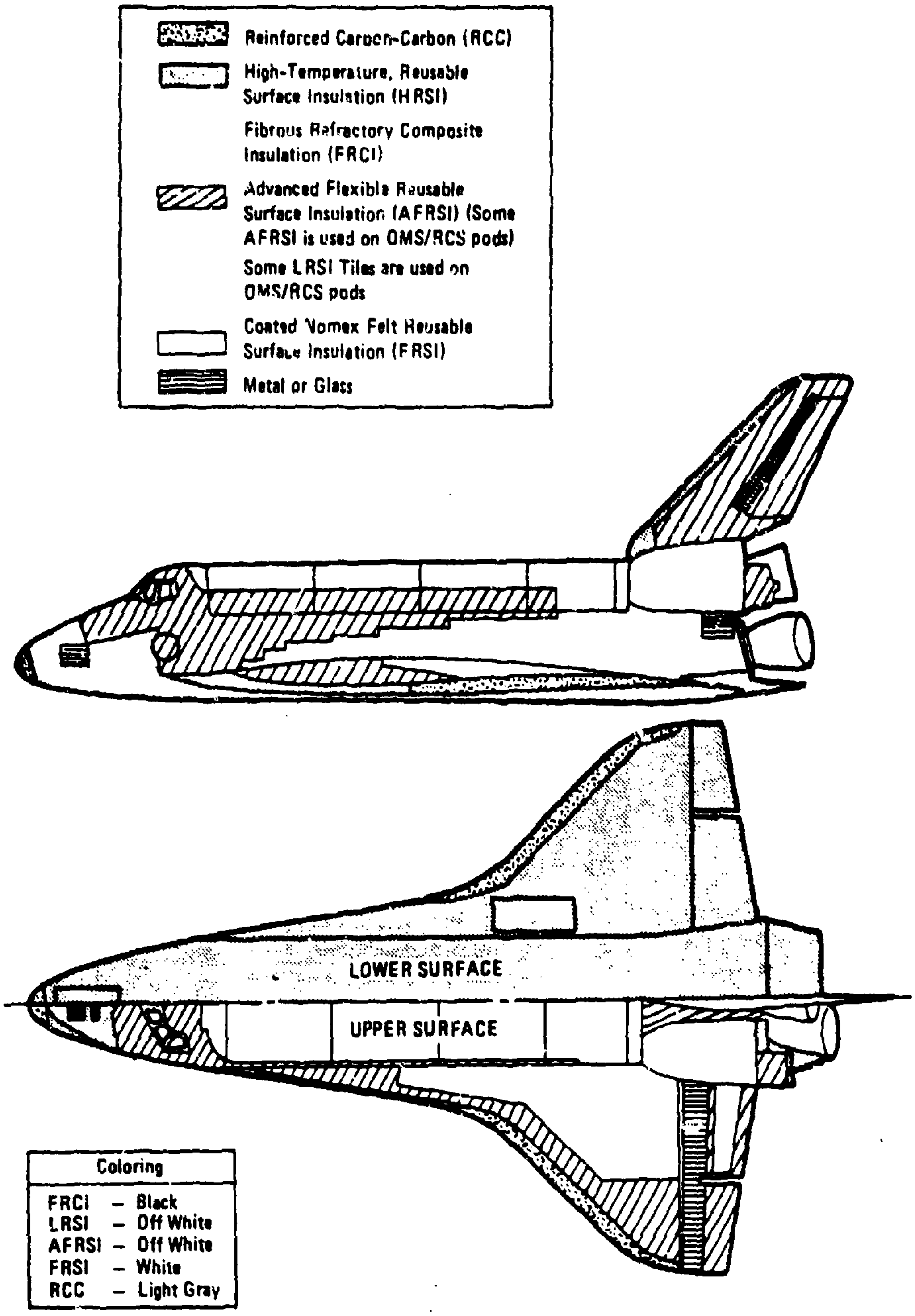

Figure C1-14 Thermal Protection System, Orbiter 103 and Subsequent Orbiters

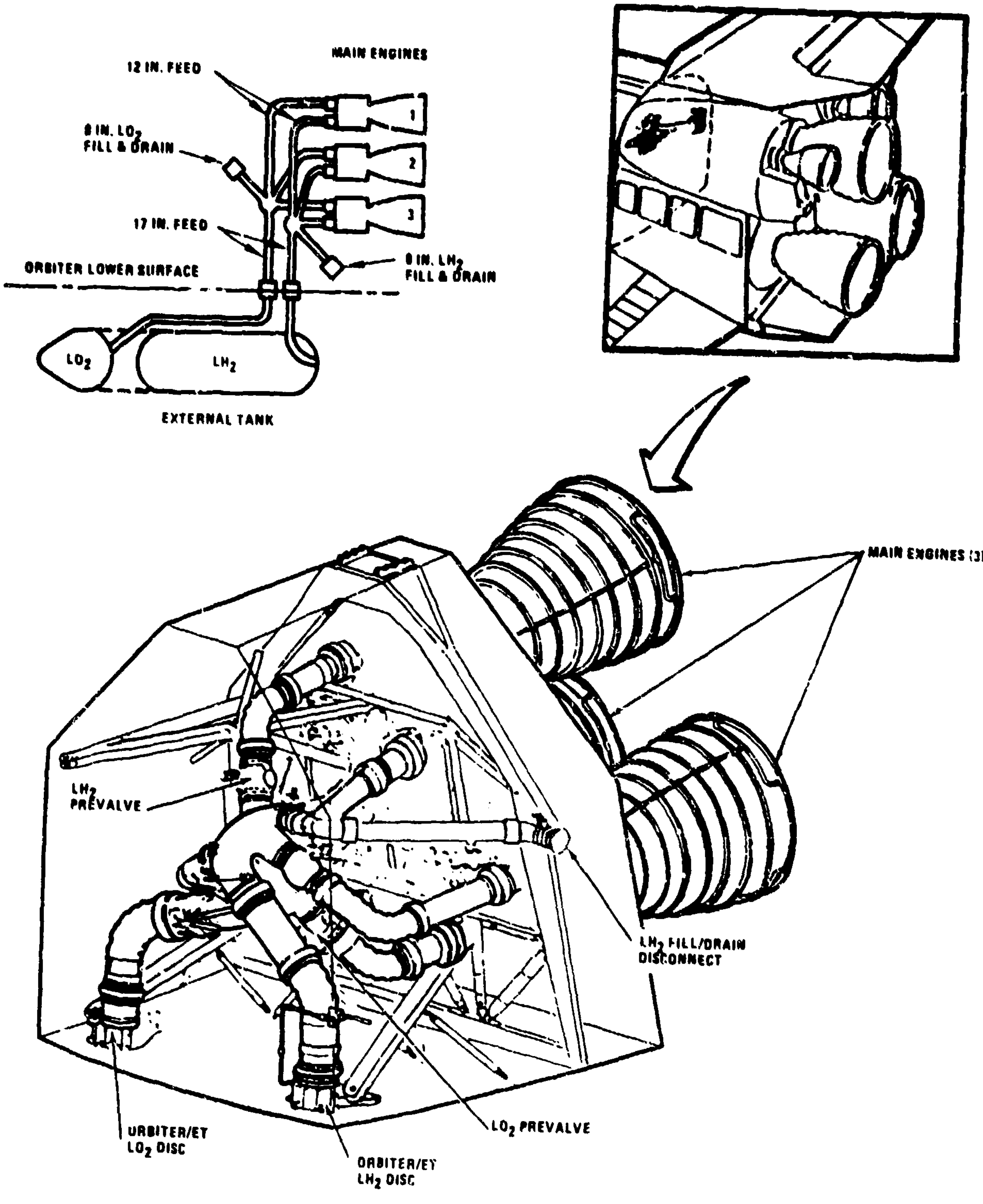

Figure C1-15 Space Shuttle Main Engines

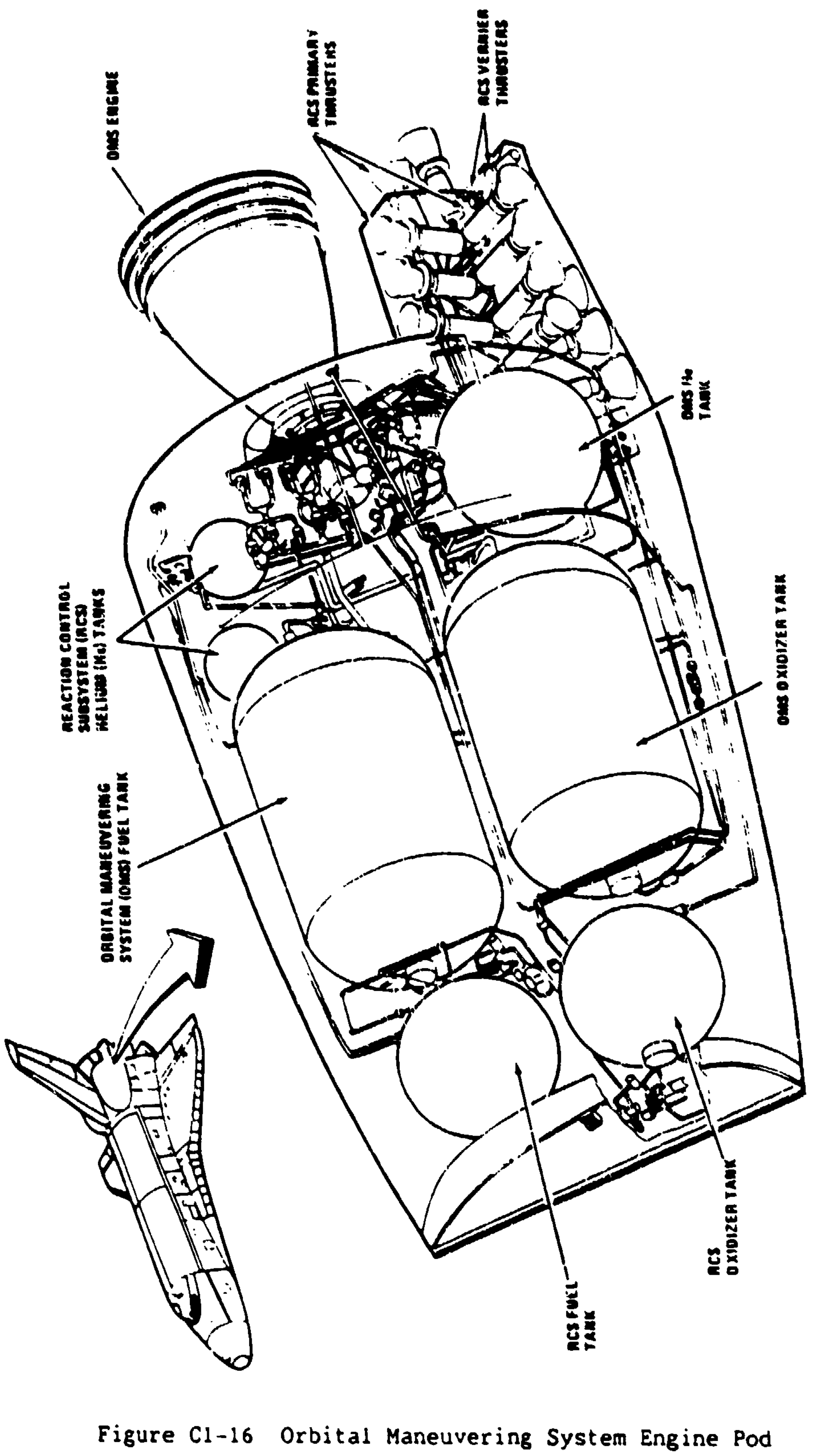

Figure C1-16 Orbital Maneuvering System Engine Pod

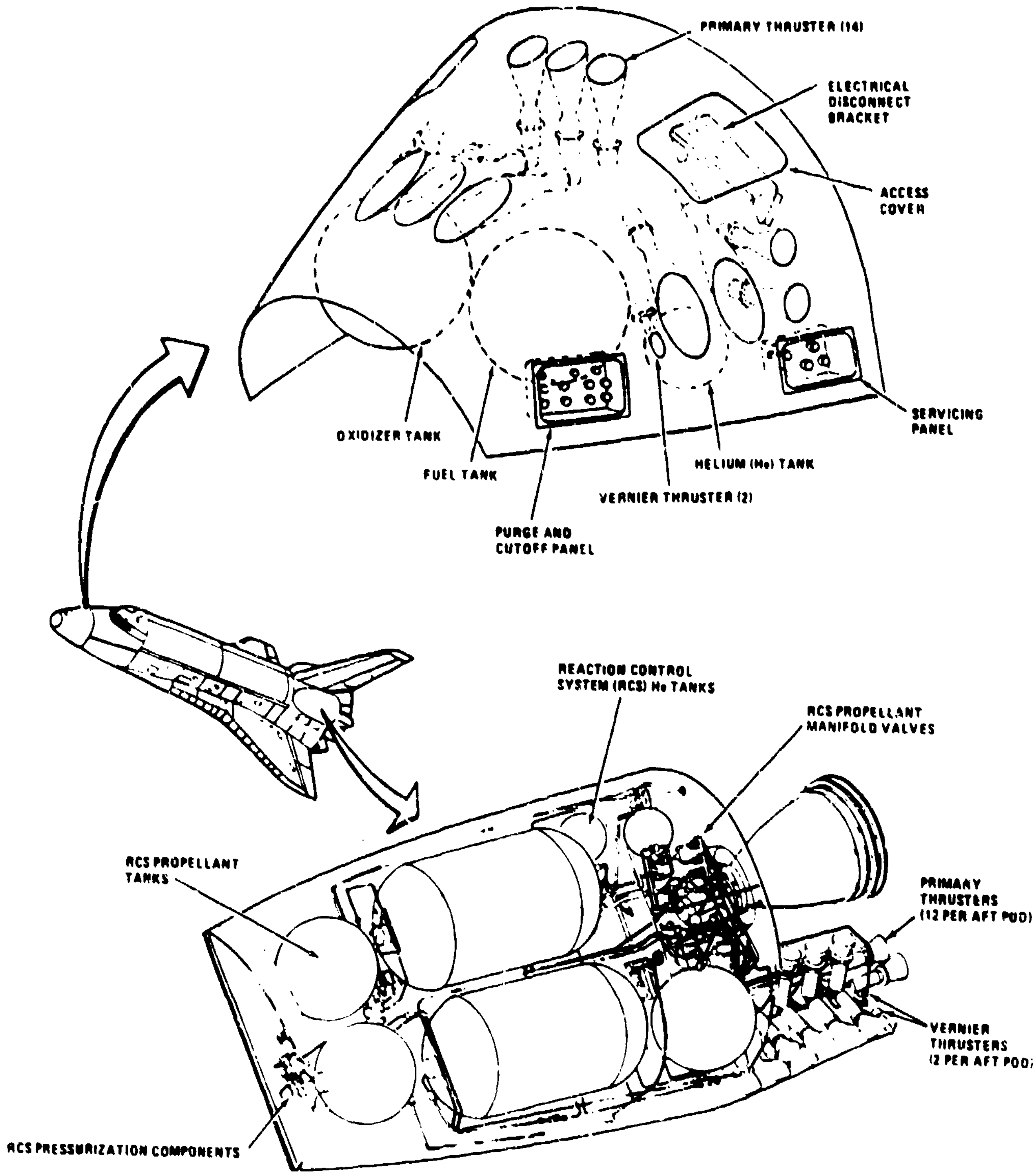

Figure C1-17 Orbiter RCS

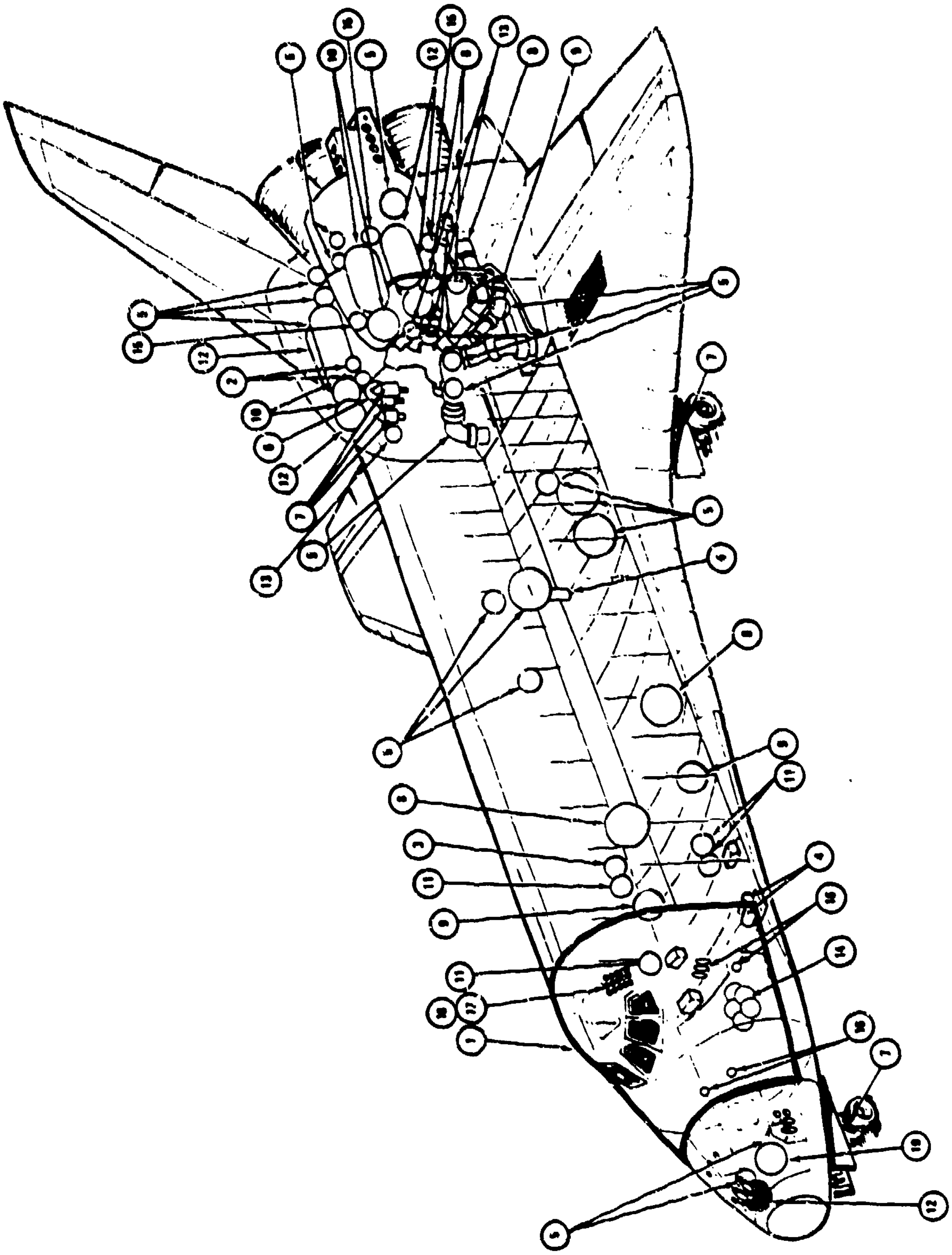

Figure C1-18 Pressure Tanks in Orbiter

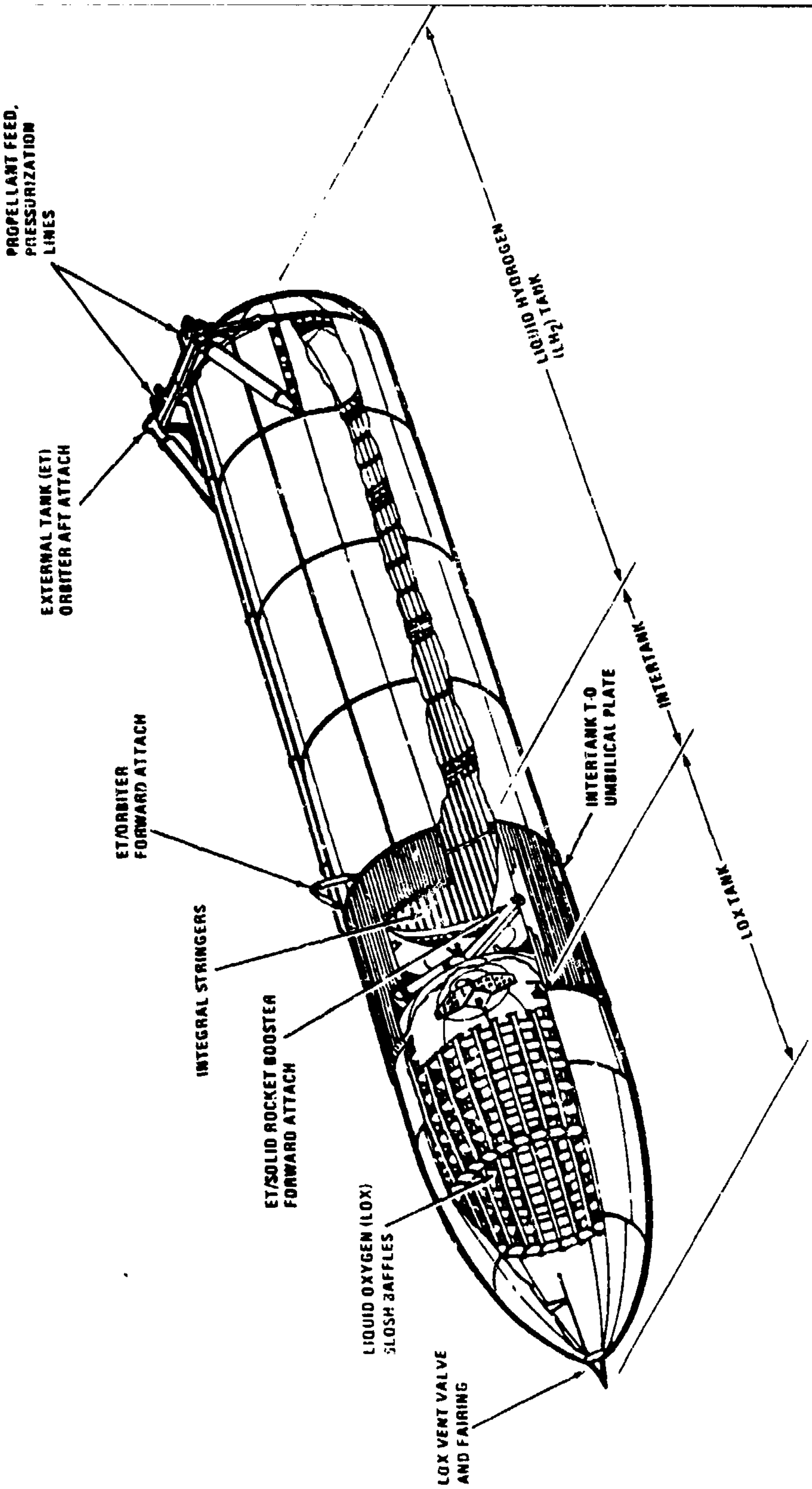

Figure C1-19 Shuttle External Tank

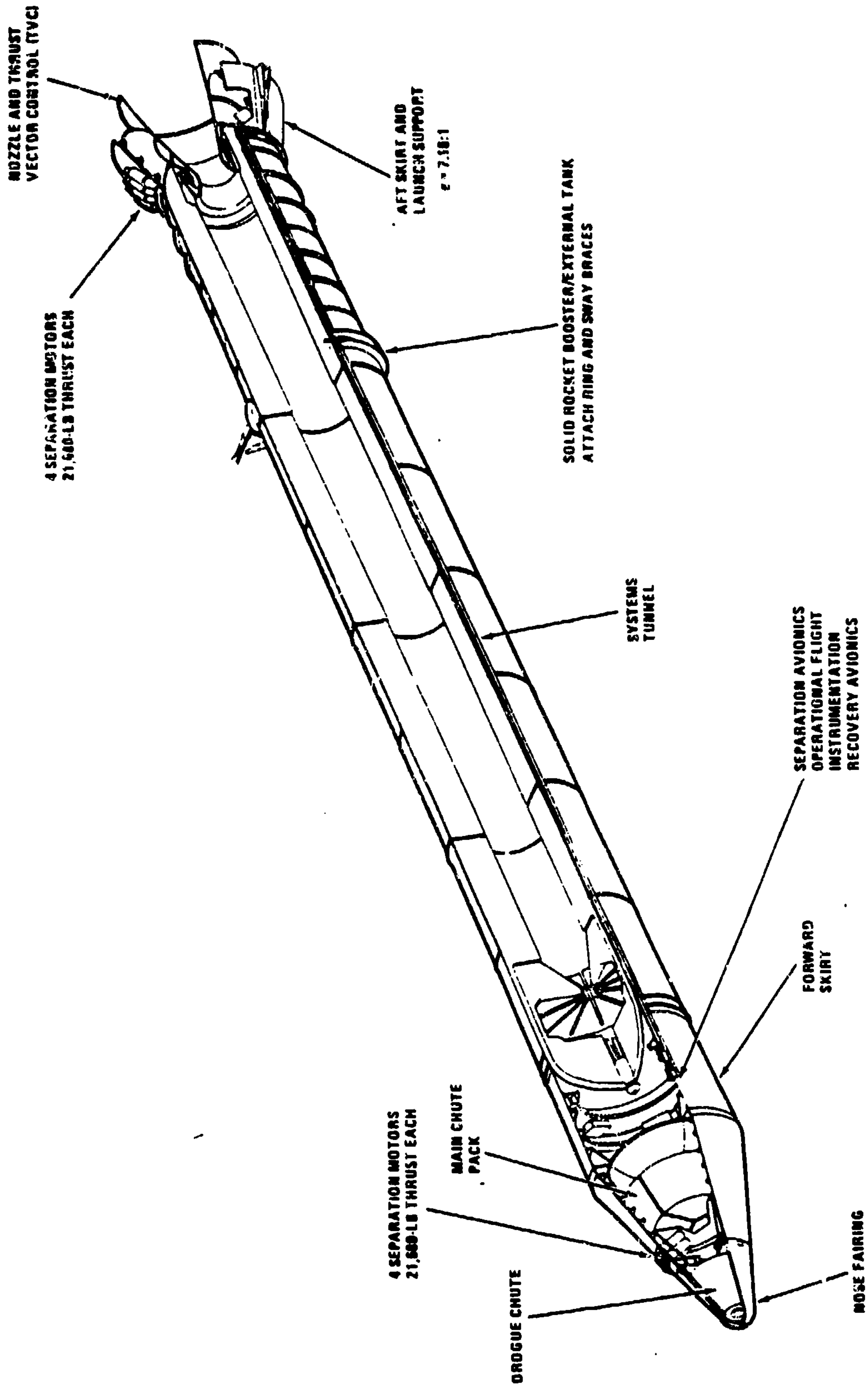

Figure C1-20 Shuttle Solid Rocket Motors

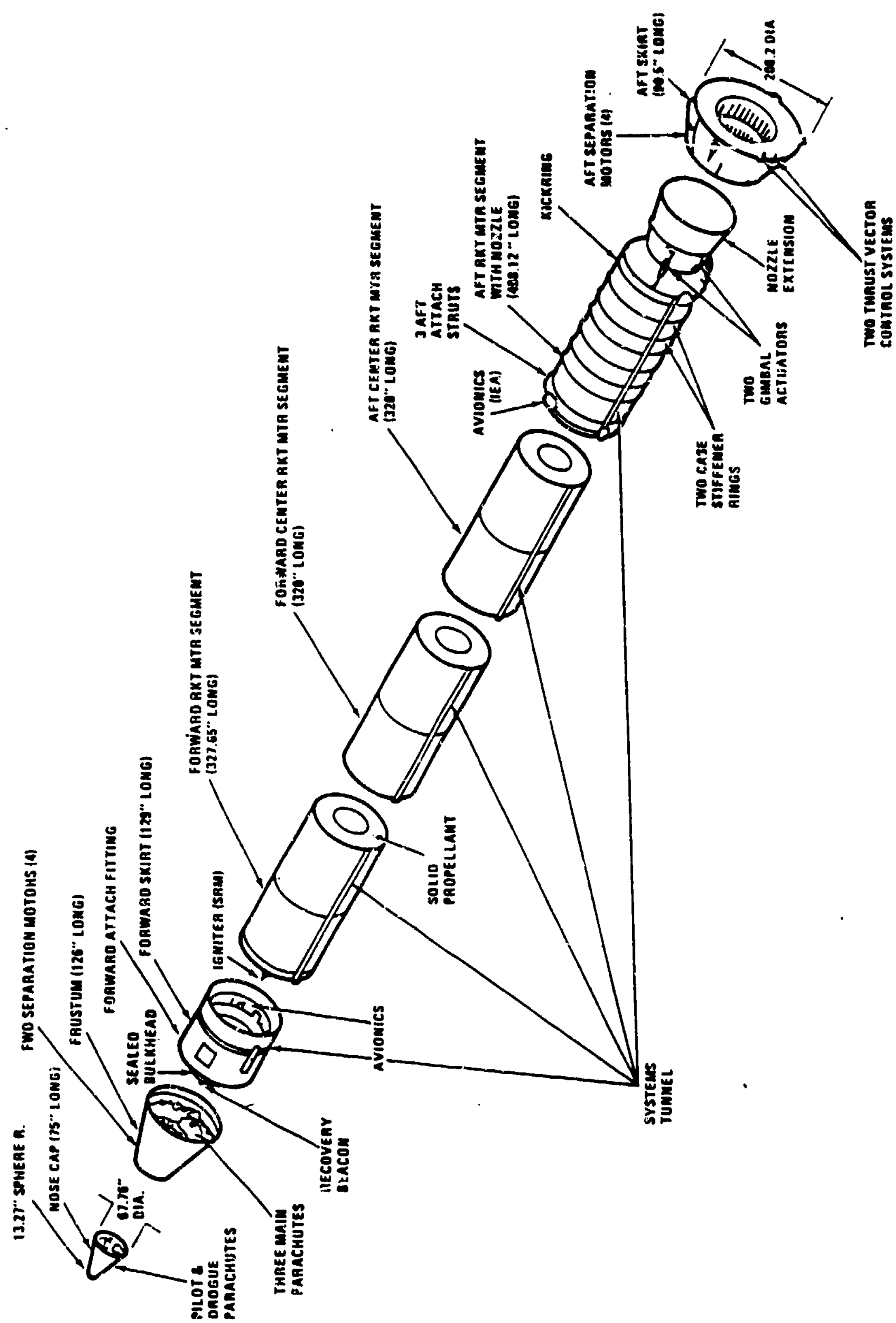

Figure C1-21 Solid Rocket Motor Parts - Overall Structure

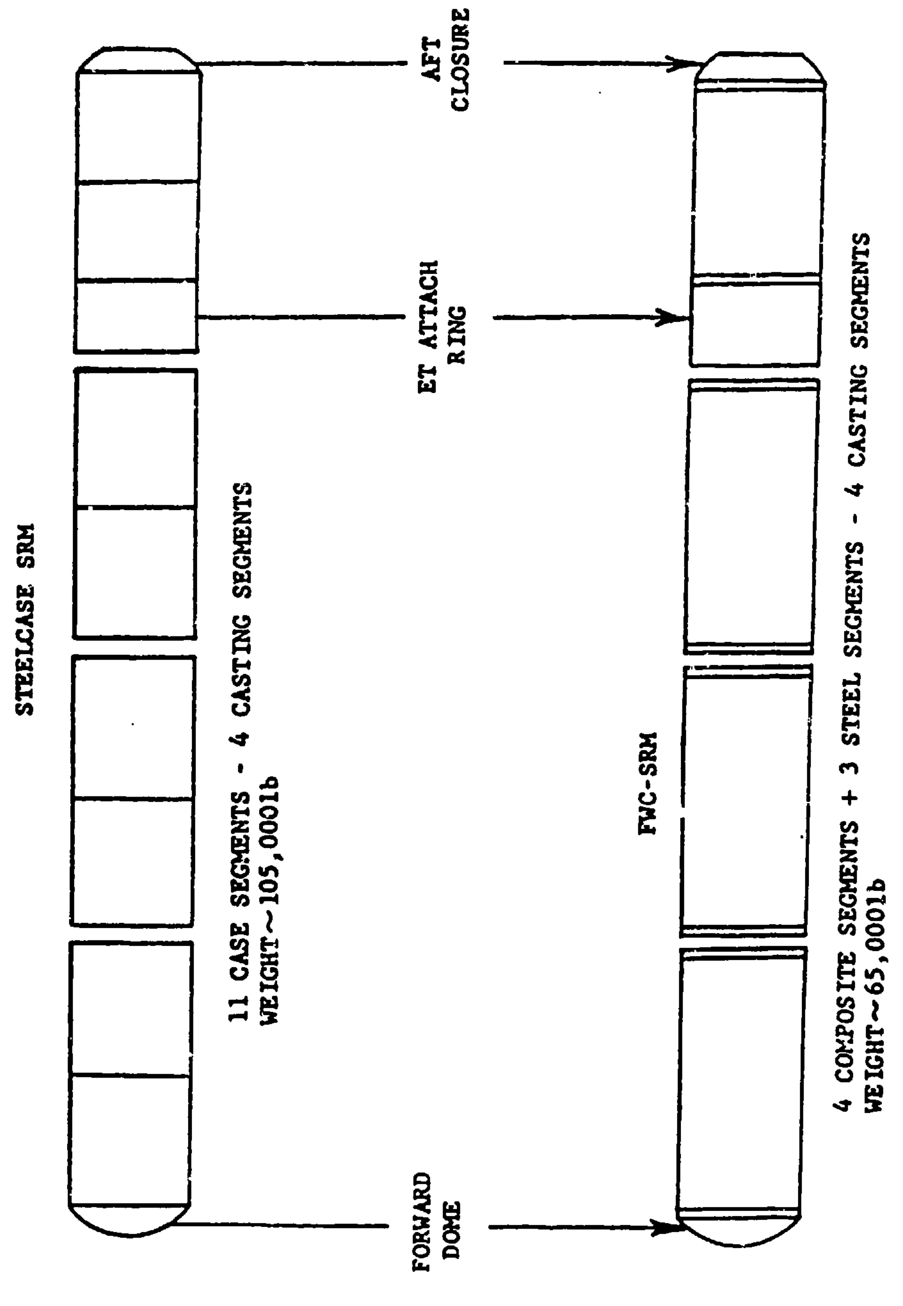

Figure C1-22 Solid Rocket Motor Parts - (Steel-case SRM and Filament Wound Case SRM)

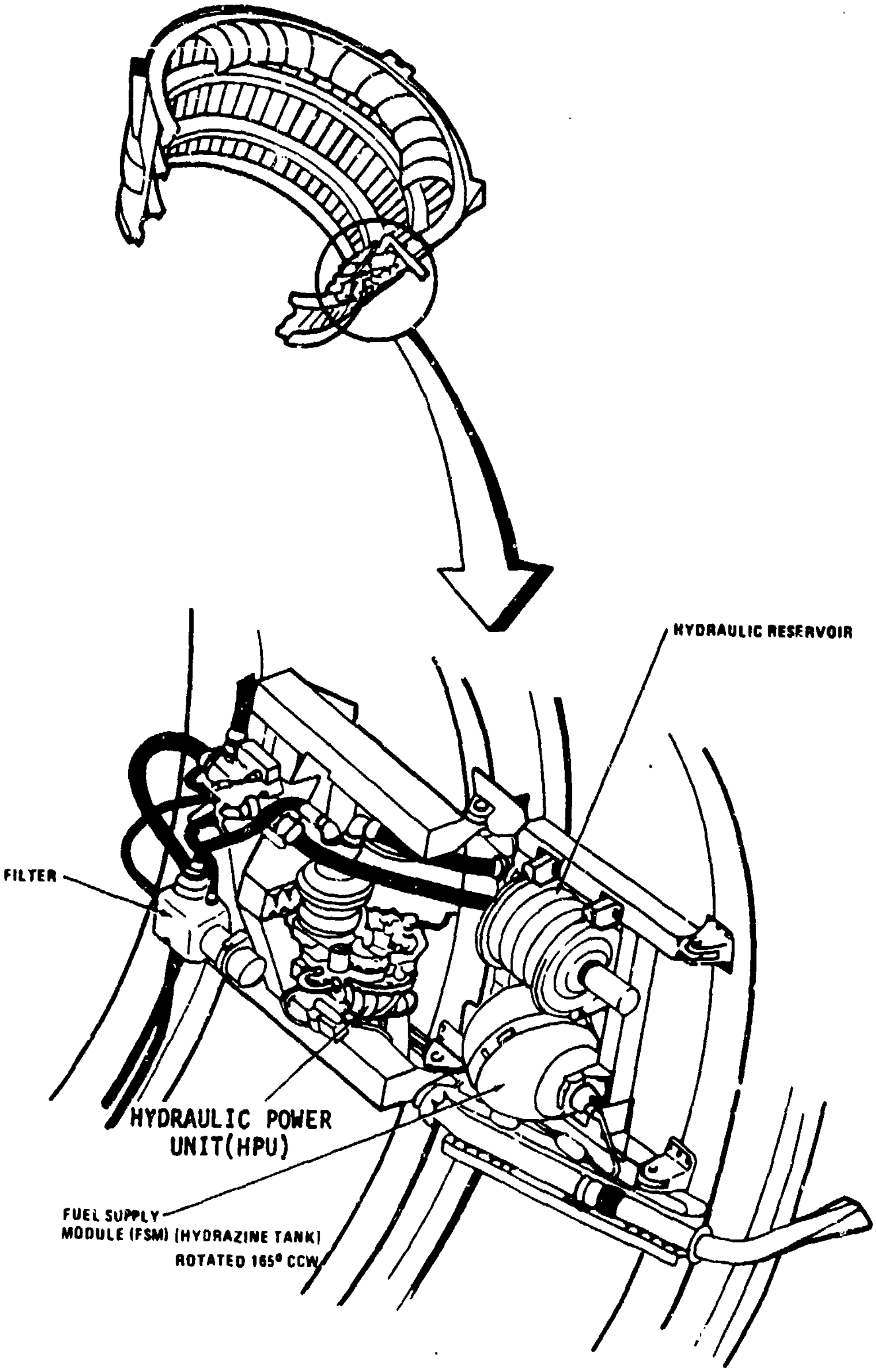

Figure C1-23 Hydrazine Tank on a Shuttle SRM

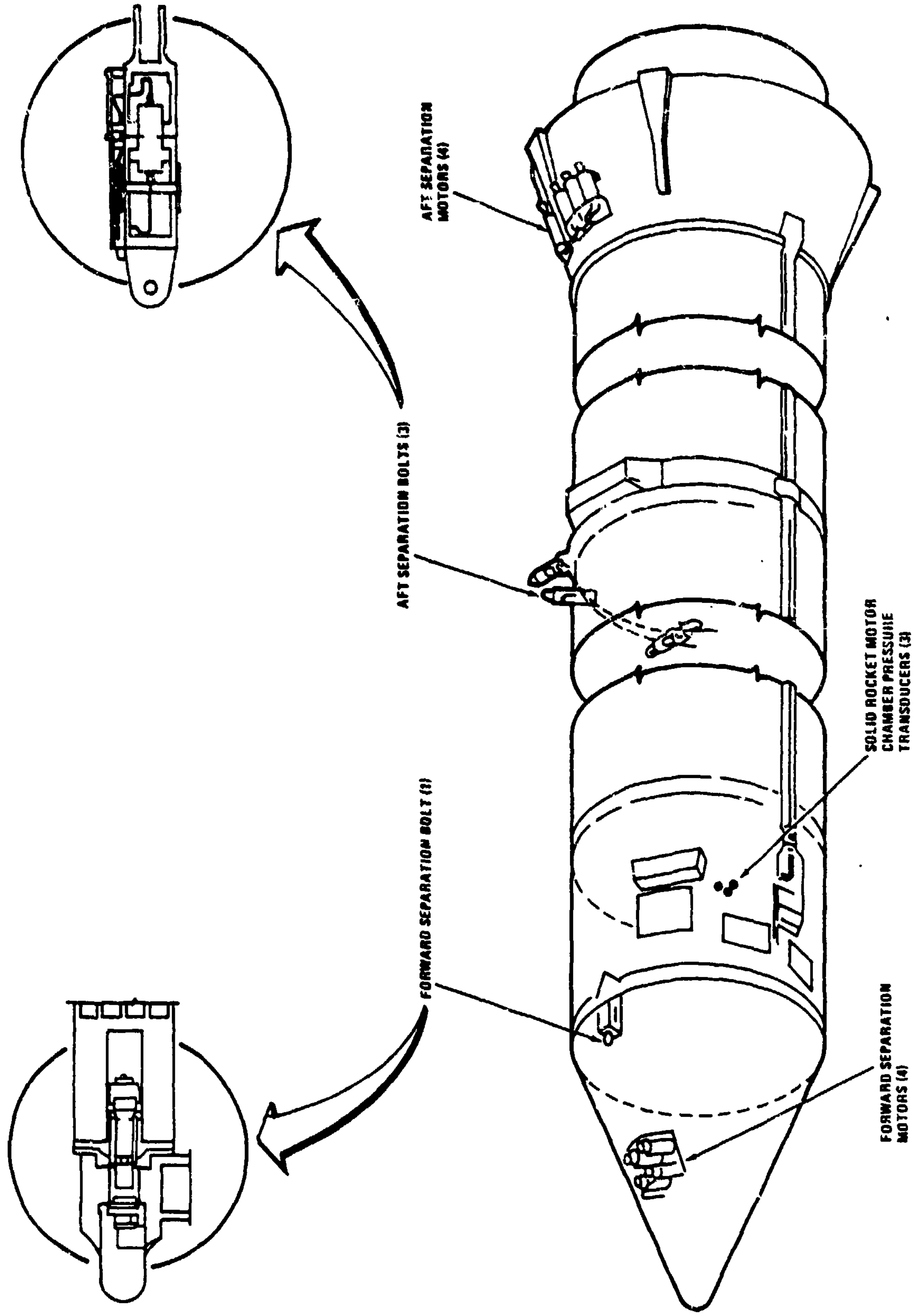

Figure C1-24 Shuttle Solid Rocket Motor Separation System

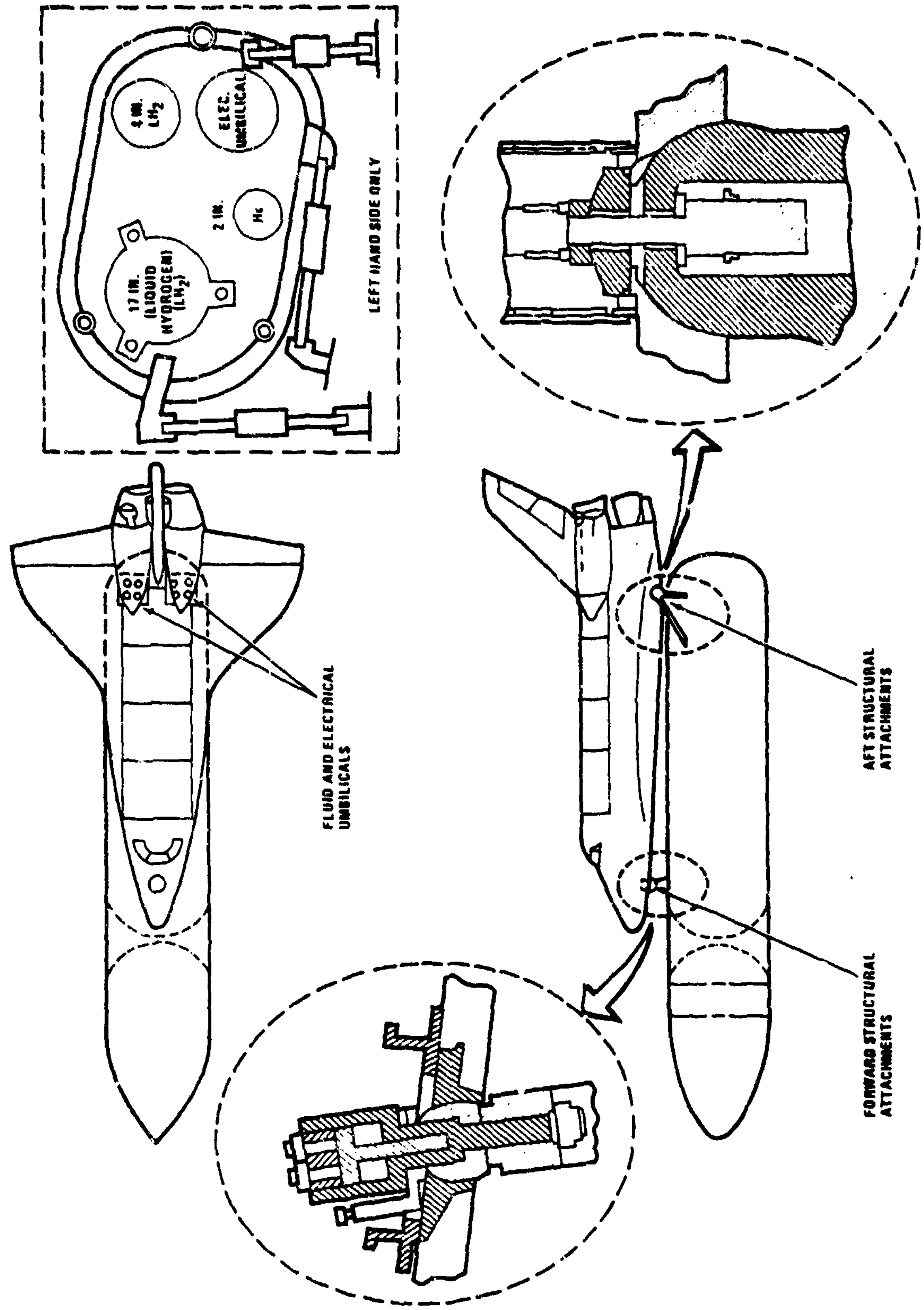

Figure C1-25 Orbiter/External Tank Separation System

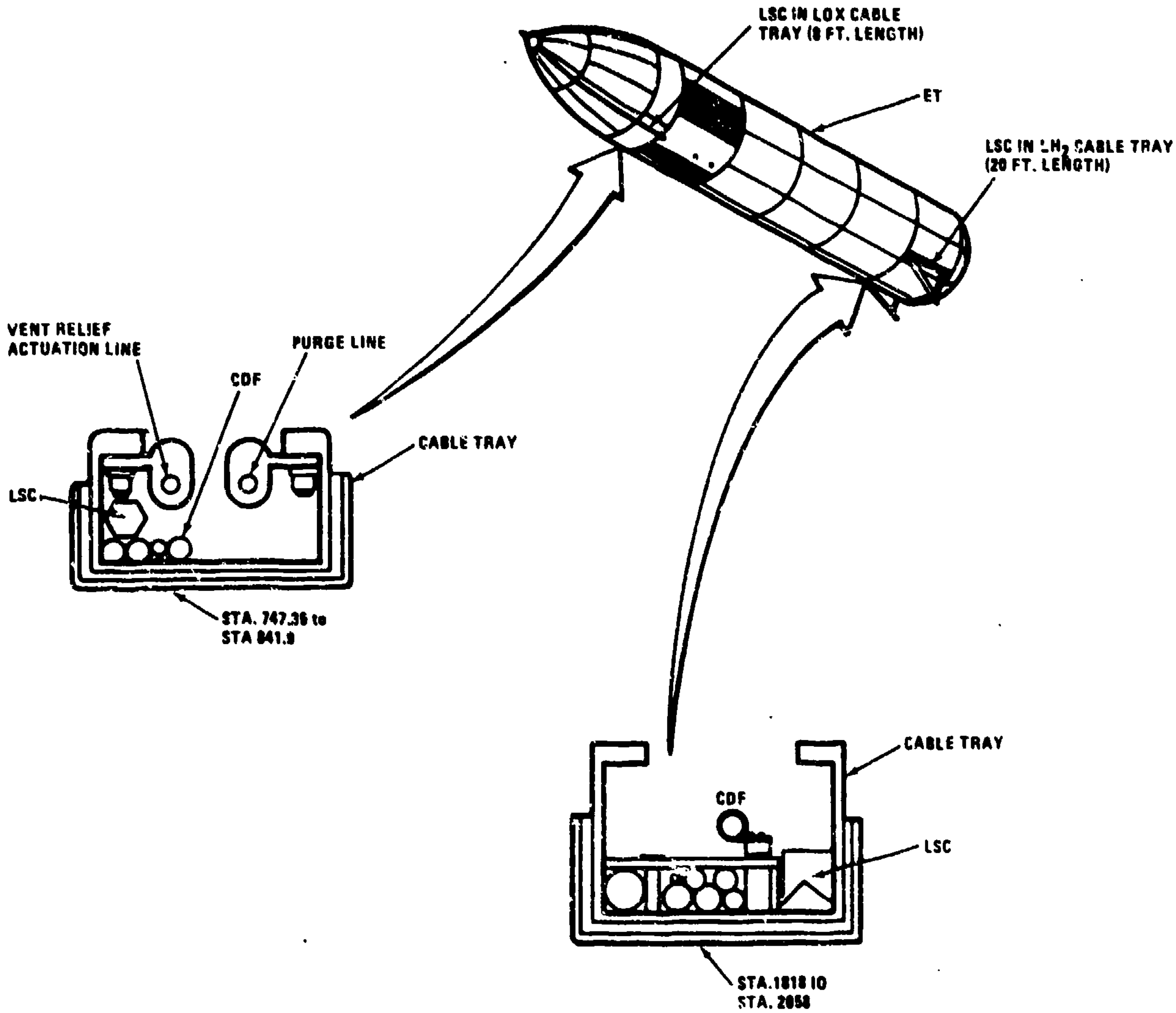

Figure C1-26 Shuttle ET Command Destruct Charges

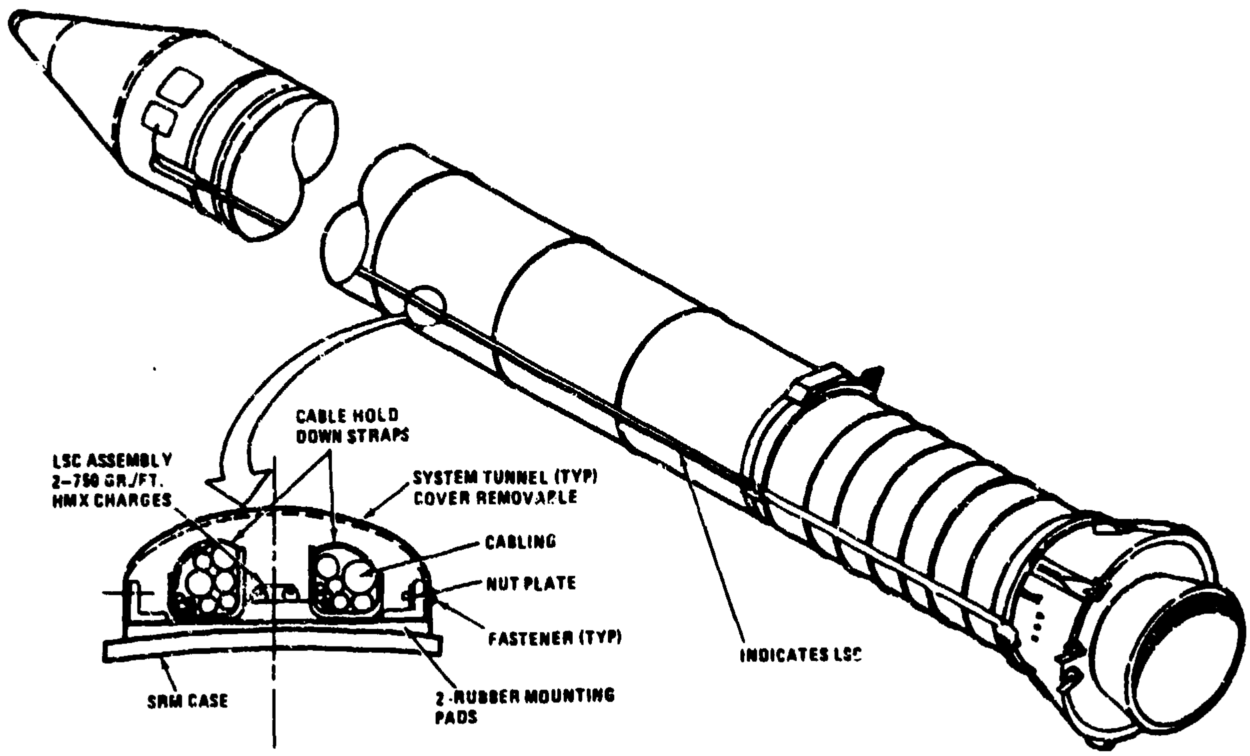

Figure C1-27 Shuttle SRM Command Destruct Charges

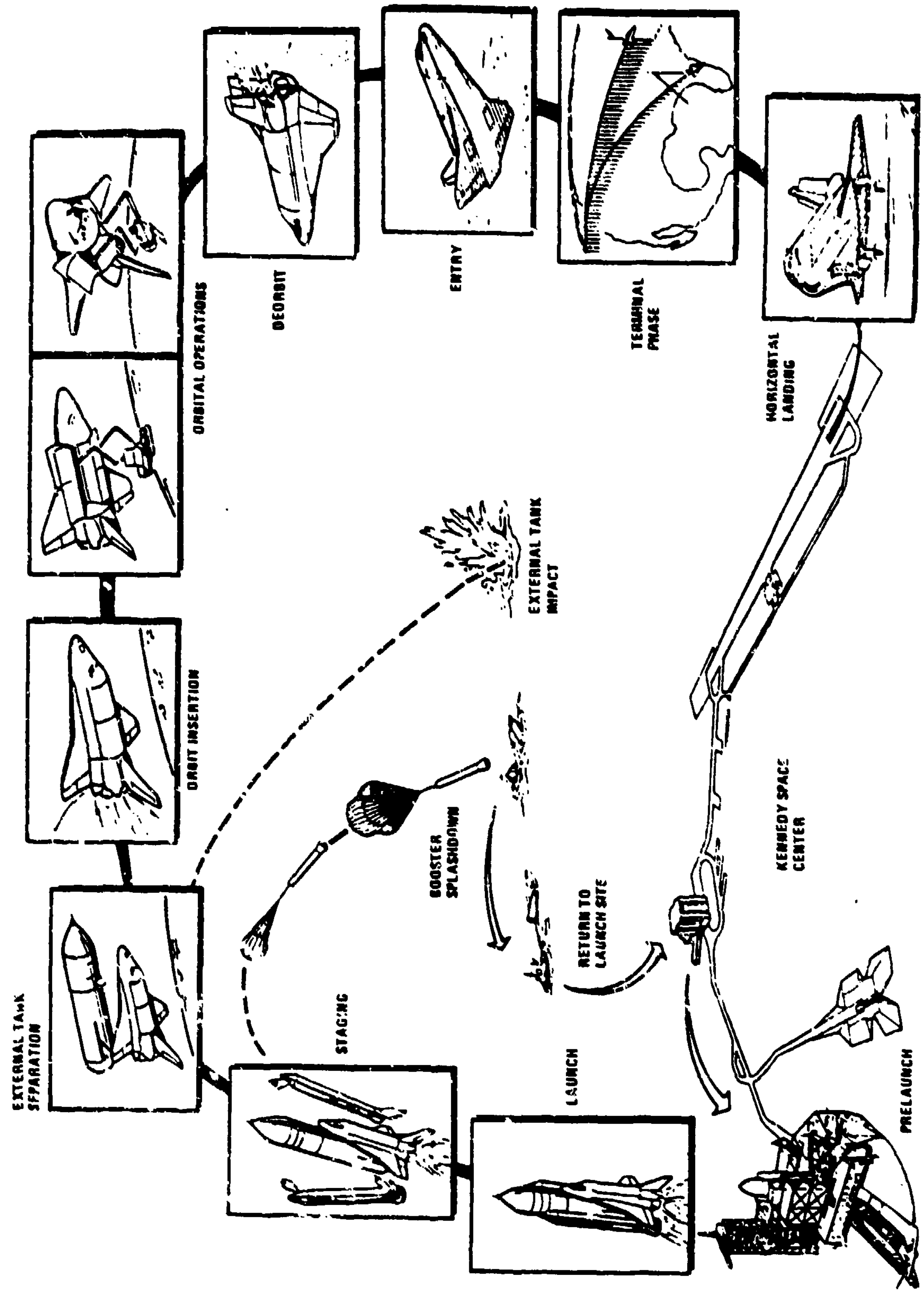

Figure C1-28 Basic Mission Cycle for Shuttle

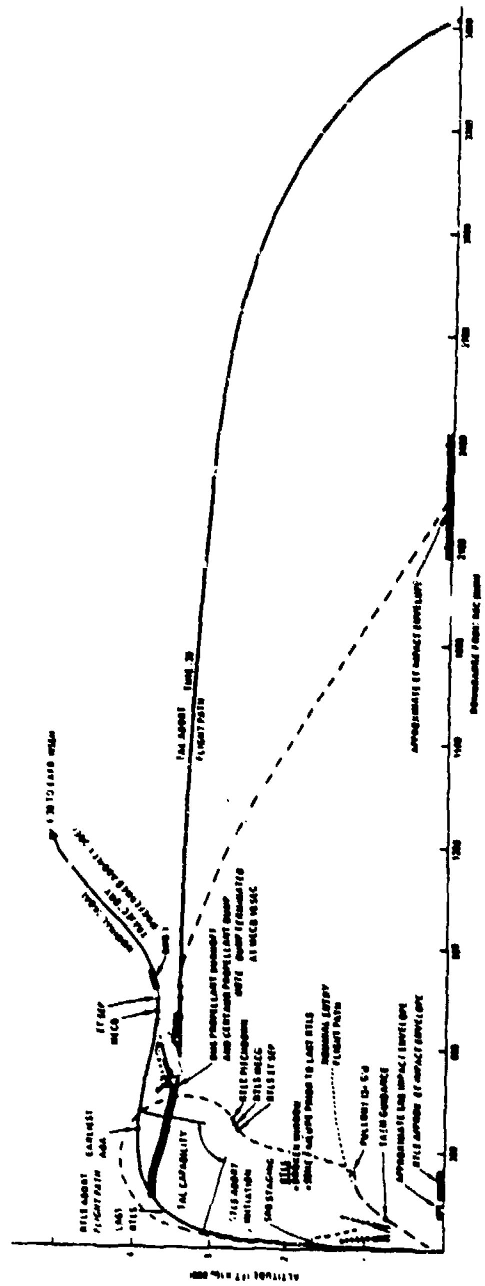

Figure C1-29 RTLS and TAL Aborts Altitude vs Range

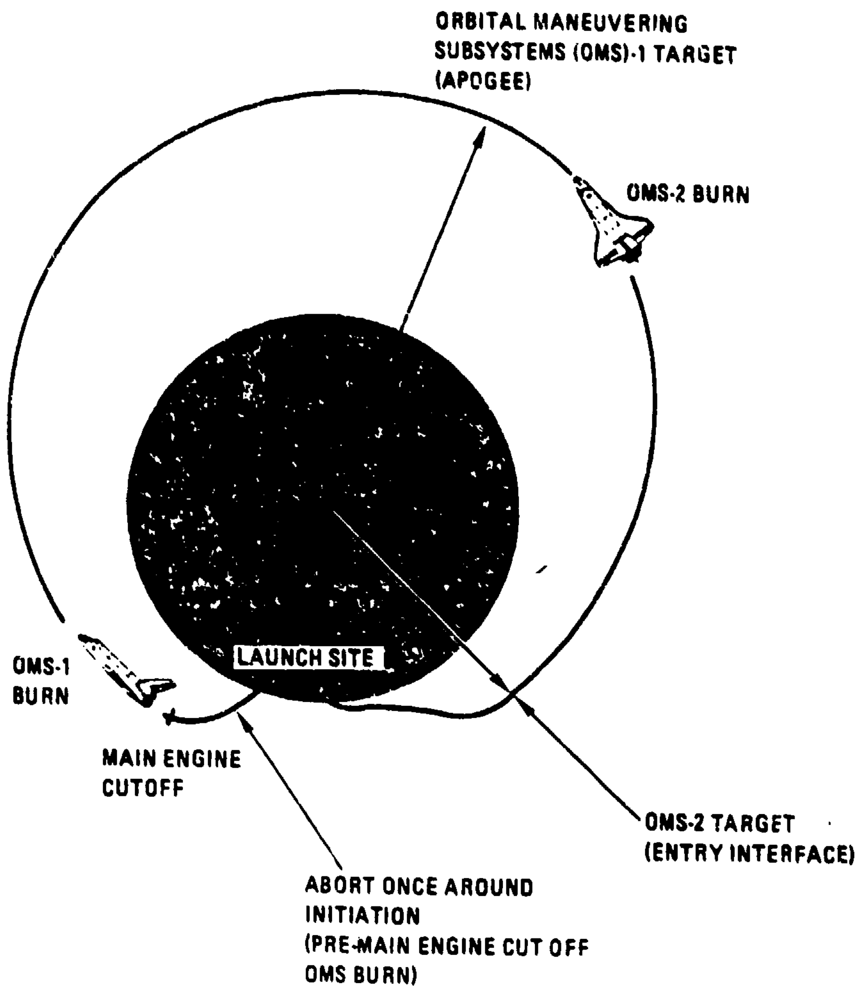

Figure C1-30 Profile of AOA

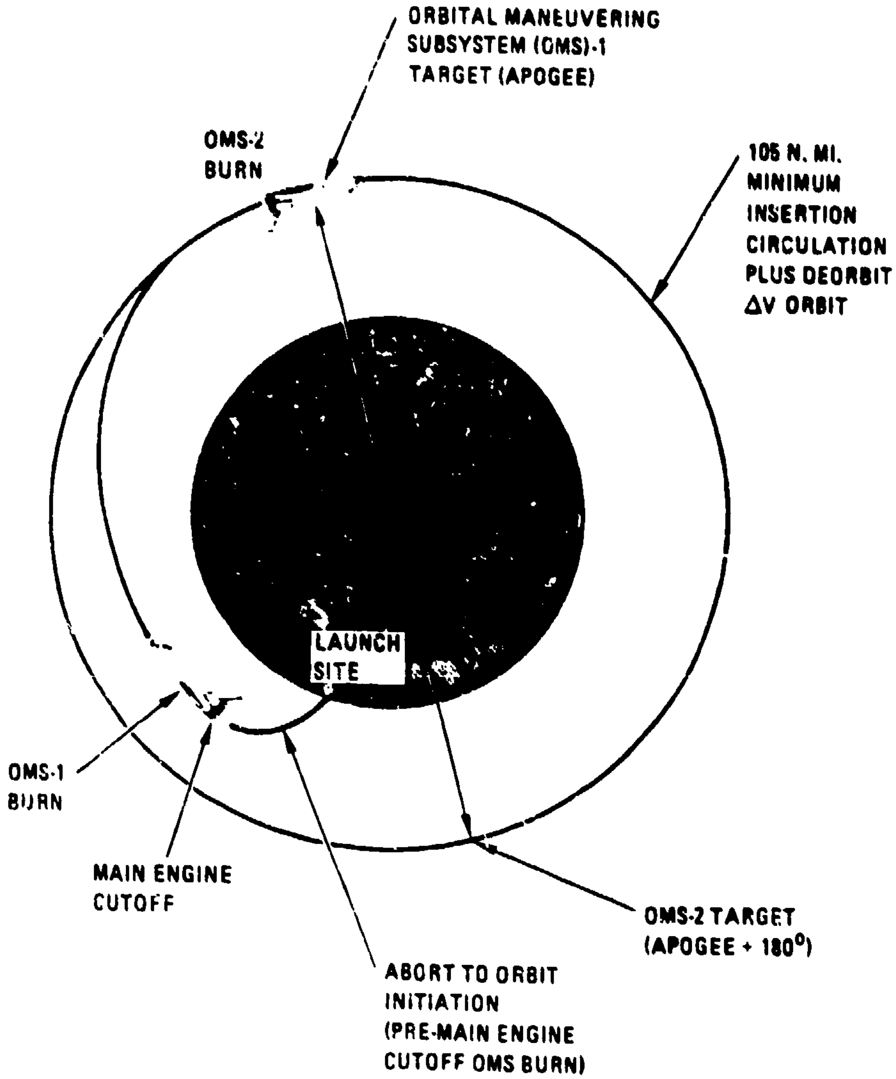

Figure C1-31 Profile of ATO

Appendix C1 Supplement Space Transportation System--IUS

APPENDIX C1 SUPPLEMENT (C1(S))
SPACE TRANSPORTATION SYSTEM (STS)

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SPACE TRANSPORTATION SYSTEMS (STS)

C1(S).0 INTRODUCTION

The data of this Appendix were extracted from the National STS Program document "Space Shuttle data for Planetary Mission Radioisotape Thermoelectric Generator (RTG) Safety Analysis, JSC 08116, Revision A, October 1, 1987. This description includes IUS missions.

C1(S).1 SPACE TRANSPORTATION SYSTEM - GENERAL DESCRIPTION

This section presents a general description of the Space Transportation System (STS). This section contains three main subsections. The first discusses the STS as a whole. The second covers the Shuttle itself: the Orbiter, the External Tank (ET), and the pair of Solid Rocket Boosters (SRBs). The third covers the launch site at the Kennedy Space Center (KSC). Although any given STS mission may well include an upper stage and payload, these components are discussed in their own sections in this appendix. Figure C1(S)-1 shows the Galileo Shuttle launch configuration. The gross weight of the vehicle, with a maximum 65,000 pound payload, is nearly 4.5 million pounds.

The basic mission of Space Shuttle is to be a space transport, carrying payloads into space and returning to Earth for another load. What happens to the payloads in space depends on the mission. On some missions the payload is released into orbit. On other missions the payload is a spacecraft and an upperstage rocket; the rocket boosts the spacecraft into another orbit or into an Earth-escape trajectory. On a third kind of mission, the payload stays in the Shuttle Orbiter and is carried back to Earth. On yet a fourth kind of mission, the Orbiter crew retrieves a satellite already in orbit and returns it to Earth.

Figure C1(S)-2 shows the basic mission cycle. It starts with the launch of the Shuttle vehicle, with the SRBs and the Space Shuttle Main Engines (SSMEs) firing. After approximately 2 minutes, at about 26 miles altitude, the SRBs are jettisoned and fall back to Earth for recovery. After approximately 8 minutes of flight, before the Orbiter attains orbit, the SSMEs are shut down, and the ET is separated and falls back to Earth for oceanic termination. The orbital maneuvering system engines thrust the Orbiter into orbit. The mission-dependent orbital operations then ensue and, on their completion, the Orbiter returns to Earth to be made ready for its next flight.

In its entirety, the STS consists of the Shuttle, a variety of standard payload carriers, ground integration and launch facilities, and ground based payload operations control centers. The ground integration and launch facilities consist of the KSC and Vandenberg Air Force Base (VAFB) complexes, although the latter is not yet operational for Shuttle launches. There are three ground-based payload operations control centers located at the Jet Propulsion Laboratory (JPL), the Goddard Space Flight Center (GSFC), and the Johnson Space Center (JSC). Communications are via the world-wide Space

Tracking and Data Network (STDN), the Tracking and Data Relay Satellite System (TDRSS), and the Orbiter and/or carrier command and data handling system. Direct payload communication is also possible. The Orbiter is transported between ground sites such as Edwards Air Force Base (EAFB) and KSC a top the Shuttle Carrier Aircraft (SCA), a modified Boeing 747 aircraft.

C1(S).1.2 Orbiter, External Tank, and Solid Rocket Boosters.

This section describes the Shuttle Vehicle: the Orbiter, the ET, and the SRBs. The description here contains eight principal parts: subsystems, structures, operation cycle, propulsion, stage separation, command destruct system, pressure tanks, and Reaction Control Subsystem (RCS).

Figures C1(S)-3 through C1(S)-6 show the Shuttle Vehicle, with side, top, front, and back views.

In places in this document, the locations of items are given in terms of vehicle coordinates. Figure C1(S)-7 shows the coordinate systems.

The X, Y, Z coordinate systems for the Orbiter, ET, SRB, and Shuttle System are designated by the subscript letters O, T, B, and S. The unit of measurement is the inch to the required decimal place. In each coordinate system, the X-axis zero point is located forward on the nose tip; that is, the Orbiter nose tip location is 236 inches aft of the zero point (at X_O 236); the ET nose cap tip location is at X_T 322.5; and the SRB nose tip location is at X_B 200. In the Orbiter, the horizontal X_O, Y_O reference plane is located at Z_O 400, which is 336.5 inches above the ET horizontal X_T, Yt reference plane located at ZT 400. The SRB horizontal X_B, Y_B planes are located at $+Y_S$ 250.5 and $-Y_S$ 250.5. Also, note that the Orbiter, ET, and Shuttle System center X, Z planes coincide.

From the X = 0 point, aft is positive, and forward is negative for all coordinate systems. Looking forward, each Shuttle element Y-axis point right of the centerplane (starboard) is positive, and left of center (port) is negative. The Z-axis of each point within all elements is positive with Z = 0 located below the element, except for the SRBs in which each Z-coordinate point below the SRB X_B, Y_B reference plane is negative and each point above that plane is positive.

The Shuttle System and Shuttle elements coordinate systems are related as follows: The ET X_T 0 point coincides with X_S 0, the SRB X_B 0 point is located 543 inches aft, and the Orbiter Y_O, Z_O reference plane is 741 inches aft of X_S 0.

C1(S).1.2.1 Subsystems. This section describes the subsystems of the Orbiter, the ET, and the SRBs.

The Orbiter contains 17 subsystems. They are structure; thermal protection; thermal control; main propulsion; orbital maneuvering; reaction control; mechanical (including ET separation); auxiliary power units; hydraulic; electrical power; power reactant storage and distribution; environmental control and life support; purge, vent, and drain; avionics; guidance, navigation, and control; communications and tracking; and data processing and software.

The following paragraphs describe these subsystems. Figures C1(S)-8 through C1(S)-11 show the layout of the Orbiter and many of the parts of the subsystems.

The majority of the Orbiter structure is of conventional aluminum construction protected by Reusable Surface Insulation (RSI). Section C1(S).1.2.2 describes the structure in more detail.

The Thermal Protection Subsystem (TPS) keeps the outer skin of the Orbiter airframe from getting too hot, both during ascent and during reentry. The TPS materials are attached to the outside of the primary structural shell of the Orbiter. There are several types of TPS materials; where each type is used depends on the temperatures expected. The various types are as follows:

a. Coated Nomex felt Flexible RSI (FRSI) is used in areas where temperatures are less than 750 degrees F for entry and 830 degrees F for ascent: i.e., upper cargo bay door, mid and aft fuselage sides, upper wing, and Aft Propulsion System (APS) pod.

b. Low-temperature Reusable Surface Insulation (LRSI) is used in those areas where temperatures are below 1200 detrees F and above 750 degrees F under design heating conditions.

c. High-temperature Reusable Surface Insulation (HRSI) is used in those areas exposed to temperatures below 2300 degrees F and above 1200 degrees F under design heating conditions.

d. Advanced Flexible Reusable Surface Insulation (AFRSI) in an alternative to LRSI and is used in those areas where temperatures are below 1800 degrees F and above 750 degrees F under design heating conditions. AFRSI is used extensively on later Orbiter Vehicles and as improvements in certain locations on earlier Orbiters.

e. Reinforced Carbon-Carbon (RCC) is used on areas such as wing leading edge and nose cap where predicted temperature exceed 2300 degrees F under design heating conditions.

F. Thermal window panes and metal (forward) RCS fairings and elevon upper surface rub seal panels.

g. Thermal barriers are installed around operable penetrations (egress hatch, landing gear doors, etc.) to protect against aero thermal heating.

The TPS is a passive system. It has been designed for ease of maintenance and for flexibility of ground and flight operations while satisfying its primary function of maintaining acceptable airframe outer skin temperatures. The basic RSI materials are of three types: (1) ceramic LI-900, LI-2200, and FRCI-12 materials used for HRSI and LRSI, (2) Nomex felt material used for FRSI, and (3) quilted fibrous silica material used for AFRSI. HRSI and LRSI differ in the coatings applied for waterproofing, handling, and optical property control. HRSI has a black ceramic coating with a design surface emittance of 0.85 and a design solar absorbance of about 0.85; LRSI has a white ceramic coating with a design surface emittance of 0.8 and a design solar absorbance of 0.32. The FRSI has a pigmented silicone coating with design optical properties the same as those for the LSI coating to rigidize its surface without affecting its thermal properties.

Figure C1(S)-12 through C1(S)-14 show the TPS materials and their location on the different Orbiters.

The Thermal Control Subsystem (TCS) maintains subsystems and components within specified temperature limits for all mission phases, including prelaunch, launch, Earth orbit, entry, and post-landing. The integrated thermal control management uses available heat sources and heat sinks supplemented by passive thermal control techniques. Those techniques are primarily the use of insulation, thermal control coatings, and thermal isolation to attenuate heat transfer to or from critical spacecraft compartments and hardware. Supplemental heaters are provided as required when these methods are not adequate to provide the necessary temperature control.

The main propulsion subsystem (MPS) of the Orbiter includes three engines, which burn for about 8 minutes from just before lift-off until slightly before attaining orbit. The rated (sea-level) thrust of each engine is 375,000 pounds. The engines are gimbaled for steering. The fuel is Liquid Hydrogen (LH_2), and the oxidizer is Liquid Oxygen (LOX). The propellants for the SSMEs are in the ET. Section C1(S).1.2.4.1 describes the SSMEs in more detail.

The Orbital Maneuvering Subsystem (OMS) includes two engines. The thrust from those engines carries the Orbiter into orbit after the SSMEs are shut down, provides for maneuvering while in orbit, and retards the Orbiter out of orbit for reentry. The rated (vacuum) thrust of each engine is 6000 pounds. The engines are gimbaled for steering. The fuel is monomethylhydrazine (MMH) and the oxidizer is nitrogen tetroxide (N_2O_4). Section C1(S).1.2.4.3 describes the OMS engines in more detail.

The RCS of the Orbiter includes 44 thrusters (38 primary, 6 vernier) for attitude control and three-axis translation during orbit insertion and on-orbit and entry phases of the Orbiter flight. The propellant are MMH (fuel) and N_2O_4 (oxidizer). Section C1(S).1.2.8 describes the RCS system in more detail.

The mechanical subsystems of the Orbiter operate the aerodynamic control surfaces, landing/deceleration system, payload bay doors, deployable radiators, payload retention and payload handling subsystems, and provide for disconnecting propellant feedlines between the Orbiter and the ET and for separating the Orbiter from the ET.

The auxiliary power units (APUs) drive the hydraulic pumps. The units work by decomposition of hydrazine (N_2H_4).

The hydraulic subsystem provides hydraulic power for many of the mechanical devices in the Orbiter, for steering the SSMEs, and for controlling the propellant valves in the SSMEs.

The electrical power system supplies electricity from hydrogen-oxygen fuel cells. The power reactant storage and distribution subsystem provides hydrogen and oxygen to the electrical fuel cells and oxygen to the Environmental Control and Life Support Subsystem (ECLSS). The ECLSS provides a habitable environment for the crew. The purge, vent, and drain subsystem provides environmental control of the Orbiter's unpressurized structural cavities, which includes most of the Orbiter except for the crew module. The avionics subsystem includes all the electronic gear in the Orbiter. The guidance, navigation, and control subsystem provides guidance commands for control inertial navigation, and automatic and manual control capability. The communications and tracking subsystem provides communication between the Orbiter and ground stations, the Orbiter and released payloads, and between the Orbiter and the TDRSS. The data processing and software subsystem provides computing capability for the Orbiter.

The ET (see Figure C1(S)-15) supplies the Orbiter MPS with Liquid Hydrogen (LH_2) and Liquid Oxygen (LOX) at prescribed pressures, temperatures, and flow rates. Both the LH_2 and LOX tanks are equipped with a vent and relief valve to permit loading, pressurization, and relief functions. Tank level sensors provide for propellant loading and shutdown signals. The ET is thermally protected with a nominal 1-inch thick Spray-On Foam Insulation (SOFI), with additional SOFI and a charring Super-Lightweight Ablator (SLA 561) in places to withstand localized high heating. Since the ET is an expendable element, the ET subsystems are designed for single usage to minimize costs.

The ET reacts the SRB thrust through its innertank and provides attach structure to the Orbiter to react the SSME thrust. At lift-off, the ET contains 1.55 million pounds of usable propellant. At Main Engine Cutoff (MECO), the ET is separated from the Orbiter before orbital velocity is achieved. The ET then proceeds on a ballistic reentry path for a safe impact in the ocean.

Linear-Shaped Charges (LSCs), one on each tank in the ET, can split the tanks on command for range safety. Section C1(S).2.1 describes the range safety system in more detail.

Two SRBs burn in parallel with the Orbiter MPS to provide initial ascent thrust. The primary elements of the SRB are the motor, including case, propellant, igniter, and nozzle; structural systems; separation, Operational Flight Instrumentation (OFI), and recovery avionics; separation motors and pyrotechnics; and deceleration, range safety destruct, and Thrust Vector Control (TVC) subsystems. Each SRB (steel-cased) weighs approximately 1.293 million pounds.

Figure C1(S)-16 shows an SRB. The following paragraphs describe the elements of an SRB.

Each SRB produces 2.9 million pounds of thrust at sea level. The engines are gimbaled for steering. Section C1(S).1.2.4.2 describes the SRB engine in more detail.

The SRB case is segmented. The skirts at the aft end of the SRBs sit on the mobile launch platform before lift-off.

Two lateral sway braces and a diagonal attachment at the aft frame provide the structural attachment between the SRB and the ET. The SRB forward attachment to the ET is by a single thrust attachment at the forward end of the forward skirt. The same forward skirt is used for attaching the main parachute riser attachments.

The SRBs are released from the ET by pyrotechnic devices at the forward thrust attachment and the aft sway braces. Eight separation rockets on each SRB, four aft and four forward, separate the SRB from the Orbiter and the ET.

The forward section provides room for the SRB electronics, recovery gear, range safety destruct system, and forward separation rockets. The parachute deceleration subsystem consists of a pilot parachute, a ribbon drogue parachute, and three ribbon main parachutes.

C1(S).1.2.2 Structures. This section describes the structural elements of the Orbit, the ET, and the SRBs.

C1(S).1.2.2.1 Orbiter Structure. Figures C1(S)-17 through C1(S)-20 show the Orbiter structure.

The majority of the Orbiter structure is of conventional aluminum construction protected by RSI.

The forward fuselage structure is composed of 2024 aluminum alloy skin/stringer panels, frames, and bulkheads. The crew module, which is supported within the forward fuselage by four attach points, is welded to create a pressure-tight vessel. The module has a side hatch for normal ingress and egress and a hatch from the airlock into the payload bay.

The mid fuselage structure, which is primarily of aluminum construction, connects to the forward fuselage, aft fuselage, and the wing. It runs from X_O 582 to X_O 1307 and provides the support for the Orbiter payloads including the payload bay doors. In the forward portion is the forward wing glove fairing, which runs from X_O 582 to X_O 807. The side wall just forward of the wing carry-through structure provides the inboard support for the main landing gear. The total lateral landing gear loads are reacted by the mid fuselage structure.

The mid fuselage skins are integrally machined except the panels above the wing from X_O 1040 to X_O 1307 which are aluminum honey comb panels. The integrally machined skins have longitudinal T-stingers. The bottom panel between X_O 1191 and X_O 1307 provides a dual function of carrying transverse wing loads as well as body bending loads and is of waffle construction. The side skins in the wing interface from X_O 80. to X_O 1307 are also machined but have vertical stiffeners except from X_O 1278 to X_O 1307, which is a waffle configuration.

There are 12 main frame assemblies which stabilize the mid fuselage structure and react wing and payload loads. The frame assemblies consist of vertical side elements and horizontal elements. The side elements are machined, and the horizontal elements have machined flanges with boron/aluminum time shear trusses. In addition to the 12 main frames, there are 13 machined side wall stub frames.

In the upper portion of the mid fuselage are the sill and door longerons. The machined sill longerons are not only primarily body bending elements, but also serve to take the longitudinal loads from the payloads. Attached to the door longerons and associated backup structure are the 13 payload bay door hinges. These hinges provide the vertical reaction from the payload bay doors, and five of the hinges react the payload bay door shears. The sill longeron also provides the base support for the payload manipulator arms and its storage provisions, rendezvous sensor, and payload bay door actuation system. To avoid excess weight, the payload manipulator arm will be removed for the Galileo and Ulysses missions.

The mid fuselage provides the wing carry-through structure between Stations 1191 and 1307. It consists of the lower skin and upper wing box carry-through skin and seven longitudinal rib members. The center rib is composed of integrally machined caps, shear web, and vertical stiffeners, and the other six have boron/aluminum tube truss members. The upper wing box carry-through skin, X_O 1191 to X_O to 1307, carries the transverse wing loads and is machined with lateral T-stingers.

The aft fuselage includes a thrust-type internal structure of diffusion-bonded elements that transfers the SSMEs' thrust loads to the mid fuselage and the ET. The external surface of the aft fuselage is of standard construction except for the removable Aft Propulsion System (APS) pods which are constructed with graphite epoxy skins and frames. A machined aluminum plate heat shield with thermal insulation at the rear of the vehicle provides protection to the SSME systems.

The wing is of conventional aluminum alloy construction. Corrugated spar web, truss-type ribs, and riveted skin-stringer and honeycomb covers are used. The elevons are of aluminum honeycomb construction and are split into two segments to minimize hinge binding and interaction with the wing.

The vertical tail is of conventional aluminum alloy construction consisting of a two-spar, multirib, integrally machined skin assembly. The tail is attached to the aft fuselage by bolted fittings at the two main spars. The rudder/speed brake assembly is divided into upper and lower sections. Each section is also split longitudinally and individually actuated to serve as both rudder and speed brake.

These major structural assemblies are mated and joined together with rivets and bolts. The mid fuselage is joined to the forward and aft fuselages primarily with shear ties, with the mid fuselage skin overlapping the bulkhead caps at X_0 582 and X_0 1307. The wing is attached to the mid fuselage and aft fuselage primarily with shear ties, except in the area of the wing carry-through where the upper panels are attached with tension bolts. The vertical tail is attached to the aft fuelage with bolts which work in both shear and tension. The body flap, which is constructed with aluminum honeycomb covers, is attached to the aft lower fuselage by four rotary actuators.

C1(S).1.2.2.2 <u>External Tank Structure</u>. The main parts of the ET are the LOX tank at the forward end, an intertank, and a LH_2 tank at the aft end. The LH_2 tank is about three times as large as the LOX tank. Figure C1(S)-15 shows the ET.

The ET consists of a forward LOX tank, an unpressurized intertank, and a LH_2 tank. The LOX tank (volume = 19,500 ft^3) is an aluminum alloy monocoque structure composed of a fusion-welded assembly of performed, chemicaled gores, panels, machined fittings, and ring chords. The LOX tank is designed to operate at a nominal pressure range of 20 to 22 psig. The tank contains antislosh and antivortex baffles as well as an antigeyser system to control conditions. A 17-inch diameter feedline conveys propellant through the intertank and externally aft to the ET/Orbiter disconnect. The tank's double wedge nose cone reduces drag and heating and also serves as a lightning rod.

The intertank is semimonocoque cylindrical structure with flanges on each end for joining the LOX and LH_2 tanks. The intertank contains the SRB thrust beam and fitting which distribute SRB loads to LOX and LH_2 tanks. The intertank provides an umbilical plate which interfaces with a ground facility arm. The umbilical plate accommodates purge gas supply, hazardous gas detection, and hydrogen gas boil-off during ground operations. The intertank consists of mechanically joined skin, stringers, and machined panels of aluminum alloy. The intertank is vented in flight.

The LH_2 tank (volume = 53,518 ft^3) is a semimonocoque structure composed of four fusion-welded barrel sections, five beam ring frames, and forward and aft 0.75 ellipsoidal domes. The LH_2 tank is designed to operate after lift-off at a nominal pressure of 32 to 34 psig; the lift-off pressure is 40.9 to 44.1 psig.

The tank contains an antivortex baffle and a siphon outlet to transmit propellant to the ET/Orbiter disconnect through a 17-inch diameter line. The LH_2 tank has provisions for the ET/Orbiter forward attach strut, two ET/Orbiter aft attach fittings, the thrust distribution structure, and the aft SRB/ET stabilizing strut attachments.

To reduce the ET weight and thus achieve greater payload capability, certain changes have been made to the original ET described above. These changes, which have reduced the dry weight of the ET by over 6000 pounds are incorporated in the Lightweight Tank (LWT) version of the ET. The LWT first flew on STS-6 and became the standard version from STS-8 onward, after STS-7 used the last of the original version ETs. Among the changes incorporated in the LWT were the elimination of five LOX slosh baffle rings, deletion of the anti-geyser line and incorporation of helium inject into the main feed line.

C1(S).1.2.2.3 SRB Structure. There are two versions of the SRB (Figure C1(S)-21), and the major difference between them is the structural material of the Solid Rocket Motor (SRM) case itself. The initial version has a Steel Case (SC) SRM, while the second version will have a composite Filament-Wound Case (FWC). Although the SC-SRB is operational, the FWC-SRB is still under development. The major advantages of the FWC-SRB is its lighter weight - about 30,000 pounds less than the SC-SRB, thus making possible a heavier STS payload. However, because of the higher cost of the FWC, it will only be used when its payload advantage is needed. The Inertial Upper Stage (IUS) planetary missions are planned to use the steel case SRBs.

Overall, the SRB is 150 feet long and is 148 inches in diameter, although the FWC diameter is slightly greater because of its thicker case. The inert weight of the SC-SRB is 190,000 pounds, which is about 30,000 pounds greater than the FWC-SRB. The propellant weight of each is about 1.1 million pounds, and the subdivision of this total weight is described in section C1(S).1.2.4.2. The major elements consist of the structural assemblies)aft skirt, forward skirt, and nose assembly) and the four SRM segments. The parachutes are mounted in the nose assembly, electronics in the forward skirt, and the TVC system in the aft skirts. The structural assemblies are designed for 40 uses, the SC-SBM for 20 uses, the electronics and TVC hardware for 20 uses, and the parachutes for 10 uses. The FWC-SRM is designed for single time use only, although the feasibility of reusing it is under study.

C1(S).1.2.2.3.1 Solid Rocket Motor. The SRMs are cast and delivered to the launch site in four segments. The SC segments are roll formed D6ac steel with pinned clevis joints. Two O-ring seals in each joint provide redundancy for the maintenance of pressure integrity. The design and fabrication of the case are a scaled-up version of the Titan III motor cases. Structural design factor of safety for the SC is 1.4, typical of man-rated vehicles. Only the four straight cylindrical sections of the FWC-SRM (Figure 1-21b) are actually filament wound, the other segments being steel as before. The materials used for the FWC are Hercules AS4W-12K graphite fiber (modulus 33 X 10^6 psi), and Hercules HBRF-55A resin (bisphenol A epoxy, 1, 4 butanedial epoxy and curing agent consisting of a blend of methylene dianiline with a m-phenylene dianiline). The FWC design safety factors are 1.4 on metal parts and membrane structure, and 2.0 on composites at joints and discontinuities.

The composite propellant is a proven Polybutadiene Acrylonitrile (PBAN) propellant used in the Minuteman and poseidon systems. More than 200 million pounds of this propellant have been produced. The propellant is vacuum cast and case bonded.

The SRM nozzle is a 20 percent submerged, omnidirectional movable nozzle. The throat diameter is 54 inches and the diameter at the end of the exit cone is 148 inches. The nozzle has an aft pivoted, flexible bearing that provides an omniaxial TVC deflection capability of plus or minus 8 degrees. All metal parts of the nozzle are designed for 20 uses.

C1(S).1.2.2.3.2 Aft Skirt. The aft skirt provides attach points to the launch support structure and support to the Space Shuttle on the Mobile Launcher Platform (MLP) for all conditions prior to SRB ignition. The aft skirt provides aerodynamic protection, thermal protection, and mounting provisions for the TVC subsystem and the aft mounted separation motors. The aft skirt provides sufficient clearance for the SRM nozzle at the full gimbal angles. The aft skirt kick ring provides the necessary structural capability to absorb and transfer induced prelaunch loads.

The aft skirt structure assembly is a welded and bolted conical shape, 146 inches in diameter at the top, 212 inches at the bottom, and is 90.5 inches in height. It is configured for left-hand and right-hand assemblies, is fabricated using 2219 aluminum with D6ac steel rings, and weighs approximately 12,000 pounds.

C1(S).1.2.2.3.3 Forward Skirt. The forward skirt comprises all structure between the forward SRM segment and the ordnance ring. It includes an SRB/ET attach fitting which transfers the thrust loads to the ET and a forward bulkhead which seals the forward end of the skirt. The forward skirt provides the structure to react parachute loads during deployment and descent, and provides an attach point for towing the SRB during retrieval operations.

Secondary structure is provided for counting components of the ELectrical and Instrumentation (E&I) subsystem, rate gyro assemblies, range safety components, and interconnecting cables. The skirt assembly is sealed to provide additional flotation capability.

The forward skirt is 146 inches in diameter and 125 inches in height. It consists of a 2219 aluminum welded cylinder assembly made from machined and brake-formed skin panels and welded thrust post structure. The forward skirt weighs approximately 6400 pounds.

C1(S).1.2.2.3.4 Ordnance Ring. The ordnance ring, 146 inches in diameter, provides a plane for pyrotechnically separating the frustum from the forward skirt during the parachute deployment process. The ring is machined from a 2219 aluminum ring forging and provides mounting provisions for the linear-shaped charge used in the severance function.

C1(S).1.2.2.3.5 Frustum. The frustum houses the main parachutes, provides the structural support for the forward separation motors, and incorporates flotation devices and location aids (flashing light and Radio Frequency (RF) beacon) for water retrieval operations. It is fabricated using machined 2219 aluminum shear beams, ring fittings, separation motor supports, main parachute supports, and 7075 aluminum formed skins. The frustum weighs approximately 3800 pounds.

C1(S).1.2.2.3.6 Nose Cap. The nose cap houses both the pilot and drogue parachutes and is separated from the frustum by three pyrotechnic thrusters to initiate the parachute deployment sequence. The nose cap is basically an aluminum monocoque structure with a hemispherical section at the forward end. The base is 68 inches in diameter and the overall height is 35 inches. The structure is a riveted assembly of machined 2024 aluminum sheet skins, formed ring segments and cap, and a machined separation ring. Its weight is approximately 300 pounds.

C1(S).1.2.2.3.7 Systems Tunnel. The systems tunnel is located outboard on each SRB and houses the electrical cables and linear-shaped charge of the range safety system. The tunnel provides lightning, thermal, and aerodynamic protection and mechanical support for the cables and destruct charge. It is manufactured from 2219 aluminum and extends from the forward skirt along the motor case to the aft skirt. The tunnel is approximately 10 inches wide and 5 inches high. Its floor plate is vulcanized and bonded to each motor segment by Thiokol, the SRM contractor. The overall weight of the systems tunnel is approximately 600 pounds.

C1(S).1.2.3 Operation Cycle. This section describes the operation of the Shuttle Vehicle. One operation cycle extends from when the Orbiter lands to finish one mission until it lands again to finish the next one. The cycle covers ground preparation, launch, ascent, orbit, and return. Figure C1(S)-2 shows the entire cycle.

The ground preparations proceed in five phases. They are landing; safing, maintenance, and checkout; premate preparation; Shuttle assembly; and prelaunch.

The first actions after the Orbiter lands and which require about 1 hour to accomplish are the attachment of ground cooling and towing equipment and the removal of the flight crew. The Orbiter is then towed to the Orbiter Processing Facility (OPF) where the vehicle is safed. Thereafter, the OMS pods/RCS, returning payload, and OMS propellant kit are removed, and maintenance activity on the vehicle commences. The OMS/RCS are refurbished and reinstalled, the vehicle is checked out, the payload is installed and the vehicle/payload interfaces checked.

The vehicle is then moved to the Vehicle Assembly Building (VAB) where it is lifted, erected, and mated to the SRB/ET that were stacked and mated while the Orbiter was still in the OPF. When the Space Shuttle Vehicle and the mobile launch platform are ready to be moved, they can be moved or maintained in this configuration for a long period.

Loading the propellants into the ET will begin 8.5 hours before launch. As depicted in Figure C1(S)-22, this operation commences with line chilling and progresses to slow fill, fast fill, and topping. Because topping of the ET LH_2 and LOX tanks is essentially complete several hours before launch, it is necessary to keep replenishing these tanks until a few minutes before launch.

Approximately 3 hours before launch, the flight crew will enter the Orbiter. The ground control will end 31 seconds before launch when automatic sequencing of events from the Orbiter will begin.

The launch part of the operation cycle includes five principal events: ignition of the SSMEs; sending the signal to light the SRBs; lift-off; clearing the tower; and the passage of the instantaneous impact point beyond shallow water. The SSMEs are started by commands from the Orbiter onboard computers beginning at 6.6 seconds before lift-off, using a staggered start sequence (Engines 3, 2, and 1 started sequentially in that order at 120 millisecond intervals). When the computers in the Orbiter determine that all three SSMEs are operating normally, with at least 90% of rated thrust, the computers send the signals to start the SRBs. These signals are sent 40 milliseconds before lift-off. Once the SRBs light, lift-off will occur either the SRBs and the SSMEs running. After approximately 7 seconds of flight, the vehicle will clear the tower; and after approximately 34 seconds of flight, the instantaneous impact point will be over water at least 20 fathoms deep.

The ascent part of the operation cycle includes several important events for nuclear safety analysis. The dynamic pressure on the vehicle will grow quite rapidly to be about 650 lb/ft^2 after about 40 seconds of flight, will hold at a peak of approximately 705 lb/ft^2 between 50 and 60 seconds and then will decline sharply to about 50 lb/ft^2 by the time the SRBs burn out (about 119 seconds into the flight). Once the SRBs burn out (signaled by the drop of the SRM chamber pressure to below 50 lb/in^2), they will be jettisoned and will fall clear of the Orbiter and ET after about 128 seconds of flight. The SRBs are lowered by parachutes into the ocean and are recovered and returned to the launch site for recycle. After the SRBs fall away, the Orbiter and ET will continue, propelled by the SSMEs.

After about 500 seconds of flight, the SSMEs will be shut down; and soon afterwards, the ET will be released to break up and fall into the ocean. Approximately one or two minutes after MECO the OMS engines in the Orbiter will be started for a short burst to propel the vehicle to orbit altitude. After the orbiter has coasted to the desired altitude, it will be leveled using the RCS, and the OMS engines will be fired again to circularize the orbit.

The orbit part of the operation cycle may differ considerably from one mission to another. For some missions, the crew will deploy or retrieve a satellite and return to Earth in one revolution. For other missions, the Orbiter will stay in space as long as 30 days. For the Galileo and Ulysses missions, the orbital part will include releasing the spacecraft and the IUS upper stage, moving some distance away, staying in orbit for several hours while the crew eats and sleeps, and then preparing to return to Earth. The typical time profile for the IUS planetary missions from lift-off to cargo deployment is shown in Table C1(S)-1.

The return part of the operation cycle includes firing the OMS engines to retard the Orbiter out of orbit and gliding down for a landing.

Following the completion of orbital operations, the Orbiter is oriented to a tail-first attitude. After the OMS provides the deceleration thrust necessary for deorbiting, the Orbiter is reoriented nose-forward to the proper attitude for entry. The orientation of the Orbiter is established and maintained by the RCS down to where the atmospheric density is sufficient for the pitch and roll aerodynamic control surfaces to be effective (about 250,000 feet altitude and 26,000 feet per second velocity). The yaw RCS remains active until the vehicle reaches an angle of attack of about 10 degrees (about 80,000 feet altitude).

The Orbiter entry trajectory provides lateral flight range to the landing site and energy management for an unpowered landing. The trajectory, lateral range, and heating are controlled through the attitude of the vehicle by angle of attack and bank angle. The angle of attack is established at 38 degrees for the theoretical entry interfaces of 400,000 feet altitude. The entry flightpath angle is -1.19 degrees. The 38 degree attitude is held until the speed is reduced to 21,200 feet per second (about 220,000 feet altitude), is then reduced gradually to 28 degrees at 17,200 feet per second (about 190,000 feet altitude); it is held at 28 degrees until speed is reduced to 8500 feet persecond (about 150,000 feet altitude), and then reduced gradually to 6 degrees where the speed is about 1500 feet per second (about 70,000 feet altitude) at the beginning of Terminal Area Energy Management (TAEM).

During the final phase of descent, flightpath control is maintained by using the aerodynamic surfaces. TAEM is initiated to provide the proper vehicle approach to the runway with respect to position, energy, and heading. Final touchdown occurs at an angle of attack of about 16 degrees. The maximum landing speed for a 32,000 pound payload, including dispersions for hot day effects and tailwinds, is about 207 knots.

The parameters of the nominal entry just described may vary according to individual mission requirements.

C1(S).1.2.4 Propulsion. This section describes the engine for propulsion of the Orbiter. There are three engine systems to consider: SSMEs, SRBs and OMS. The SSMEs and the SRBs together provide the thrust for lift-off and for the first 2 minutes or so of ascent. After the SRBs burn out, the SSMEs thrust the Orbiter on, almost to orbit. After the SSMEs are shut down, the OMS engines provide thrust for attain in orbit, maneuvering while in orbit, and decelerating out of orbits.

C1(S).1.2.4.1 Main Engines. The SSME is a reusable, high-performance, liquid-propellant rocket engine with variable thrust. Three engine are clustered on each Orbiter vehicle. They are ignited on the ground at launch and burn for an average of eight minutes during the vehicle boost phase.

Figure C1(S)-23 shows the SSMEs and the propellant feedlines from the ET.

The propellants for the SSMEs are LH_2 (fuel) and LOX (oxidizer). The rated thrust of each SSME is 375,000 pounds at sea level and 470,000 pounds in vacuum. The nominal engine chamber pressure is 3000 lb/in^2. The nozzle exit diameter is 94 inches. Each SSME weighs about 6950 pounds. The nozzle is gimbaled, with hydraulic actuators, for steering.

C1(S).1.2.4.2 SRBs. The propellant for the SRBs is a composite-type solid propellant formulated of polybutadiene acrylic acid acrylonitrile terpolymer binder, ammonium perchlorate, and aluminum powder. A small amount of burning rate catalyst (iron oxide) is added to achieve the desired propellant burning rate. The propellant is in four segments; Figure C1(S)-21a shows the segments and their lengths. The propellant in the forward segment weighs 299,000 pounds; in each of the two center segments, 272,000 pounds; and in the aft segment, 266,000 pounds. The total propellant weight in both SRBs is 2,218,000 pounds. Each SRB provides 2.9 million pounds of thrust at sea level.

The higher thrust level required during the lift-off portion of the Shuttle flight results from increased burning surface provided by the 11-point star configuration in the forward segment. After the lift-off portion of the flight, the thrust is reduced with burnout of the starred section. The thrust progress is slighter after 52 seconds. Reduced vehicle acceleration is achieved by burning out the aft most portion of the aft segment and the programmed burnout of slivers in all four segments.

The nozzle of the SRM is gimbaled for steering with hydraulic actuators. The hydraulic power supply for the TVC subsystem consists of hydrazine-fueled APUs on each SRB. The SRB TVC subsystem works in conjunction with the TVC system for the SSMEs and provides the preponderance of gimbal authority for the Shuttle during the first stage flight. Pitch, roll, and yaw commands are provided by the Orbiter flight control system.

C1(S).1.2.4.3 Orbital Maneuvering Subsystem Engines. The OMS engines use the hypergolic propellants MMH (fuel) and N_2O_4 (oxidizer). There are two OMS engines on the Orbiter. They are in pods at the top of the aft fuselage, one on each side of the vertical tail. The nominal thrust of each engine is 6000 pounds. The nominal engine chamber pressure is 125 lb/in^2. The nozzles are gimbaled for steering, with electromechanical actuators. The OMS propellant tanks in the two pods can carry enough propellant for a change in velocity of 1,000 feet per second when the vehicle carries a payload of 65,000 pounds. Crossfeedlines connect the propellant tanks in the two OMS pods, so that propellants from either pod can run either engine. Also, the RCS thrusters can burn propellants from the OMS tanks. Figure shows an OMS engine pod, which also contains parts of the RCS.

C1(S).1.2.5 SRB/ET and ET/Orbiter Separation. This section describes the mechanisms for separating the SRBs from the ET and the ET from the Orbiter. On a nominal flight, the SRBs will separate from the tank about 2 minutes after lift-off, and the tank will separate from the orbiter about 9 minutes after lift-off.

Basically, the way of separating the SRBs from the ET is to fracture four separation bolts on each SRB and to propel the SRBs from the ET by eight solid rocket thrusters on each SRB. Figure C1(S)-25 shows the separation bolts and separation thrusters.

The primary mode for initiating the SRB separation sequence is accomplished when the chamber pressure of both SRBs is equal to or less than 50 psig. Separation cue monitoring is initiated at To+105 seconds, and the Orbiter General Purpose Computers (GPCs) will issue an SRB separation cue plurality signal when the voting logic has detected two pressure cue signals from both the left and right SRBs/ A backup cue is used if the primary separation cue should malfunction. This backup cue is based upon mission elapsed time of approximately 130.6 seconds from lift-off. SRB release (mechanical and electrical) is initiated 4.0 seconds after start of the separation sequence if the separation conditions criteria of dynamic pressure, roll rate, pitch rate and yaw rate are not exceeded; otherwise the separation is inhibited until the required conditions are within acceptable limits. The crew has the capability of overriding the inhibits. Manual separation can be achieved if either the primary separation cue (two chamber pressure cue signals from each of the left and right SRBs) or backup separation cue (mission elapsed time of approximately 130.6 seconds from lift-off) has been achieved.

The forward attachments between the SRB and the ET are mated fittings held together by a separation bolt. At separation, National Aeronautics and Space Administration (NASA)-standard initiator pressure cartridges at the ends of the separation bolt fire, and the resulting pressure wave drives pistons to break the bolt. Operations of either cartridge on the bolt will break it. The aft attachments are similar, and they work in a similar way. There are three separation bolts in the aft attachments.

The solid propellant thrusters, four forward and four aft on each SRB, propel the SRBs away from the ET. Each solid propellant thruster gives a 23,000-pound thrust.

The method used to separate the Orbiter from the ET is to fracture one bolt and eight nuts to sever structural and umbilical connections, and to use the RCS to propel the Orbiter away from the tank. Figure C1(S)-26 shows the separation mechanism.

The forward attachment between the Orbiter and the ET is a spherical bearing with an attach-bolt. Firing either of two pressure cartridges on the bolt will break it. There are two structural attachments aft, one on the left and one on the right. Firing either of two pressure cartridges on each attach-bolt will break the frangible nut and release the bolt. There are two umbilical attachments aft, one on the left and one on the right. Each attachments contains three attach-bolts, and firing either of two pressure cartridges on each bolt will break the frangible nut and release the bolt.

C1(S).1.2.6 <u>Pressure Tanks</u>. This section describes the pressure tanks in the Orbiter, the ET, and the SRBs. The Orbiter carries many pressure tanks. Table C1(S)-2 lists them, and Figure C1(S)-29 shows their locations. The index numbers on the table refer to the circled numbers on the figure.

The ET carries two pressure tanks, which together essentially are the ET. Each one by far dwarfs all the tanks in the Orbiter. The LOX tank, at the forward end of the ET, is made of aluminum alloy. Its volume is 19,500 ft^3, and its maximum relief pressure is 37 lb/in^2. Besides the two tanks, there are feedlines carrying propellants under pressure, which run on the outside of the ET on the side toward the Orbiter. Figure C1(s)-15 shows the two tanks of the ET.

Each SRB carries several pressure tanks, which are parts of the hydraulic power supply systems of the SRB. Two tanks on each SRB hold hydrazine, the fuel for the APUs. Figure C1(S)-30 shows a hydrazine tank on an SRB as part of the TVC subsystem, and Figure C1(S)-16 shows the location of the TVC subsystems on an SRB.

C1(S).1.2.7 Orbiter Reaction Control Subsystem. This section describes the RCS of the Orbiter.

The RCS employs 38 bipropellant primary thrusters and 6 vernier thrusters to provide attitude control and three-axis translation during the orbit insertion, on-orbit, and entry phases of flight.

The RCS consists of three propulsion units, one in the forward module and one in each of the aft propulsion pods. All modules are used for ET separation, orbit insertion, and orbital maneuvers. Only the aft RCS modules are used for entry attitude control.

The RCS propellants are N_2O_4 and MMH. The design mixture ratio of propellants allows the use of identically sized propellant tanks for both fuel and oxidizer. The nominal propellant capacity of the tanks in each module is 928 pounds of MMH and 1477 pounds of N_2O_4. An interconnect between the OMS and the RCS in the aft pods permits the use of OMS propellant by the RCS for on-orbit maneuvers. In addition, the interconnect can be used for crossfeeding propellants between the right and left hand RCS pods.

The thrust of each primary thruster is 870 pounds (in vacuum), and the thrust of each vernier thruster is 24 pounds (in vacuum). Figure C1(S)-31 shows the RCS.

C1(S).1.3 KSC LAUNCH SITE

The Space Shuttle launch site at KSC is described in this section. Figure C1(S)-32 is a map of KSC and Cape Canaveral Air Force Station (CCAFS). The map shows the runway for Orbiter landings; the VAB where the Shuttle will be mated on a mobile launch platform; and launch pads 39A and 39B. These two launch pads are shown in greater detail in Figures C1(S)-33 and C1(S)-34. Figure C1(S)-35 gives a detailed view of the area around the VAB.

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Table C1(S)-1 Typical IUS/Planetary Mission Event Timeline

Event	Time (Hr:Min:S)
SSME Ignition	-0:00:06.6
SRB Ignition Command	0:00:00
First Motion/Lift-Off	0:00:00.24
Begin Roll Program (V=125 fps)	0:00:07
End Roll Program (V=300 fps)	0:00:15
Begin First Throttle Down (V=428 fps)	0:00:20
Begin Second Throttle Down (V=716 fps) (Optional)	0:00:31
Begin Throttle Up (V=1599 fps)	0:01:08
SRB Separation	0:02:05
3G Throttling Begins	0:07:39
MECO Command	0:08:34
Zero Thrust	0:08:40
ET Separation	0:08:52
OMS-1 TIG	0:10:34
OMS-1 Cutoff	0:13:22
OMS-2 TIG	0:46:10
OMS-2 Cutoff	0:48:29
Open Payload Bay Doors	1:15:00
IMU Realinement	3:00:00
Meal	4:00:00
IUS/Spacecraft Checkout	5:00:00
IUS Deploy	6:40:00
IUS Engine Ignition	7:20:00

Table C1(S)-2. Pressure Tanks in Orbiter (next 4 pages)

Index Number(a)	Contents	Subsystem	Maximum Operating Pressure, lb/in^2g	Volume in^3	No. of Tanks	Tank Material
1	Air (Crew Cabin)	ECLSS	15	4,363,000	1	Aluminum Alloy
2	Ammonia	ECLSS	550	3,024		Titanium
3	Breathing Oxygen (Not on OV-099 or OV-104 Mission Kit only)	ECLSS	3300	8,181	1	Inconel with Kevlar Winding
4	Freon-21 ($CHCl_2F$)	ECLSS	230	4,337	2	Aluminum
5	Helium	FRCS	4000	3,008	2	Titanium with Kevlar Winding
		ARCS	4000	3,008	2/Pod	Titanium with Kevlar Winding
		OMS	4875	30,033	1/Pod	Titanium with Kevlar Winding
		MPS	4500	29,894	7	Titanium with Kevlar Winding
		MPS	4500	8,122	3	Titanium with Kevlar Winding

(a) Index numbers refer to circled numbers on Figure CI(S)-29
(b) Also in main-engine feed lines.

Table C1(S)-2. Pressure Tanks in Orbiter (Cont.)

Index Number(a)	Contents	Subsystem	Maximum Operating Pressure, lb/in2g	Volume in3	No. of Tanks	Tank Material
		MPS	850	500	2	Titanium
6	Hydrazine (N_2H_4)	APU	355	11.375	3	Titanium
7	Hydraulic Oil	HYD	3000	98	3	Steel
		HYD	135	1,850	3	Aluminum
		HYD	70	98	3	Steel
8	Liquid Hydrogen (LH_2)(b)	PRSD	320	36,972	3-5	Aluminum alloy pressure vessel and outer shell
9	Liquid Oxygen (LOX)(b)	PRSD	1035	19,418	3-5	Inconel pressure vessel and aluminum alloy outer shell
10	Monomethyl-Hydrazine - ($N_2H_3CH_3$)	FRCS	350	29,376	1	Titanium
		ARCS	350	29,376	1/Pod	Titanium
		OMS	313	155,520	1/Pod	Titanium
11	Nitrogen	ECLSS	3600	8,181	4	Titanium with Kevar Winding

(a) Index numbers refer to circled numbers on Figure CI(S)-29
(b) Also in main-engine feed lines.

Table C1(S)-2. Pressure Tanks in Orbiter (Cont.)

Index Number(a)	Contents	Subsystem	Maximum Operating Pressure, lb/in^2g	Volume in^3	No. of Tanks	Tank Material
		HYD	3165	113	3	Titanium
		OMS (Eng)	3000	60	2	Titanium
		OMS (Eng)	450	19	2	Titanium
12	Nitrogen Tetroxide (N_2O_4)	FRCS	350	28,512	1	Titanium
		ARCS	350	28,512	1/Pod	Titanium
13	Water	APU	35	2,352	1	Titanium
		APU	100	435	1	Titanium
		HYD	33	4,546	3	Aluminum
14	Water (Potable, Waste, & Heat Transfer)	ECLSS/ ATCS	20	6,600	5	Aluminum
15	Oxygen	SSME (POGO ACCUM)				
16	Halon 1301	Fire Ext (Portable)	575	80	3	Steel

(a) Index numbers refer to circled numbers on Figure CI(S)-29
(b) Also in main-engine feed lines.

Table C1(S)-2. Pressure Tanks in Orbiter (Cont.)

Index Number(a)	Contents	Subsystem	Maximum Operating Pressure, lb/in^{2}g	Volume in^3	No. of Tanks	Tank Material
		Fire Ext (Fixed)	575	87	3	Stainless Steel
17	Water/O_2	EMU	15	227	3	Aluminum
18	O_2	EMU	1050	247	6	Stainless Steel
		EMU	7400	92	6	Inco 718

(a) Index numbers refer to circled numbers on Figure C1(S)-29
(b) Also in main-engine feed lines.

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CI(S). 2.0 SAFETY PLANS

The purpose of this section is to describe the plans to protect personnel and equipment from the consequences of STS failures. Four major subsections cover the safety plans for the range, launch/landing site, Orbiter and contingencies. This section concludes with STS mission/launch rules and the list of references used. The references should be consulted if further details are required.

CI(S). 2.1 Range Safety

CI(S). 2.1.1 Introduction

Range Safety is intrinsic to all missions flown on a National Range such as the Eastern Space and Missile Center (ESMC) (Reference 1). It encompasses all safety activities from design through test, launch, and vehicle flight. This section deals primarily with that portion of Range Safety having to do with launch and vehicle flight. The flight portion safety goal is to control and contain the flight of all vehicles, precluding impact of intact vehicles or pieces from an abort which could endanger human life, cause damage to property or result in embarrassment to the U. S. Government. Although the risk can never be eliminated completely, Range Safety attempts to minimize the risks while not unduly restricting the probability of mission success.

CI(S). 2.1.2 Ground Range Safety Systems (RSS)

All launches carry a flight termination systems which, in conjunction with the ground transmitter and tracking system, allows the Range Safety Officer to control launch vehicle flight by shutting down thrusting engines or destroying a vehicle if it violates established flight destruct criteria. The ground systems' purpose is to ensure an operating environment that permits maximum flexibility in achieving flight objectives while reducing the risk to personnel, property, and the flight vehicle to an acceptable level. The Ground RSS, which provides the Range Safety Officer (RSO) with the capability for monitoring launches of the Shuttle, is comprised of the following:

a. Range Safety Display System (RSDS).

b. Command/Control System.

c. Vertical Wire Skyscreen (VWS).

d. Range Safety Closed-Circuit Television (TV).

e. Real Time Telemetry Display System.

f. Communications and Data Support Systems.

CI(S) 2.1.3 Flight Vehicle Range Safety System.

a. Each flight vehicle carries a Range Safety Flight Termination System (FTS) which, when activated from a ground signal, can shut down thrusting engines (not applicable to the STS) and/or destroy the vehicle.

b. The Space Shuttle FTS allows the intentional destruction of the SRBs and ET if the flight deviates too far from the nominal. On command from the Range Safety Officer, Linear Shaped Charges (LSC) split the two tanks in the ET as well as the cases of the SRMs.

The onboard systems for the three elements (one ET and two SRBs) are connected so that, if either SRB receives a destruct, all three receive it. The system in each element is redundant (two parallel LSCs) to assure reliability.

Electronically, each of the SRBs and the ET carry a command destruct receiver which will receive the arm and destruct signal from the ground system. Electronically, the three independent systems are cross-strapped so that a signal received by one can initiate destruct action not he other two systems as long as normal separation has not occurred. The SRBs carry two receivers each and the ET has one. After inadvertent separation of one or both of the SRBs, destruct action could still be initiated on the separated solids and the ET, unless the system is damaged. All three would receive the signal and respond, unless SRB separation is normal, in which case the destruct systems on the SRBs are safed prior to separation.

On the ET, one LSC is on the LOX tank, and the other is on the LH_2 tank. The charges sit in the cable trays no the outside of the tanks. The charge of the LOX tank is about 8 feet long; for the LH_2 tank it is about 20 feet long, in two 10-foot sections. The destruct charges are on the side next to the Orbiter.

On each SRB, the LSC is about 80 feet long, in six 160-inch sections on the upper three segments. The destruct charges sit in the cable trays on the outside of the booster case on the side away from the ET. The LSCs are initiated (top down) to split open the 0.5" thick steel case so that the top three segments rapidly open up and depressurize (in 10 to 20 ms). As observed in recent accidents and accompanying SRB destruct actions (STS 51L and Titan 34D-9), this causes the case to "clam" open and accelerate small and large fragments to hundreds of feet per second away from the LSC-cut-line and into the ET.

Based on past experience and the combined functioning of the ground and flight portions of the Safety System, a delay of at least 4-1/2 seconds will occur between the time a shuttle vehicle could require destruct action and when the destruct event actually occurs. With a crew on board, it is unlikely that this time could be cut appreciably except under very unusual circumstances where the Range Safety Officer (RSO) could visually determine that the launch vehicle was breaking up. In this mode, it is possible that up to a second and a half could be cut from the delay time.

For many failure modes, such as ET aft dome failure, Orbiter main engine fires, SRB case burn through (not explosion), etc., there should be enough time for Range Safety action to occur. SRB case pressure rupture, SSME engine compartment or ET innertank explosion will probably lead to ET disintegration and destruction before range destruct action can be taken.

Therefore, if the vehicle has not already disintegrated, most of the more probable Shuttle failure modes will allow time for Range Safety action prior to ground impact.

CI(S). 2.2 Launch/Landing Site Safety

This subsection describes certain salient systems/efforts which contribute to launch/landing site safety of STS missions originating at KSC. These include the launch processing system, the sound suppression water system, Gaseous Nitrogen (GN^2) inerting system, meteorological systems, launch precautions, fire detection and suppression systems, hazardous gas detection and launch countdown abort procedures.

CI(S) 2.2.1 Launch Processing System

Launch of the STS-1 on April 12, 1981 marked the first mission use of the Launch Processing System (LPS). The LPS is a high-speed, digital, computer operated checkout system used to support test, checkout, launch control, and operational management of launch site ground operations. LPS consists of three subsystems: (1) Checkout, Control and Monitor Subsystem (CCMS) (2) Central Data Subsystem (CDS) and (3) Record and Playback Subsystem (RPS). CDS and RPS provide support for CCMS testing in the Firing Rooms. Located in the Launch Control Center (LCC), the LPS was responsible for most of the orbiter, tank, and booster checkout. Specific technology accomplishments in CCMS that enabled successful implementation of the LPS concept include development of the Subsystem Operator Consoles, the Common Data Buffer (CDBFR), the Processed Data Recorder (PDR) and Shared Peripheral Area (SPA), and the Ground Operations Aerospace Language (GOAL).

CI(S) 2.2.1.1 Subsystem Operator Consoles.

Each subsystem operator position in a firing room has its own keyboard and visual display system. EAch group of three keyboard and display systems is considered a "console", and operates as a unit. Each console can perform several independent tests or GOAL procedures simultaneously. For major integrated tests, all consoles are coordinated to work together by an integration console located in the rear of the firing room. Each console has a small computer and on-line disk storage capacity of 80 million words, and can hold all CCMS application procedures to be executed by its operator.

Checkout and launch functions of each console can be changed, if necessary, by reloading data from the bulk disk via the Master Console. This provides flexibility and redundant capability. In addition, the independence of each console allows maximum parallel testing, thus saving serial time.

LPS monitors thousands of measurements on the vehicle and ground support equipment simultaneously, and compares them to predefined tolerance levels. Measurements which are displayed are defined by the test engineer in the application program or through the keyboard. Some measurements will be displayed only when they go out of tolerance. This is termed "exception monitor capability". The monitor feeds outputs to console screens to display, in various predefined colors, conditions that must be evaluated by the test engineer. In many cases, these computers will automatically react to "exception" conditions and perform safing or other related functions without test engineer intervention.

CI(S) 2.2.1.2 Common Data Buffer (CDBFR)

The CDBFR is the communication center for CCMS. All communications from one computer to another computer pass through the CDBFR. The CDBFR is sequenced so that each of the computers in the network appears to be the only computer using the buffer. The CDBFR can communicate with 64 computers while temporarily storing data, messages, and commands along with error-correcting information. It assures proper retrieval of this information even when a portion of the buffer is incperative as a result of failed hardware. This assembly is transparent to the user.

Error Correction Code (ECC) is an absolute requirement to maintain system integrity. An error is defined as a bit transposition from a 1 to a 0 or vice versa. ECC is an interleaved, modified cyclic code developed from Bose, Chaudhoni, Hocquenghen (BCH) code, called cyclic alpha squared code. CDBFR hardware design requires ECC to perform the following four functions.

a. Correct single random bit errors, detect random double bit.

b. Correct at lease one of any possible double adjacent bit errors a hardware boundary or even-odd bits.

c. Detect at least two of all possible double adjacent bit errors a hardware boundary of even-odd bits.

d. Detect the all 1's and all 0's as uncorrectable errors.

Address errors are always checked by the CDBFR hardware. A CDBFR failure in the data or command path will be either corrected or detected. Corrected and detected errors generate messages to the console operators. The Address Validity Module checks all write instruction addresses against the predefined limits of each CDBFR port. An uncorrectable or out of limits address causes a write operation to abort and an error condition to be posted to the responsible computer. Single buffer failures can bring a segment of the CDBFR to a halt. A failure of the Stack Control Card prevents computer-to-computer interrupts. This condition prevents a console from issuing a request for a command to a Front End Processor (FEP), but does not stop data from being written into the CDBFR. A complete Master Scanner failure causes the entire CDBFR to become inoperable.

CI(S) 2.2.1.3 PDR/SPA

The processed Data Recorder (PDR) subsystem accomplishes data storage for realtime and post-test data retrieval. It retains about 30 minutes of test data on bulk disk for near-realtime retrieval on request from a console. The amount of data recorded to disk varies with the test activity. If there is low test activity, the disk could contain several hours of data, while high test activity will cause as little as fifteen minutes of data to be recorded. The CDBFR high-speed First In, First Out (FIFO) buffer sends data to the PDR low-speed buffer via the external Direct Memory Processor. Data is then transferred from the PDR to bulk disk. Information sent to disk is simultaneously buffered out to tape as a permanent copy for post-test data processing.

The Shared Peripheral Area (SPA) subsystem is used for near-realtime data playback. A console requests data from the SPA enabling it to perform backup recording functions if the PDR breaks down. CDS supplements data recording functions with the Test Data Recording and Retrieval (TDR) subsystem. The CDS On-Line Data Bank (OLDB) provides history retrieval of all measurements in near-realtime.

CI(S) 2.2.1.4 Ground Operations Aerospace Language

Applications programming of the Checkout, Control, and Monitoring computers used for the STS mission is accomplished using a KSC developed high-order computer language, known as GOAL. This usage permits computer programs to be compiled from functional statements as they would appear in an English language test procedure. Software engineering writes the application programs while hardware engineering verifies the application programs to ensure changes were implemented properly. Verification of application programs is performed in the Ground Software Production Facility (GSPF) using a Software Verification Procedure (SVP), similar to running an Operations and Maintenance Instruction (OMI) during hardware testing. Upon completion of verification, the as-run SVP is impounded by Software Quality Assurance (SQA). When SQA has completed a review of the SVP and requirements, the application program is considered verified and released to hardware.

CI(S) 2.2.2 Sound Suppression Water System

A sound suppression water system has been installed on the pad to protect the Orbiter and payload from damage by acoustical energy reflected from the Mobile Launcher Platform (MLP) during launch. The orbiter and payload in the payload bay is much closer to the surface of the MLP than the Apollo spacecraft was at the top of a Saturn V or Saturn IB rocket.

The system consists of an elevated tank, a valve complex, a piping system, and spray nozzles. The valve complex consists of six 48-inch butterfly valves to remotely control the water flow. Each valve is actuated by a piston-type, double cyclinder hydro/pneumatic actuator. To rapidly establish a minimum flow of 500,000 gpm through the pre-liftoff system, and 400,000 gpm through the post-liftoff system, at least two pre-liftoff and two post-liftoff valves must be fully opened within 4 seconds after receiving opening commands. The peak rate of flow from all sources is 900,000 gallons of water per minute at 9 seconds after liftoff.

Acoustical levels reach their peak when the Space Shuttle is about 300 feet above the platform, and cease to be a problem at an altitude of about 1,000 feet.

The Sound Suppression System (SSS) has successfully supported all KSC launches with no significant failures of the SSS. The SSS has suppressed any possible damage to the vehicle or to the payloads from the acoustic vibration created by the vehicle main engines. If the SSS fails before main engine start (which would be detected by the Ground Launch Sequencer software), the launch would be scrubbed. If the SSS fails after main engine start and before liftoff, possible damage would occur to the vehicle or to the payload. There is very low probability of this occurring.

CI(S) 2.2.3 Pad Environmental Control System (ECS) GN_2 Subsystem

The GN_2 subsystem is used at the pad to inert the orbiter with GN_2 during critical tanking operations. The system interfaces with the air system by way of isolation valves to prevent GN_2 from entering the system inadvertently and to isolate GN_2 from the cabin ECS duct. There are a minimum of two isolation valves between the GN_2 and air supply lines and the orbiter is verified clear of personnel prior to switching from air to GN_2. Software safeguards are also incorporated into the system.

The GN_2 is used to inert the orbiter forward, payload bay, and aft compartments through an onboard duct system consisting of three primary supply lines. The orbiter environment is normally sustained with conditioned air. However during fuel cell tanking and pre-launch tanking of the ET, the orbiter purge is switched from air to GN_2 for inerting for fire prevention if a leak of cryogens should occur. The second purpose is to provide a background atmosphere required by the hazardous gas detection system to monitor cryogenic systems for leaks. The GN_2 system is also used in "standby" mode to provide GN_2 inerting to the orbiter during hypergol servicing in case a spill or leak of fuel should occur. During the final countdown, GN_2 flow is started about two hours prior to cryogenic loading and continues through vehicle launch.

CI(S) 2.2.4 Meteorological Systems

In addition to standard meteorological data, the following special meteorological and instrumentation data processing systems are utilized by a round-the-clock team of Air Force meteorologists at the Cape Canaveral Forecast Facility (CCFF) to provide required meteorological support.

A WSR-74C weather radar has been modified to provide volumetric scan output, including constant altitude plan position indicators, vertical cuts, echo top maps, and extrapolated short range forecasts.

A Geostationary Operational Environmental Satellite System (GOES) ground station receives visible and infrared imagery for display on the Meteorological Interactive Data Display System (MIDDS).

The MIDDS provides computerized integration of the data sources available to the CCFF into a single data base where the various types of data can be molded and displayed together for forecaster use.

The Weather Information Network Display System (WINDS) provides data every 5 minutes from 16 instrumented towers, 9 temporary wind towers, and other special locations. Parameters measured include wind speed, direction, and variation, temperature, and dew point.

The Launch Pad Lightning Warning System (LPLWS) was designed to assist forecasters in monitoring impending and current atmospheric electrical activity. Data are sensed by 34 field mills located at the CCAFS and KSC. The field mills measure the vertical component of the atmospheric potential gradient at ground level. The system provides displays of the static field during periods without lightning. During lightning activity, the LPLWS program outputs the charge centers or source of the lightning activity in the cloud.

The Meteorological and Range Safety Support (MARS) system provides a user within seconds of a toxic spill, a forecast of the direction and distance that the material will take and the toxic corridor for evacuation to personnel. In put to the MARS is from the WINDS, the model used is the Ocean Breeze - Dry Gulch (OBDG).

Various computer models utilize meteorological input information to provide data to safety personnel. In addition to the OBDG for toxic diffusion, routine models in use include the BLAST for shock waves from inadvertent detonations and the REEDM for rocket exhaust diffusion. For the Galileo and Ulysses launches, a contractor is scheduled to run the EMERGE model to forecast radiological dispersion in case of an accident.

CI(S) 2.2.5 STS Launch Propellant Loading Cleared Areas

CI(S) 2.2.5.1 Blast Danger Area Controls

A Blast Danger Area, 1367 m (4485 ft) radius around the launch pad (Figure C1(S)-36) is established from the time liquid hydrogen and liquid oxygen are loaded onto the STS vehicle (approximately launch minus 8 hours). Only critical launch crews (i.e., flight crew, close-out crew, and ice inspection team) are permitted inside this area and only the flight crew is allowed within the area at launch.

CI(S) 2.2.5.2 Impact Limit Line Controls

In addition, a Launch Impact Limit Line (Figure C1(S)-36) is established prior to launch minus 30 minutes. For Launch Complex (LC)-39A, it extends from the launch complex to 4572 m (15,000 ft) west of the launch pad. Approximately 90 personnel are located inside the Impact Limit Line supporting the launch and they are provided protective equipment (communications, egress vehicle, gas mask). The Impact Limit Line for LC-39B is the same line as that established for LC-39A.

CI(S) 2.2.6 Fire Detection and Fire Suppression

The Fixed Service Structure (FSS), Rotating Service Structure (RSS), and Payload Changeout Room (PCR) are the three primary areas of fire concern. Hydrogen fire and leak detection on the FSS and Mobile Launch Platform (MLP) are also important. In general, combustible materials are prohibited on the launch structures. Any combustible or flammable materials/fluids that are required in preparing the STS for flight are restricted to those type/quantities that are required to do an 8-hour task.

The Fire Detection Systems are designed to provide early warning and alarm to the Fire Services organization. If an alarm is received during a major test, the test team is immediately notified and decides test related corrective measures.

Launch complex deluge systems are designed to extinguish hydrazine type fires and to prevent a fire, originating on a work platform or structure, from transmitting to the launch vehicle. In addition, egress ways are provided with a lower density coverage egress spray. Alarm and deluge system are further discussed in subsequent paragraphs.

CI(S) 2.2.6.1 Fixed Service Structure

a. FSS Fire Detection - A manual system comprised of Fire Alarm Stations is provided along the egress route of each level, including the Orbiter Access Arm (OAA). Automation detection is provided from heat activated detectors in the White Room, Elevator Machine Room, and Hammerhead Crane. Activation of any fire alarm device will ring fire alarms bells throughout the FSS and RSS structure and report to the Fire Alarm Control Panel located inside the main entrance to the Pad Terminal Communication Room (PTCR). From the PTCR, a fire alarm is transmitted to Safety (SF) Control.

b. FSS Fire Suppression - Electrically operated manual deluge are provided for the following areas:

   1. 75' Level - Hyper Fuel and Oxidizer Scrubbers

   2. 155' Level - LOX and LH_2 Dewars

   3. 195' Level - Orbiter Access Arm Egress

   4. 195' Level - ET Vent and Hinge Arm

   5. 255' Level - ET Gaseous Oxygen Vent Arm

All levels can be activated remotely from the Launch Control Center (LCC), and are activated automatically by the Ground Launch Sequencer during launch countdown (except OAA Egress). In addition, the 75', 155', and 195' (OAA Egress) levels can be activated locally or from the Master Deluge Panel on the launch pad surface. No fire alarm signals are initiated by water flow on the FSS. Fire hose reels are provided for first-aid firefighting on all levels.

CI(S) 2.2.6.2 <u>Rotating Service Structure</u>

a. Rotating Service Structure Fire Detection - Manual Pull Stations are provided along all emergency egress routes leading from various levels. In addition, automatic detection is provided from heat actuated detectors in the Elevator Machine Room, Air Handling Units (AHU) rooms (107' level), and on the 120' level. Actuation of any fire alarm device will ring bells throughout the FSS and the Rotating Service Structure, report to the Fire Alarm Control Panel inside the PTCR, and transmit a signal to SF control.

b. Rotating Service Structure Fire Suppression - Separate manually operated deluge systems are provided for the Valve Sleds not he 107' and 207' levels -- both levels have a Valve Sled on Side 2 and Side 4 of the structure. In addition, deluge protection is provided for the Rotating Service Structure Room located at the 207' level. Dual in-line manually operated valves are installed for initiation of water flow. Each system is provided with a pressure switch for fire alarm reporting. Fire hose reels fed from the portable water supply are installed on the 107', 120', 130' and 160' levels.

CI(S) 2.2.6.3 <u>Payload Changeout Room</u>

a. PCR Fire Detection. Manual Pull Stations are provided at the main entrances to the PCR on the 130' level. These Pull Stations are tied into the Rotating Service Structure fire alarm system and perform in the same manner as the Rotating Service Structure fire alarm devices.

b. PCR Fire Suppression. A manually activated deluge system is provided on each platform level. Flow initiated on any level will also cause water to flow from the ceiling nozzles located at elevation 207'. Dual in-line ball valves are incorporated into the design at each activation station to prevent inadvertent water flow. A pressure switch is installed on the 182' level for fire alarm reporting. Fire hose reels are installed on each level, except the 130' level, for first-aid firefighting purposes. These hose reels are fed from the Firex writer distribution system.

CI(S) 2.2.6.4 Hydrogen Fire and Leak Detection

Hydrogen fire and leak detectors are strategically located on the FSS and mobile launch platform. Activation of any detector transmits a signal to the Firing Room. No fire alarm signal is initiated.

CI(S) 2.2.7 Hazardous Gas Detection System (HGDS)

The hazardous gas tection system is used to detect leaking hydrogen or oxygen in various compartments of the Shuttle and hydrogen Tail Service Mast (TSM). The HGDS is made up of three subsystems: a mass spectrometer which detects what gases are present; a sampling subsystem which draws samples from various locations on the Shuttle to the mass spectrometer for analysis; and a control subsystem which directs the sampling and presents the data to the console operator. The Shuttle areas which can be monitored are the ET intertank, the payload bay (at the aft bulkhead), the LH_2 tail service mast, the Orbiter lower midbody, and the Orbiter aft fuselage. The HGDS can sample only one area at a time but can complete a sample from area to area every 24 seconds if necessary.

CI(S) 2.2.8 Launch Countdown Abort Modes

The countdown procedure, OMI S0007, contains three preplanned abort modes during the critical time frame from Shuttle main engine start through lift-off. These aborts; RSLS abort; Backup Flight System (BFS) engage; and SRB holdfire are described below:

CI(S) 2.2.8.1 Redundant Set Launch Sequencer (RSLS) abort can occur any time after Shuttle main engine start command issued by the RSLS software through T-0. The abort can be initiated by flight or ground software surveillance of specific parameters or by operator action at the Ground Launch Sequencer (GLS) console or NTD console.

The abort will initiate GLS safing functions. If fire detectors on the MLP zero level detect fire, the GLS software will automatically initiate firex water for the base heat shield.

Significant GLS safing functions include Orbiter Access Arm extension, SRB ignition safing and other necessary functions to establish a safe configuration. The launch team can also exercise procedural options for safing if necessary. After stable conditions are established, a recycle is implemented by procedure and flight crew normal egress effected.

CI(S) 2.2.8.2 - BFS engaged abort - a condition where the Primary Avionics Support System (PASS) GPCs (General Purpose Computers) have halted and the Backup Flight System (BFS) GPC which cannot support launch is controlling. The launch team must manually take the following actions in sequence:

a. SRB power down

b. OAA extend

c. Main engine shut down

d. APU shutdown

e. GPC and launch data buss recovery

After the above actions normal safing and recycle can be performed.

CI(S) 2.2.8.3 SRB Holdfire. Occurs if T-0 is reached and SRB ignition did not occur. The safing action is basically similar to the BFS engage abort response.

CI(S) 2.3 Orbiter Safety

This section describes plans to maintain Orbiter and cargo safety. The Orbiter has been designed to recover safely from many failures. The crew can fly the Orbiter back to Earth after certain types of failures which would doom a conventional, expendable launch vehicle. A basic requirement for the Shuttle has been for it to withstand a failure and yet be just as operational as if the failure had not occurred. It should also be able to tolerate a second failure in a flight--critical system and still return safely to Earth. There are exceptions to this requirement as noted in the Orbiter Critical Items List (CIL). The following description of Orbiter safety plans covers four exigencies: intact aborts, contingency aborts, loss of critical functions, and fast separation.

CI(S) 2.3.1 Intact Aborts

The Space Shuttle Vehicle is required to have an intact abort capability for specific failures which might occur during the powered ascent flight phase (liftoff to post-MECO OMS insertion). Intact abort is defined as safely returning the Orbiter, crew, and cargo to suitable landing site. The specific failures which will result in an intact abort are the following:

a. Complete or partial loss of thrust from one SSME

b. Loss of thrust from one OMS engine

c. Loss of two electrical power buses

d. Failure of two auxiliary power units

e. Failure of some life support equipment

These failures are considered singly without combinations.

Four basic abort modes have been developed to provide continuous intact abort capability during the ascent phase: Return-To-Launch-Site (RTLS), Transatlantic Abort Landing (TAL), Abort-Once-Around (AOA), and Abort-to-Orbit

(ATO). The four modes are available during different segments of the ascent flight to provide intact abort capability. Figure CI(S)-37 is an altitude/range profile showing the relationship between RTLS, TAL, and AOA aborts. Figures CI(S)-38 and CI(S)-39 show the flight profile of an AOA and ATO, respectively. Figure CI(S)-40 shows the overlapping abort regions for a typical Shuttle mission with an SSME failure.

CI(S) 2.3.1.1 RTLS. The RTLS abort mode will be used in the event of an SSME failure occurring between liftoff and the end of the predetemined RTLS interval. However, when an overlap occurs between successive abort mode intervals, either abort mode could be selected depending upon abort failure conditions. The RTLS abort mode is selected after separation from the SRBs (second stage flight) even though the SSME failure might have occurred in first stage flight; the vehicle will continue accelerating downrange with the two remaining SSMEs until the MPS propellant equals the amount required to reverse the direction of flight and provide acceptable Orbiter/ET separation conditions, acceptable ET impact location, and acceptable range to permit gliding safely back to the selected landing site. A typical earliest (liftoff) and latest RTLS abort sequence of events are defined in Table CI(S)-3. Figure CI(S)-41 is a altitude/time profile for RTLS-type aborts. Figure CI(S)-42 is an altitude/range profile for RTLS-type aborts.

CI(S) 2.3.1.2 TAL. The TAL abort mode will be used for an SSME failure between two engine TAL capability and press-to-MECO. During the RTLS/TAL overlap, TAL is preferred because it is more tolerant of a second SSME failure. After selection of the TAL abort select, the vehicle will accelerate downrange to the TAL MECO target. At abort select, the OMS propellant dump is initiated to achieve the correct landing weight and center-of-gravity for entry. At an I-loaded velocity (VI=15,400 ft/s) the vehicle rolls heads-up, which decreases ET heating and places the Orbiter in entry attitude prior to MECO (VI=23,800 ft/s). After ET separation the onboard computers are loaded with the entry flight software and the Orbiter glides to the landing site. Landing sites change as a function of intended orbit inclination. For example, for orbit inclinations near 28 degrees, Dakar, Senegal (on the west coast of Africa) is the in-plane landing site. For inclinations near 57 degrees, Zaragoza, Spain, is currently the primary landing site. The weather alternate landing site for both of these inclinations is Moron, Spain. There is a possibility of installing runway overrun barriers at some of the TAL/Emergency landing sites; the barrier being considered is a net, which would cause minimal damage to the Orbiter. A typical TAL abort sequence of events is defined in Table CI(S)-4.

CI(S) 2.3.1.3 Press-to-MECO (AOA or ATO). If an SSME fails after Press-to MECO, the Orbiter continues to the nominal MECO target. On some flights, a predetermined amount of OMS propellant will be dumped du[illegible] powered flight by declaring ATO shortly after the SSME failure. This weight decrease, accompanied by a small increase in thrust from the OMS engines, will increase powered flight performance. After MECO, the energy, the OMS delta-V remaining, and the OMS delta-V required for orbit are evaluated. If the OMS delta-V is insufficient for orbit insertion and subsequent deorbit, an Abort-Once-Around (AOA) will be performed with the OMS engines.

Table CI(S)-3 RTLS Abort Sequence of Events for an Earliest and Latest SSME Failure

Mission Elapsed Time (s)		EVENT
Earliest	Latest	
0	0	Lift-off of the Space Shuttle Vehicle
1	240	SSME Failure
125	125	SRB Separation
150	255	RTLS Abort Selected
312	255	OMS Propellant Dump Initiated
424	255	Power Pitch Around Initiated
586	425	OMS Propellant Dump Completed
704	579	Power Pitch Down
721	594	MECO
734	608	ET Separation
745	620	MPS Dump Initiation
765	640	Aft RCS Propellant Dump Initiation
900	775	Aft RCS Propellant Dump Completed
865	740	MPS Dump Completed
940	800	Mach 3.5
1400	1260	Landing

Table CI(S)-4 RTLS Abort Sequence of Events for an Earliest and Latest TAL Abort

Mission Elapsed Time (s) Earliest	Latest	EVENT
0	0	Lift-off of the Space Shuttle Vehicle
125	125	SRB Separation
240	355	TAL Abort Selected
240	355	OMS Propellant Dump Initiated
480	415	Vehicle Roll-To-Heads Up Attitude
435	510	OMS Propellant Dump Completed
570	520	MECO
588	538	ET Separation
600	550	+X ARCS Thruster Firing Initiated (Manual)
605	555	+X ARCS Stop, MPS Dump Initiated (Manual)
635	585	MPS Dump Stop (Auto)
685	610	Entry Flight Software Memory Load
1650	1600	Mach 3.5
2100	2050	Landing

CI(S) 2.3.2 Contingency Aborts

A contingency abort will result if two SSMEs fail prior to single engine TAL capability or if three SSMEs fail prior to AOA capability. There is a possibility of performing a Split-S abort to the KSC runway if two or three SSMEs fail in the first 20 seconds. If three SSMEs fail during the last 30 seconds of an ascent, there is sufficient energy to glide to a runway in Africa. All emergency landing sites being considered are longer than 10,000 feet. During the remainder of the ascent, a contingency abort will result with the crew bailing out at 20,000 feet and the Orbiter impacting the water at about 180 kts with an alpha of about 15 degrees; the corresponding vertical descent rate is about 80 ft/s.

CI(S) 2.3.3 Loss of Critical Function

This section lists certain types of failures that would preclude either an intact or a contingency abort. The types of failures that lead to a loss of critical function are: aft compartment explosion, rupture or explosion of the ET, burn-through of an SRB, failure of major structure, complete loss of guidance or control, failure of one SRB to ignite, loss of thrust from either SRB, hardover condition of a main engine or an SRB, failure to separate the Orbiter from the ET, failure of the nozzle of an SRB, or premature separation of an SRB from the ET.

CI(S) 2.3.4 Fast Separation

Fast ET separation will be used during a contingency abort following failure of two SSMEs and prior to single engine TAL capability, or following the failure of three SSMEs during the first stage. It can only be selected manually by the crew when software is in Major Mode (MM) 102, 103, or 601. Here MM102 is first stage ascent, MM103 is second stage ascent, and MM601 is powered RTLS. The fast separation sequence will be entered if ET separation is initiated manually in MM102 and MM601, but in MM103 it will be entered only if a second SSME failure has been confirmed in addition.

The fast sequence shortens the time between MECO and Structural Release (SR) by eliminating some of the timed intervals between commands which allowed the first function to be completed before a succeeding function is commanded. Intervals are eliminated in both the SSME Operations (OPS) and separation sequence. The SSME OPS eliminates the interval between MECO and MECO Confirmed. At the latter event, the Separation Sequencer is initiated. The time from MECO to the command ALL PREVALVES COMMANDED CLOSED is shortened from nominal six seconds (approximately) to about 3.75 seconds. The Separation Sequencer is shortened by elimination of delays for Pyrotechnic Initiator Controller (PIC) arm and fire, for feedline disconnect closure, and for umbilical door closure.

Small differences exist in the fast separation sequencer depending on the MM in effect when it is entered: if in MM103, the MPS gimbals are moved to the dump position after shutdown; otherwise they are moved to the stow position. Also, if the MM103, the -Z Cmd is issued one cycle prior to structural separation; otherwise the -Z jets are fired at structural separation. Finally, if the fast sequence is entered from MM102, the unbilical doors remain open and latched, and a contingency propellant dump is declared. Otherwise, the umbilical doors are closed in a normal sequence requiring about 66 seconds.

The fast separation sequence will not be selected in MM102 while the SRBs are burning because the Orbiter will hang up on the aft attach points and break up aero dynamically. The only planned use in MM102 is when the SRB chamber pressure has decayed to less than 50 psi for three SSMEs failed.

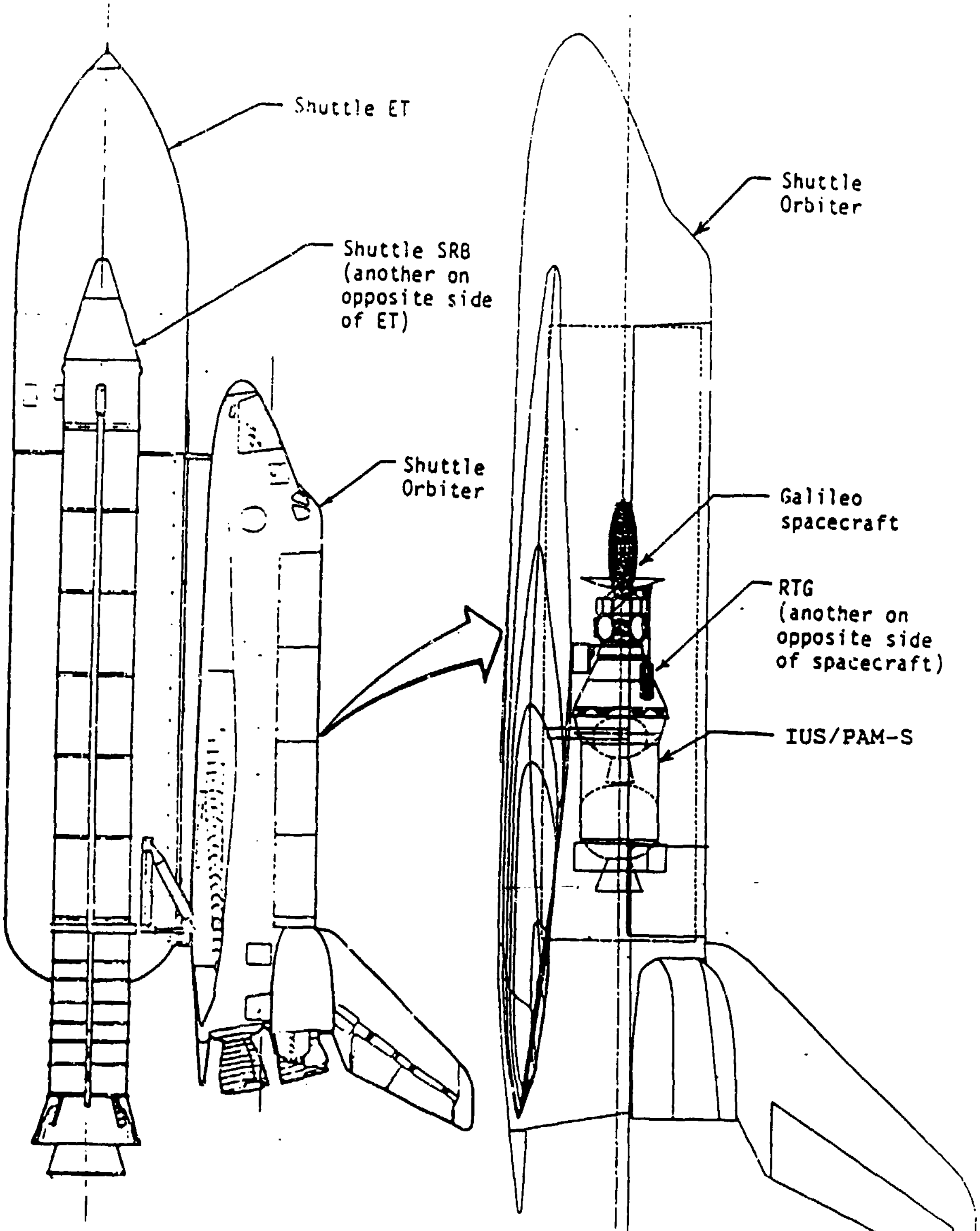


Figure CI(S)-1 Galileo Launch Configuration

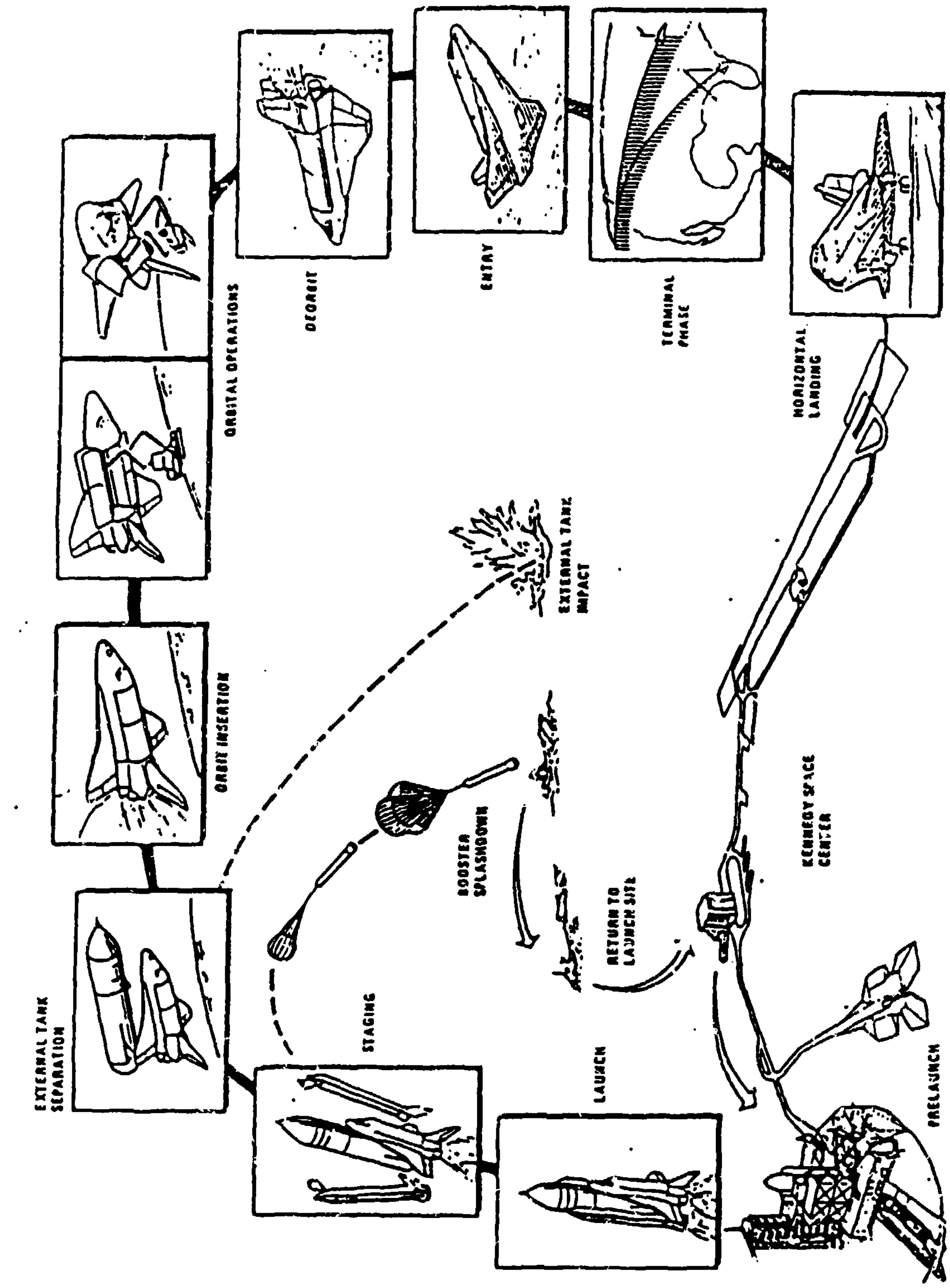


Figure CI(S)-2 Basic Mission Cycle for Shuttle

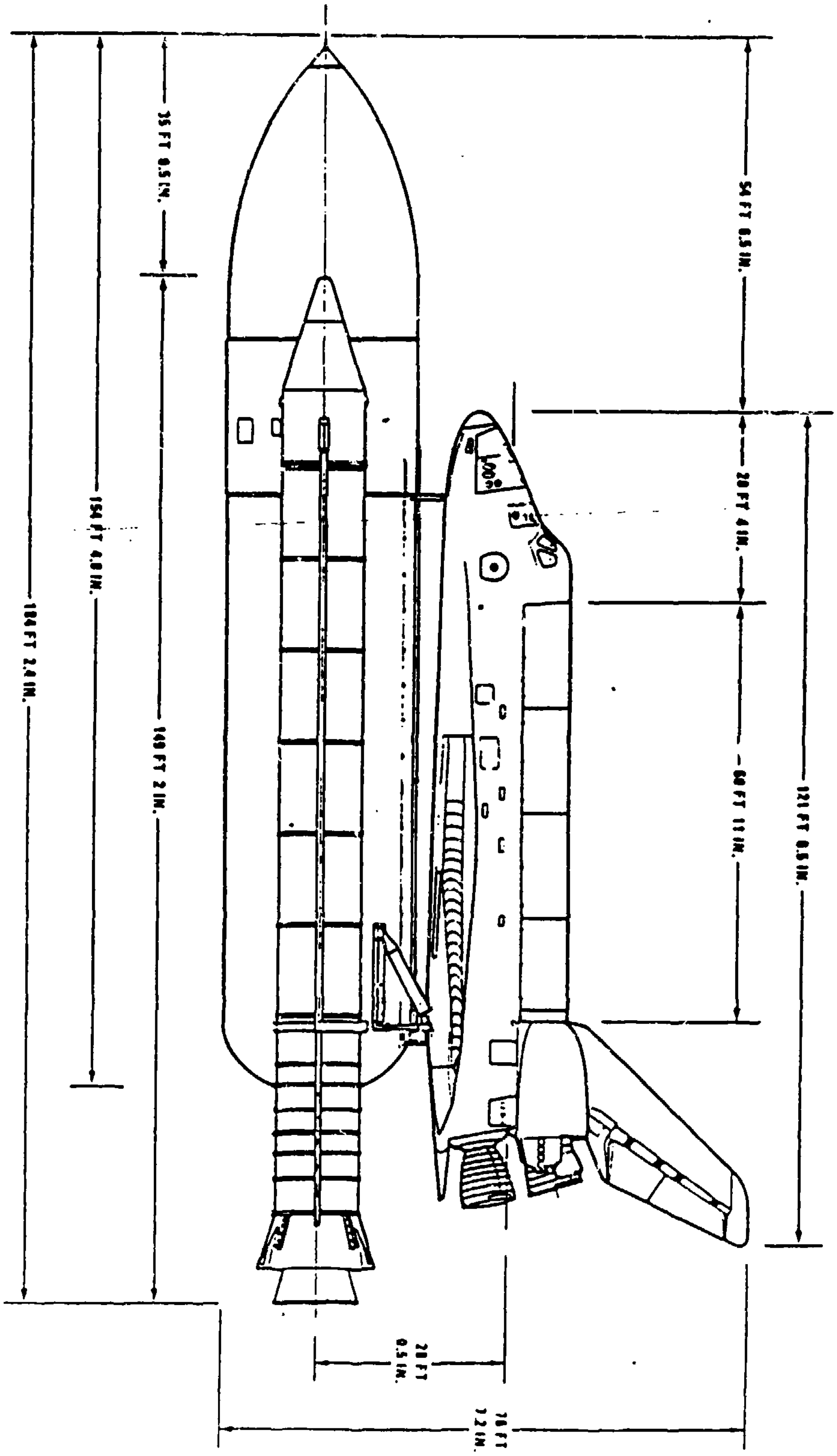


Figure CI(S)-3 Shuttle Vehicle, Side View

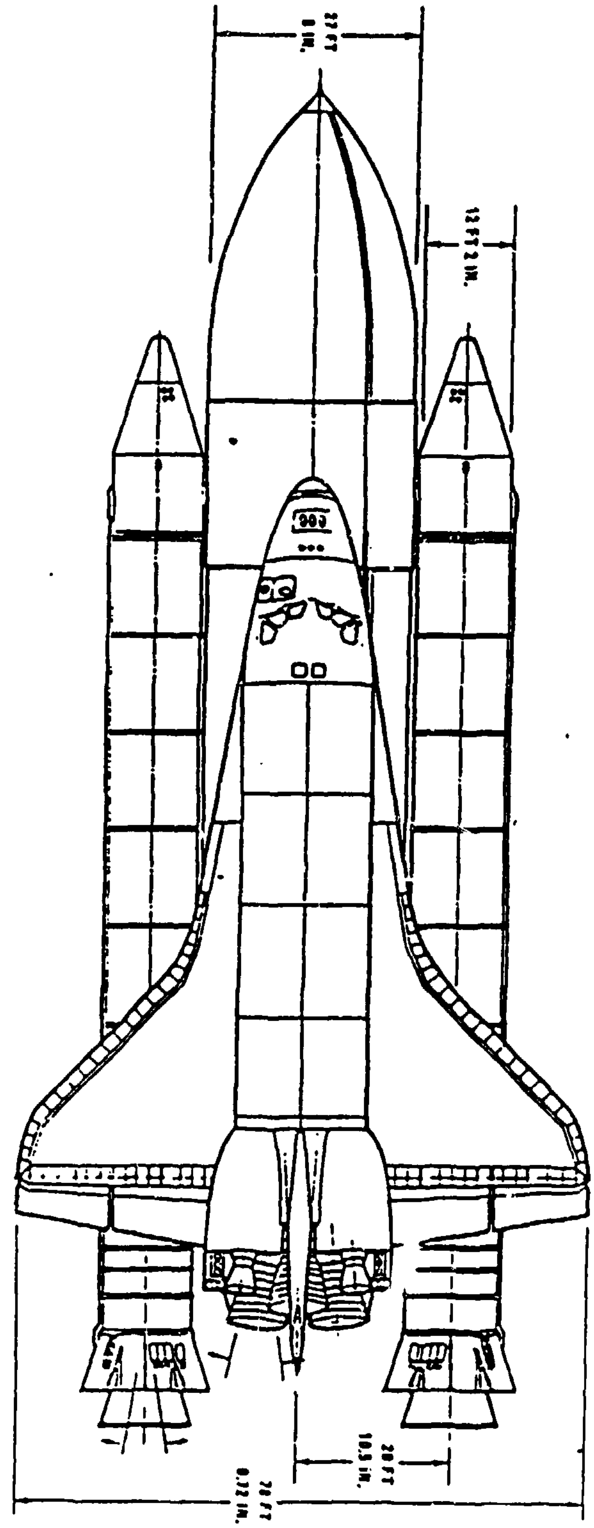


Figure CI(S)-4 Shuttle Vehicle, Top View

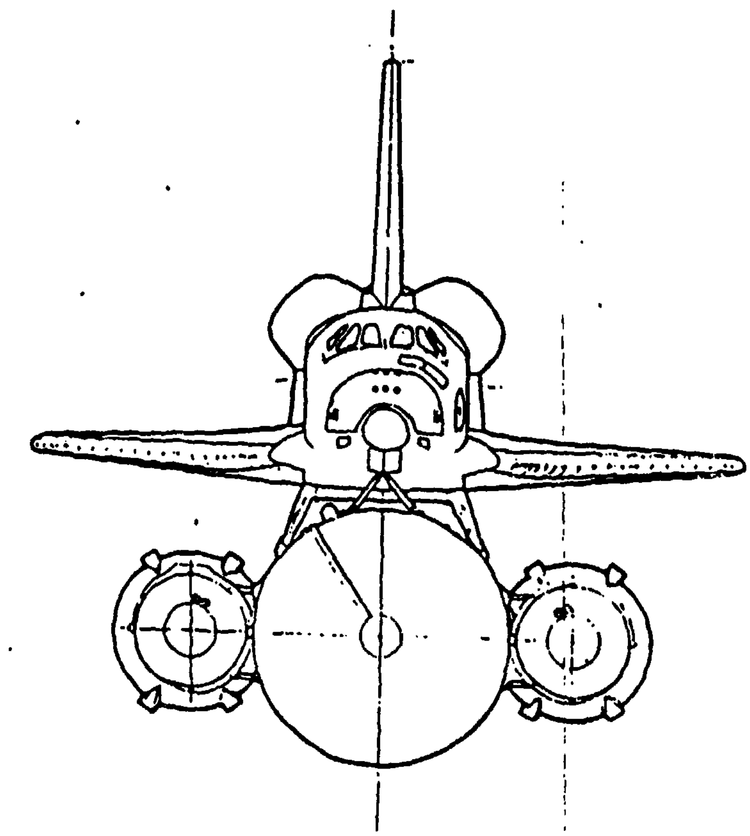


Figure CI(S)-5 Shuttle Vehicle, Front View

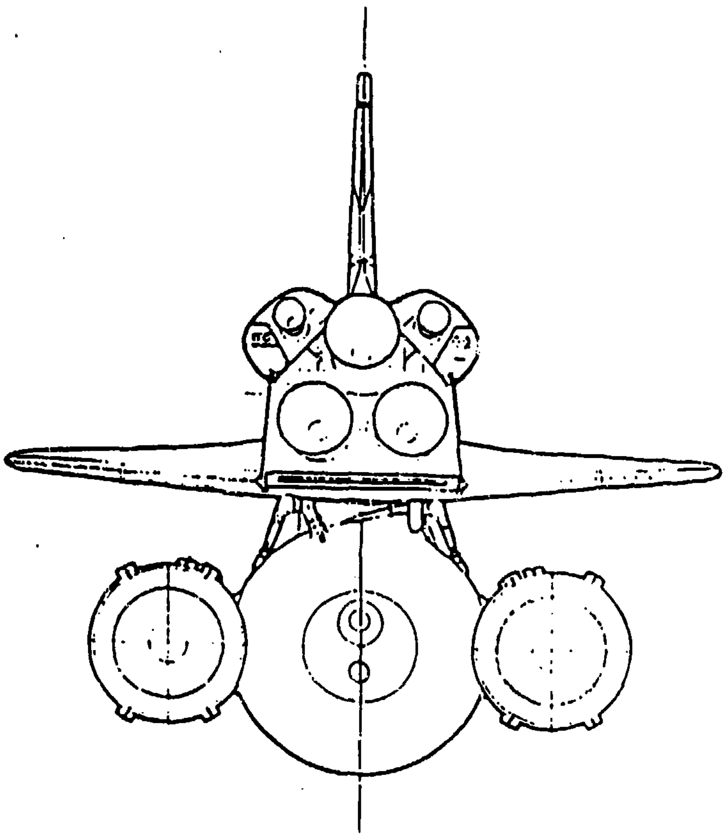


Figure CI(S)-6 Shuttle Vehicle, Back View

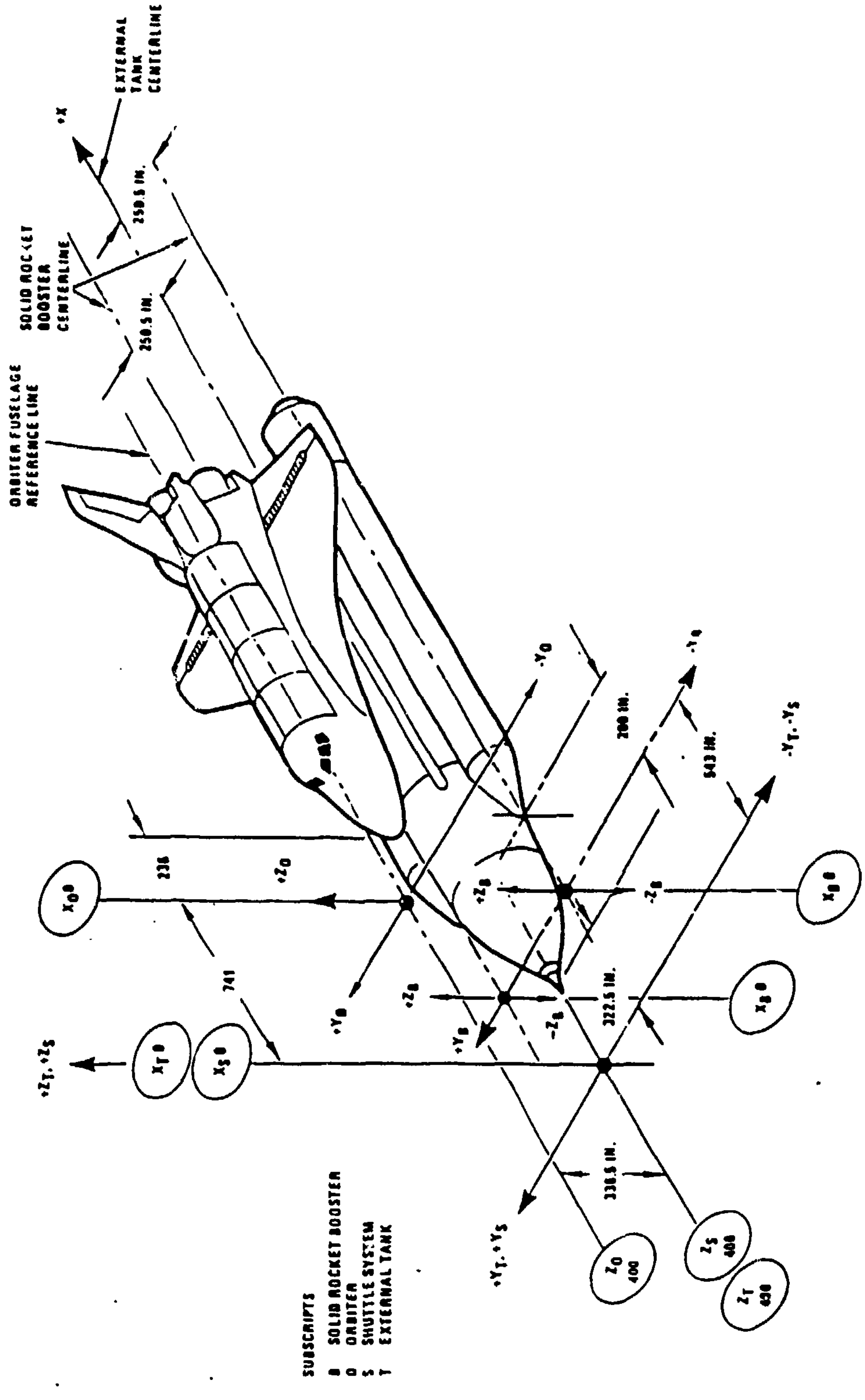


Figure CI(S)-7 Shuttle Coordinate Systems

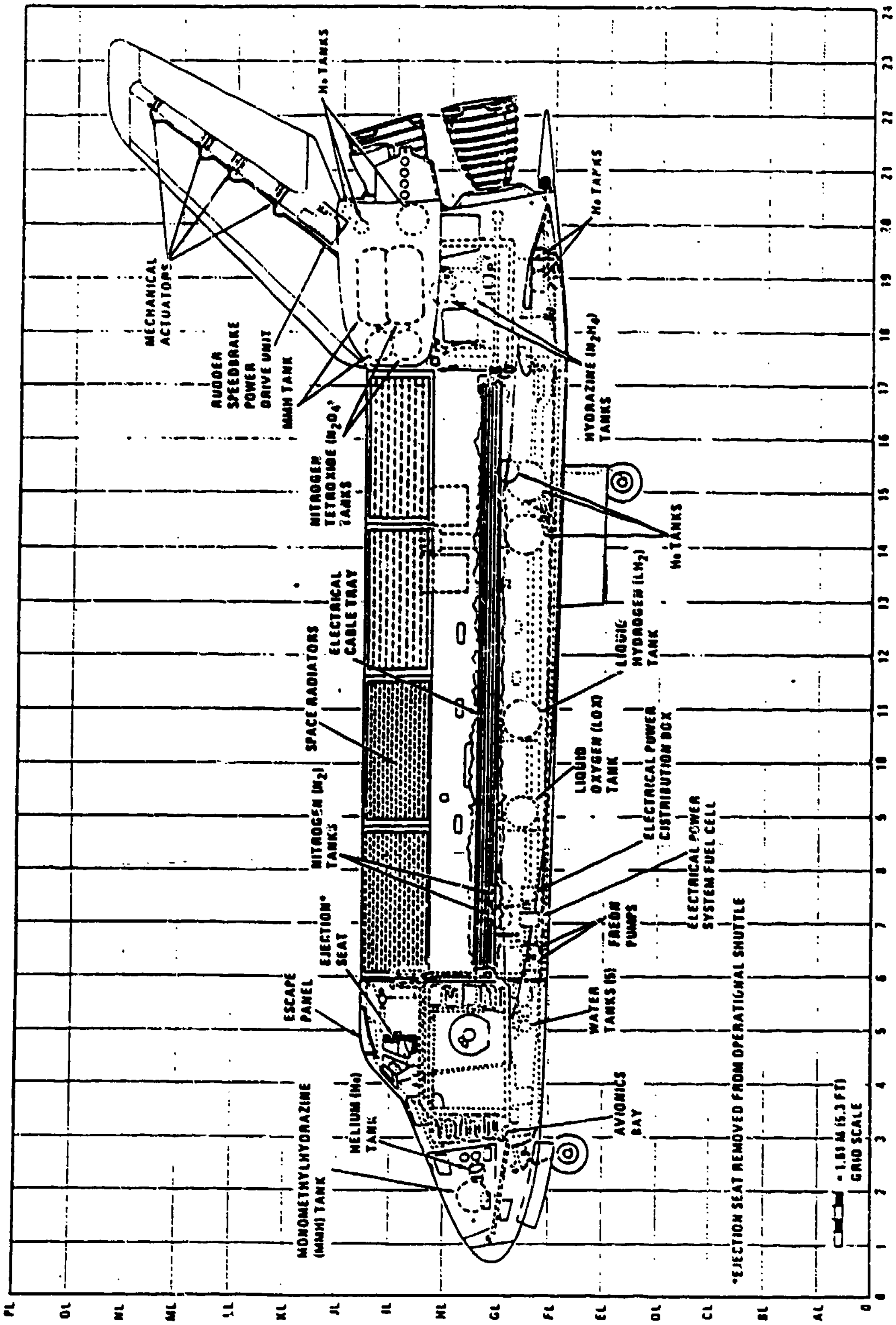


Figure CI(S)-8 Layout of Orbiter, Left-Hand View

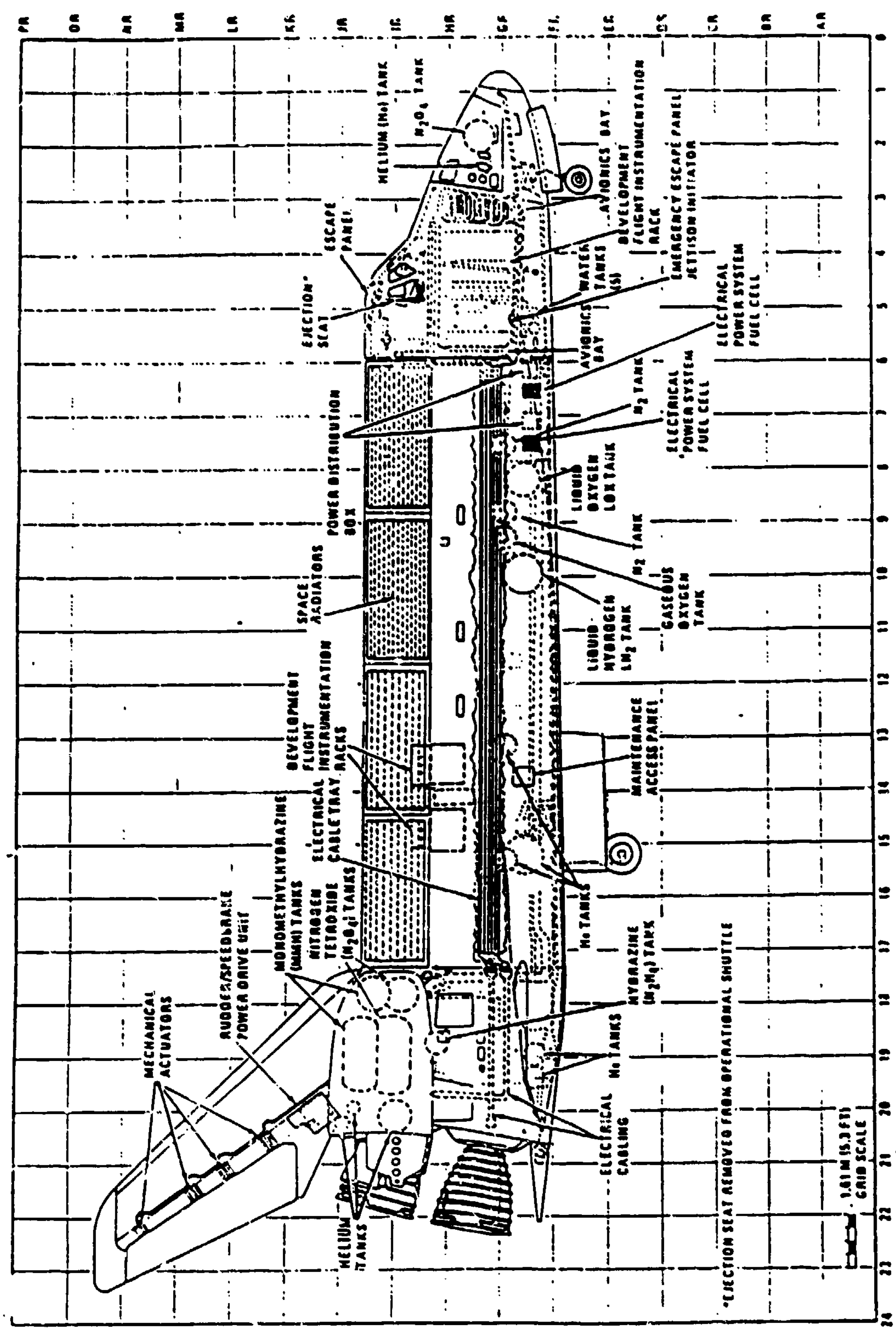


Figure CI(S)-9 Layout of Orbiter, Right-Hand View

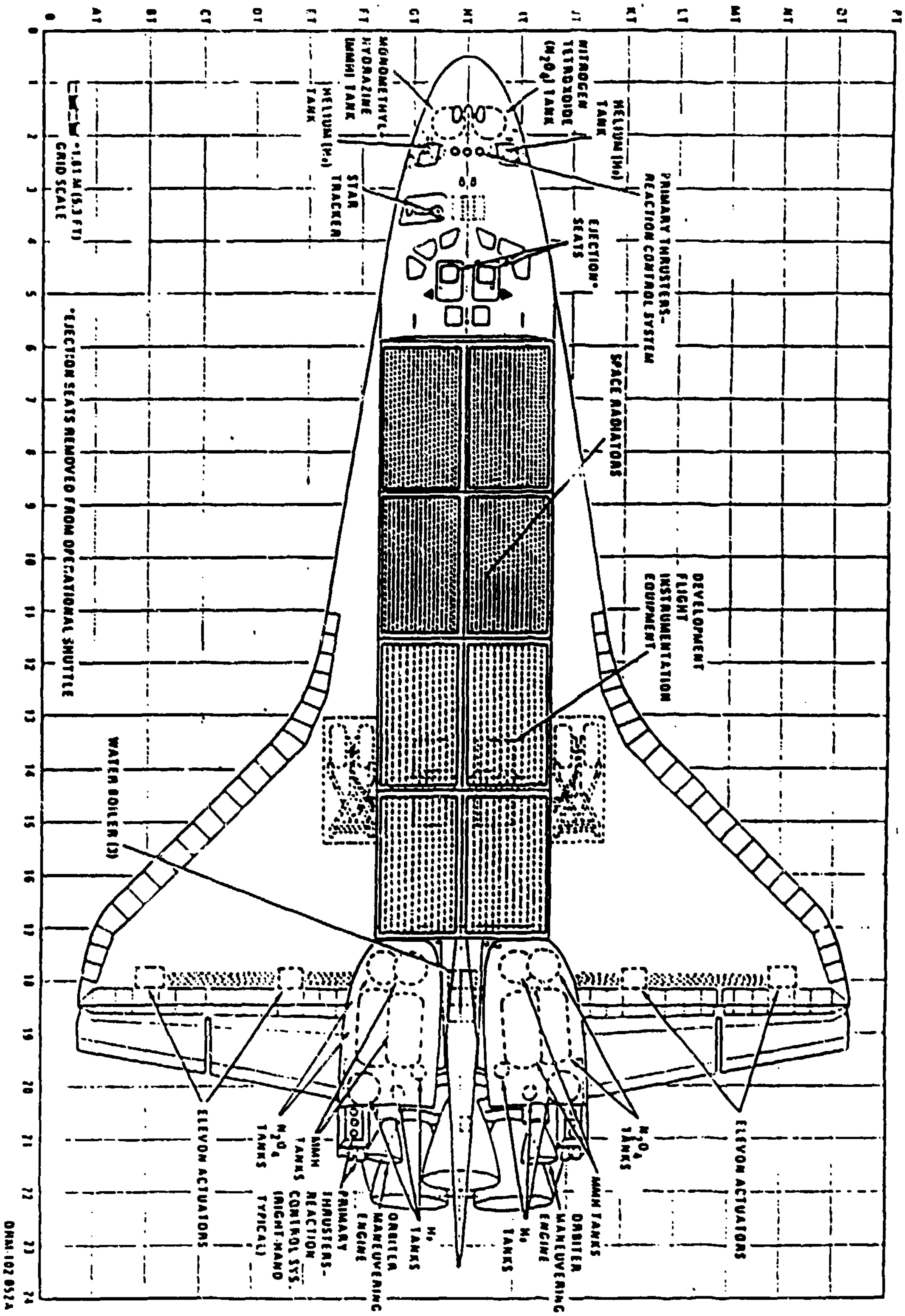


Figure CI(S)-10 Layout of Orbiter, Top View

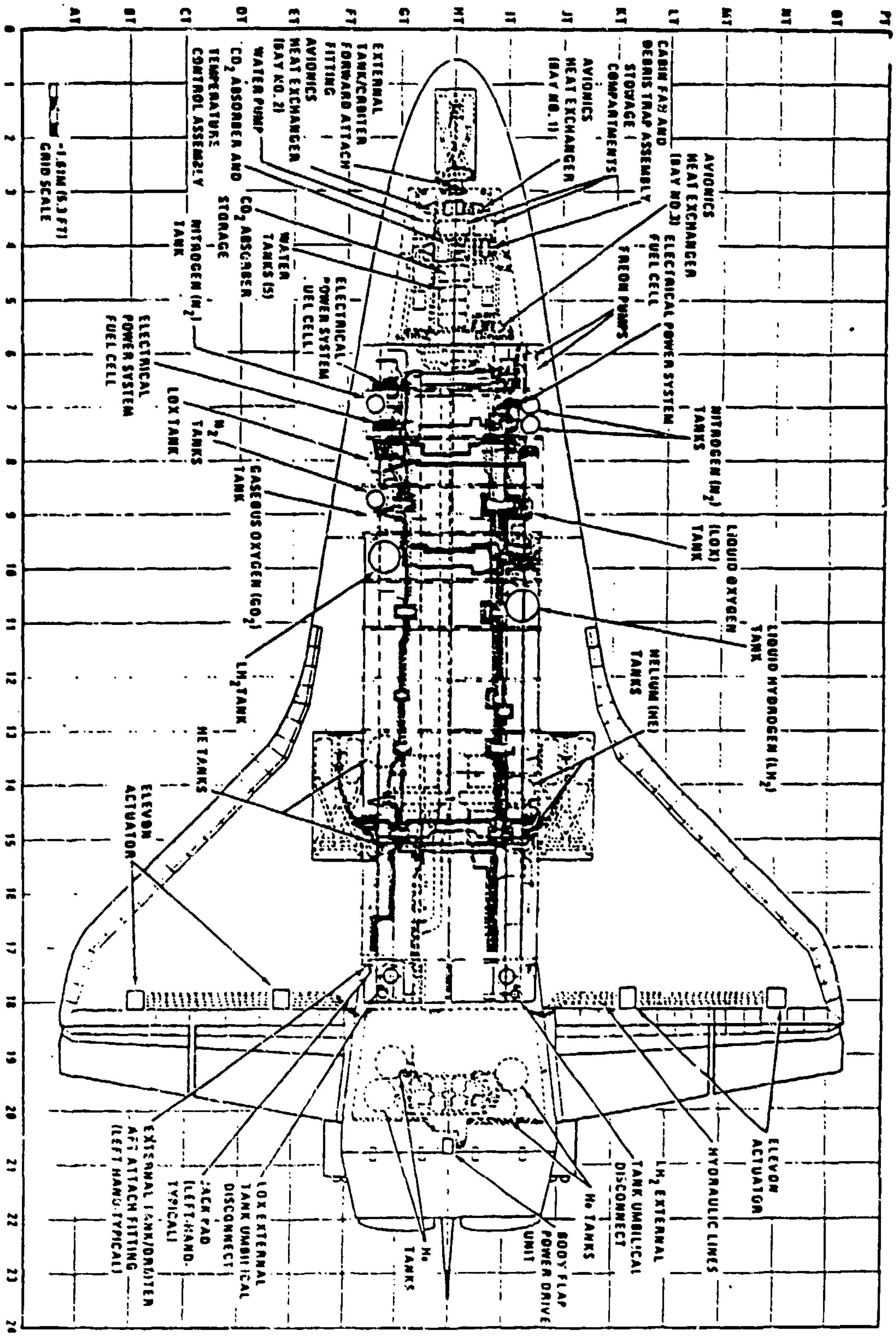
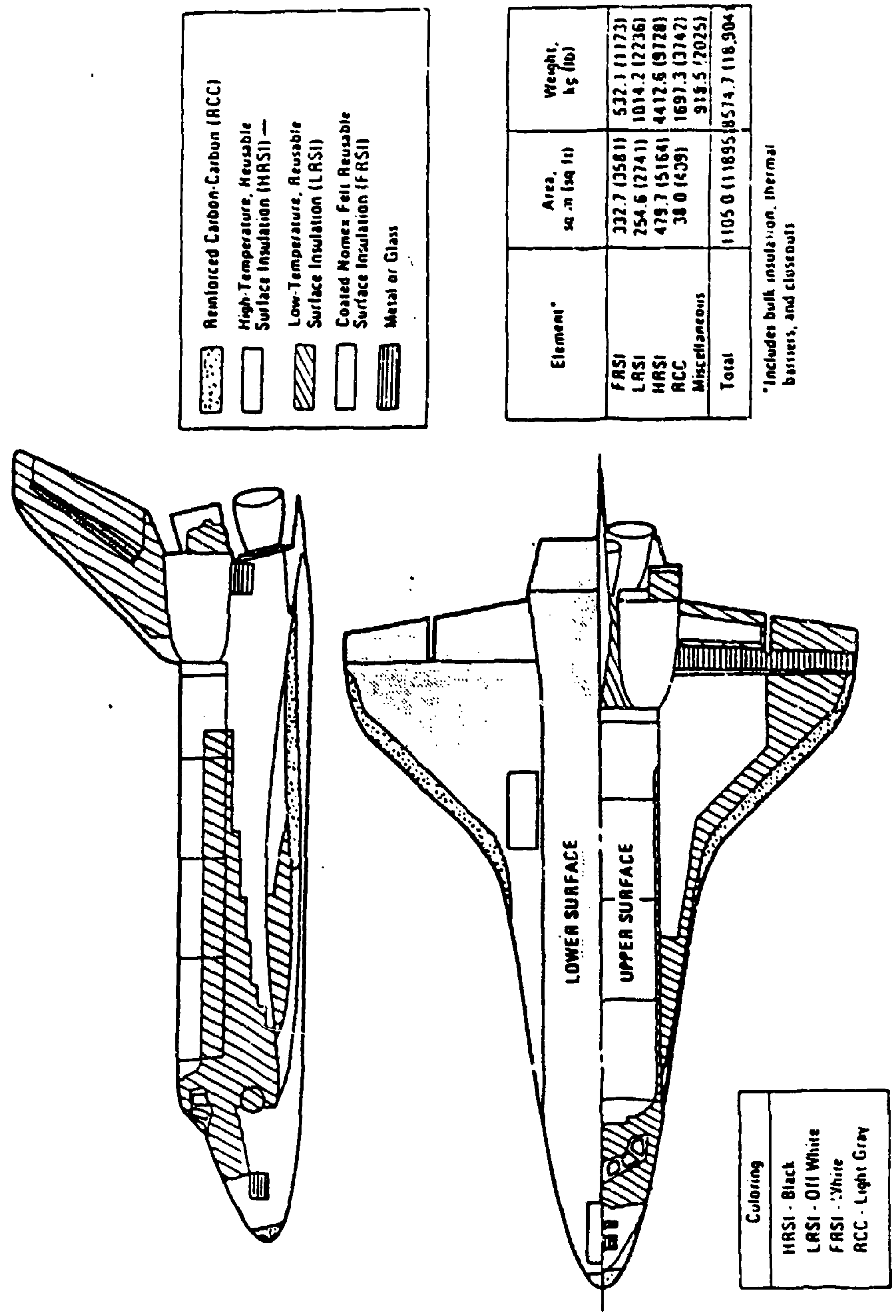


Figure CI(S)-11 Layout of Orbiter, Bottom View



Element*	Area, sq m (sq ft)	Weight, kg (lb)
FRSI	332.7 (3581)	532.1 (1173)
LRSI	254.6 (2741)	1014.2 (2236)
HRSI	479.7 (5164)	4412.6 (9728)
RCC	38.0 (409)	1697.3 (3742)
Miscellaneous		918.5 (2025)
Total	1105.0 (11895)	8574.7 (18,904)

*Includes bulk insulation, thermal barriers, and closeouts

Figure CI(S)-12 Thermal Protection System, Orbiter 102

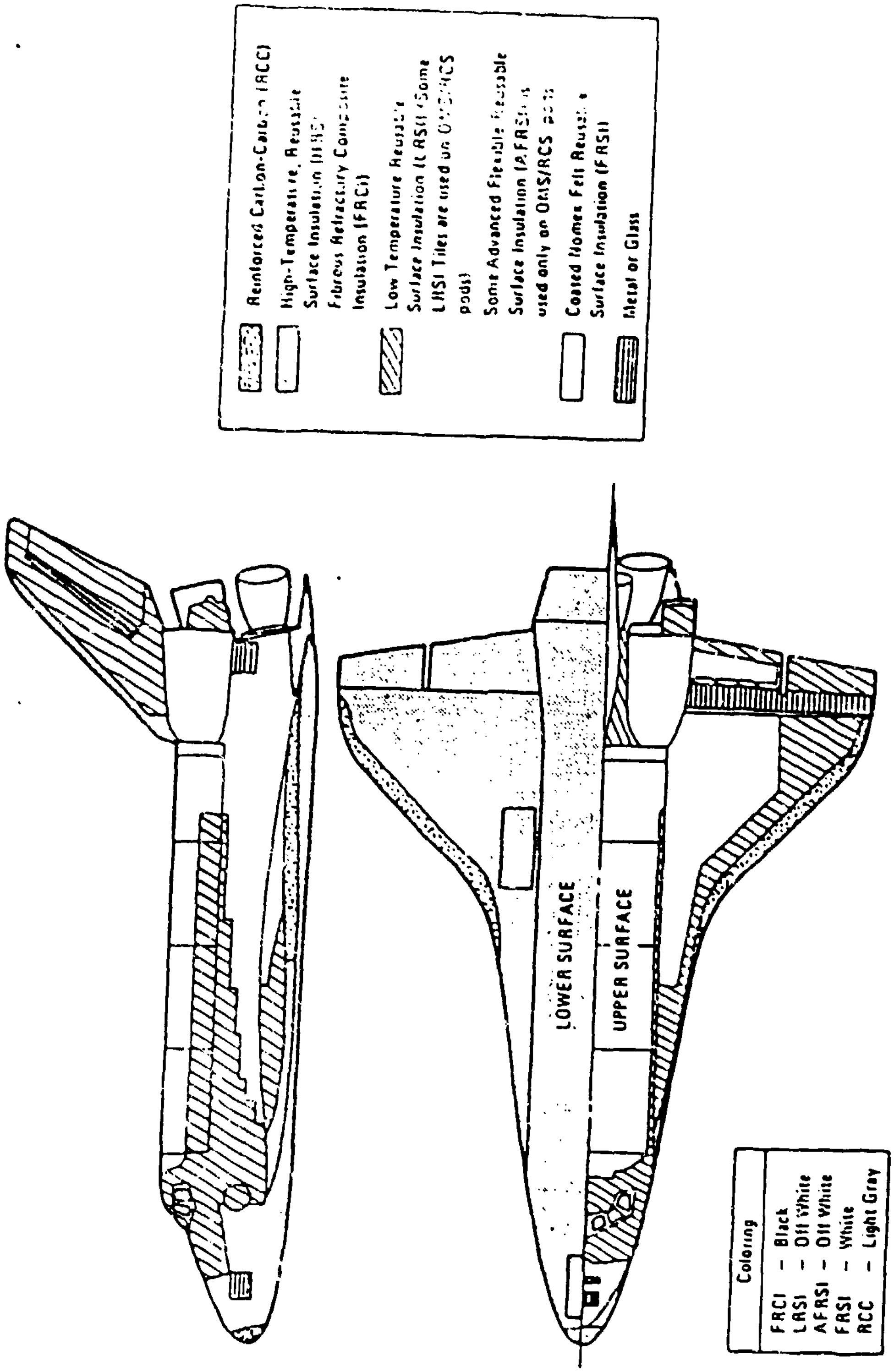


Figure CI(S)-13 Thermal Protection System, Orbiter 099 and Subsequent Orbiters

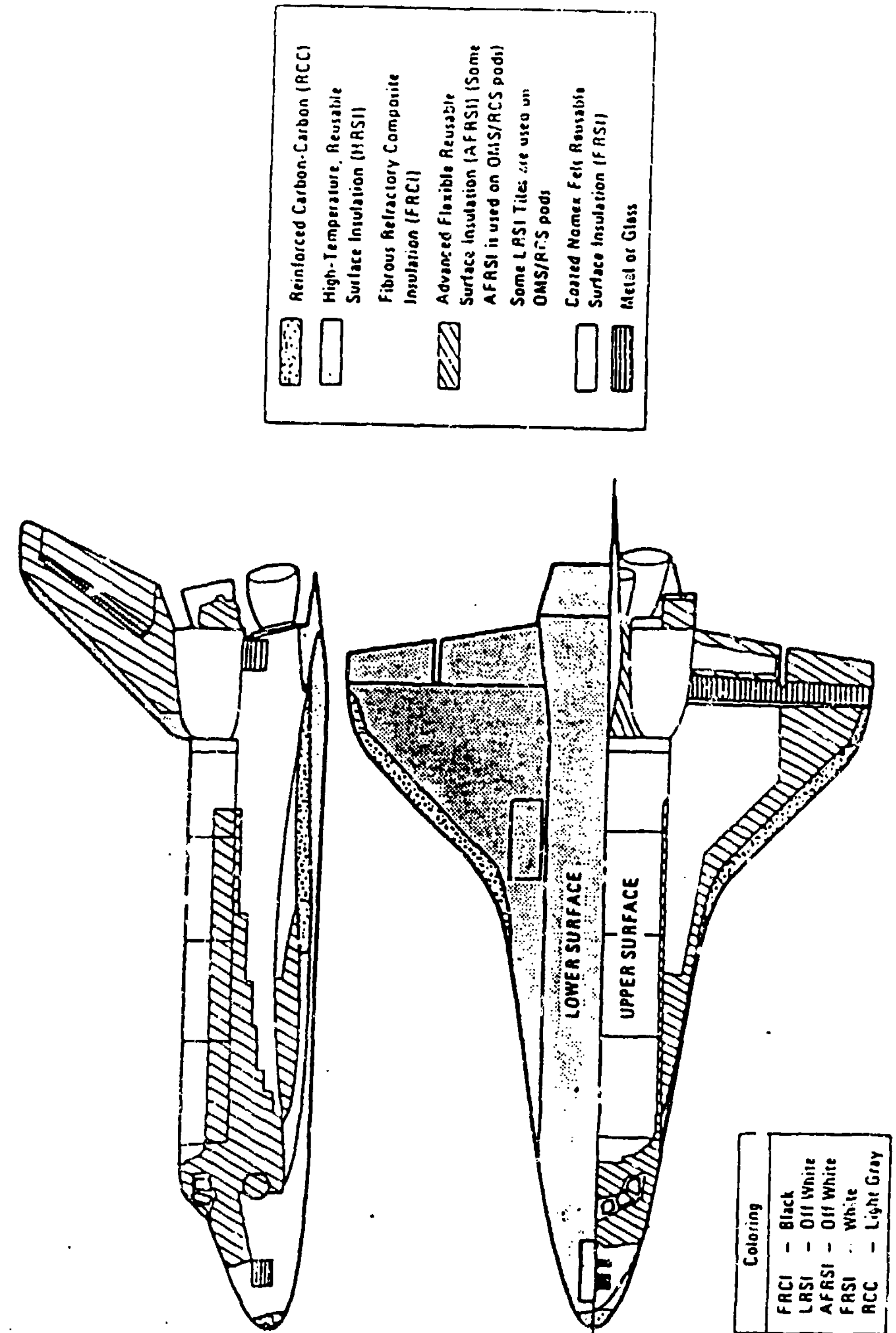


Figure CI(S)-14 Thermal Protection System, Orbiter 103 and Subsequent Orbiters

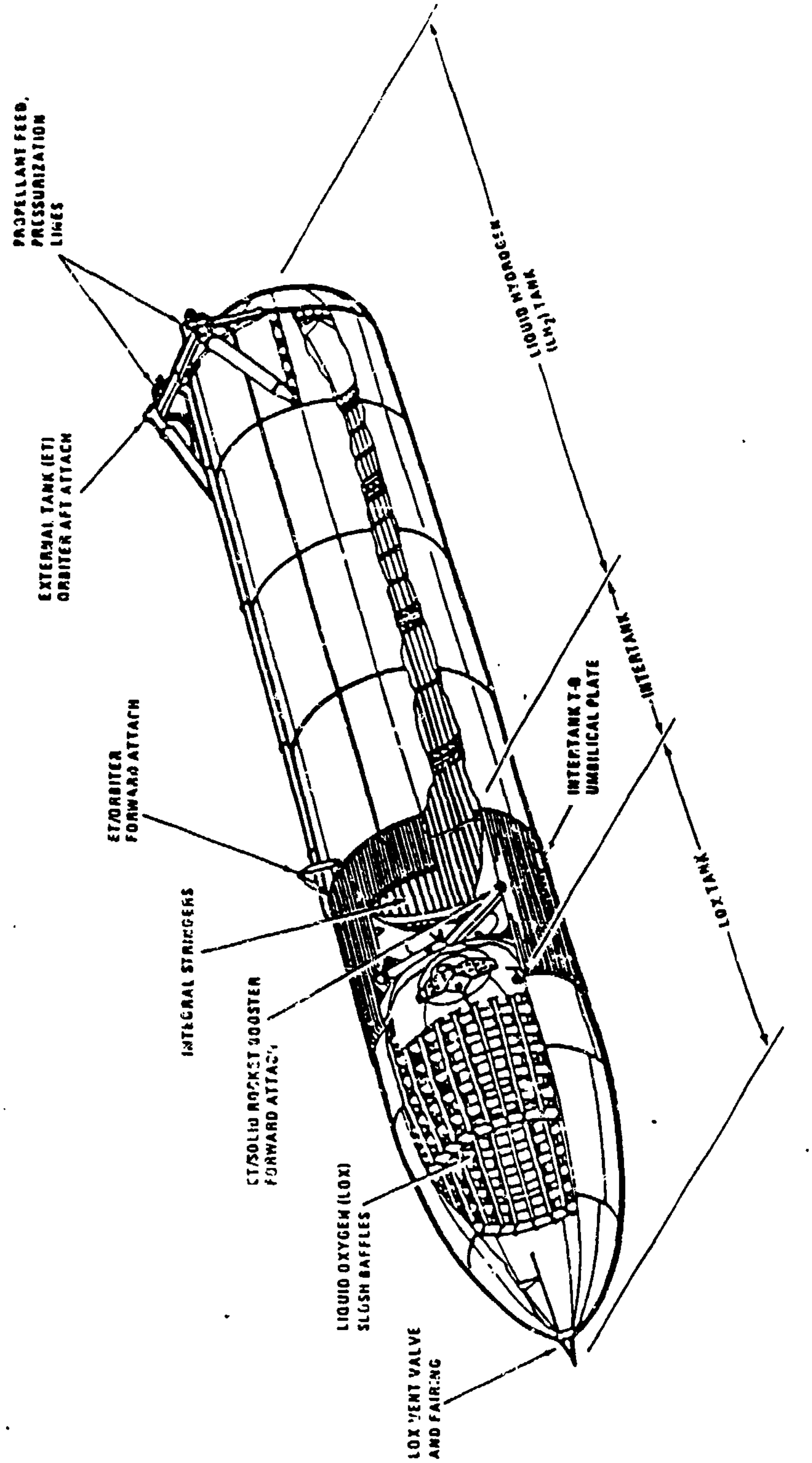


Figure C1(S)-15 Shuttle ET

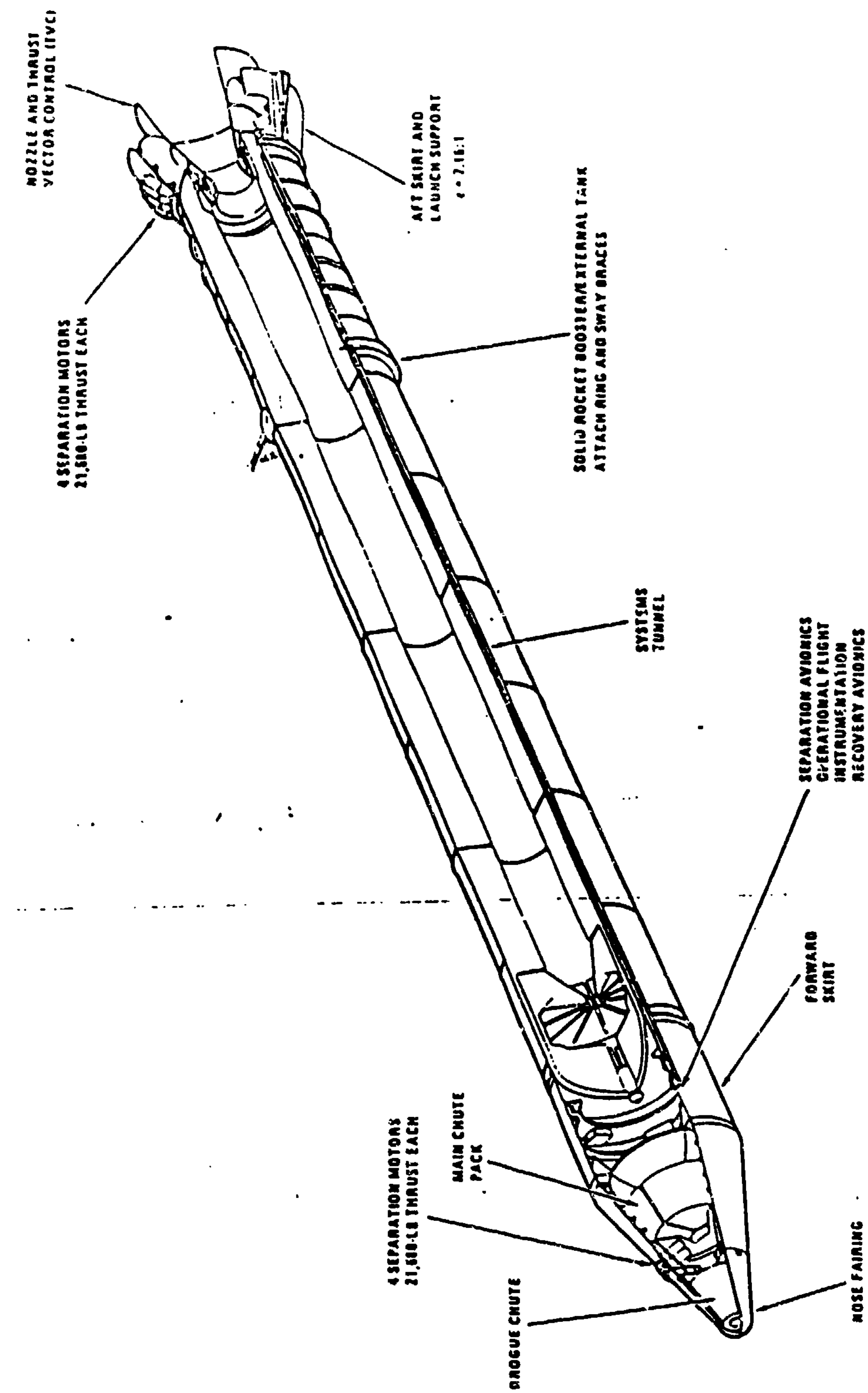


Figure CI(S)-16 Shuttle SRB

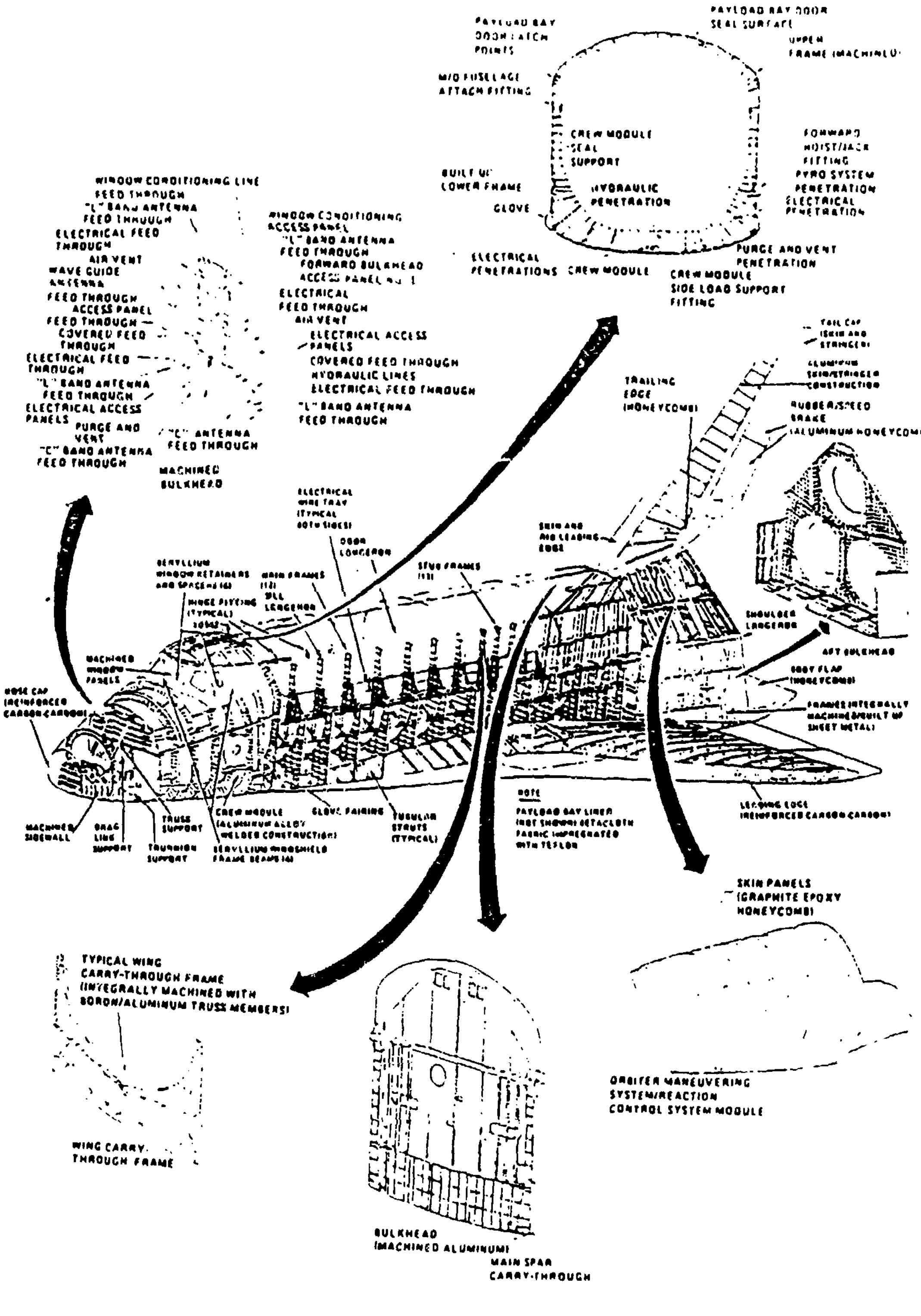


Figure CI(S)-17 Orbiter Structure, Perspective View

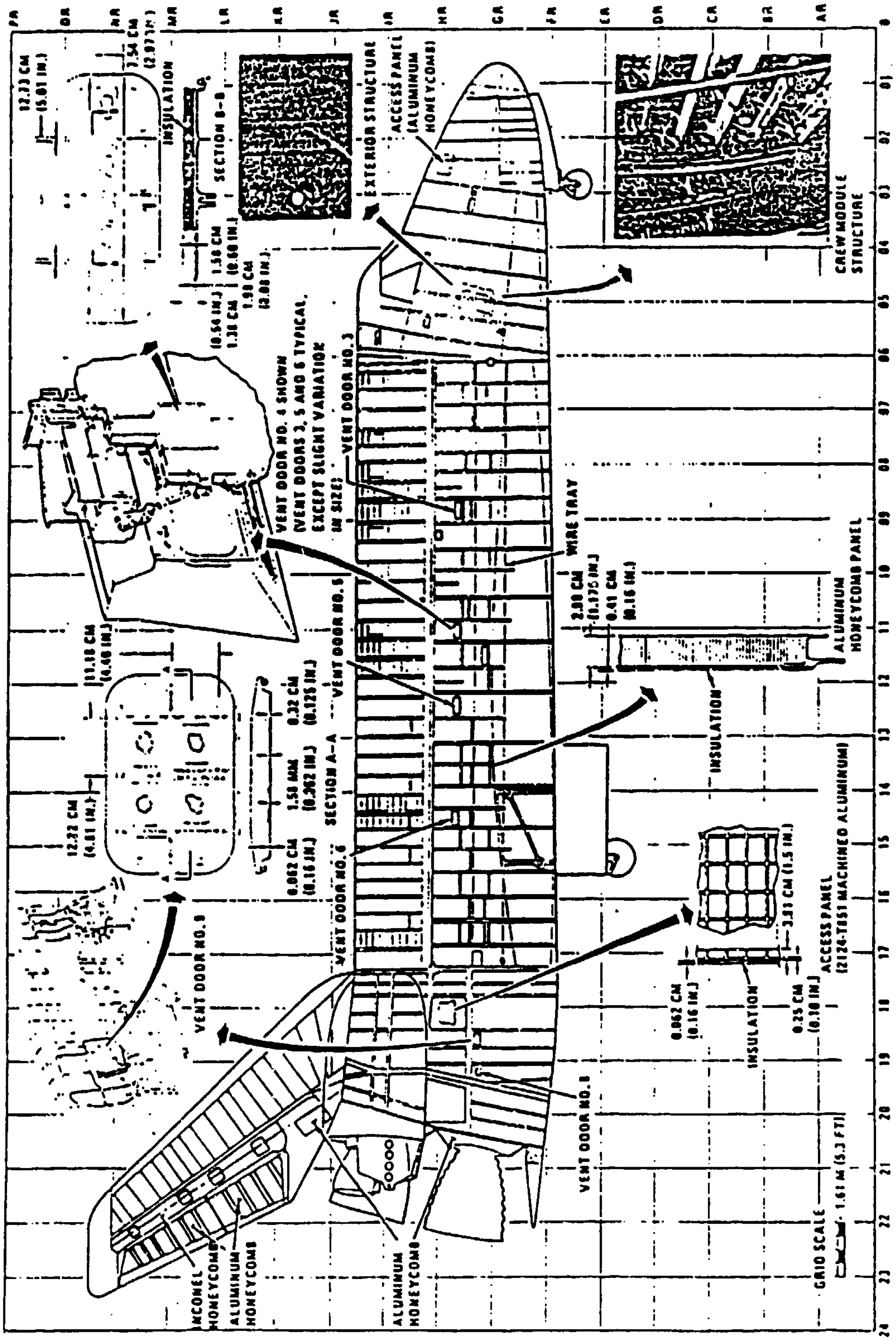


Figure CI(S)-18 Orbiter Structure, Side View

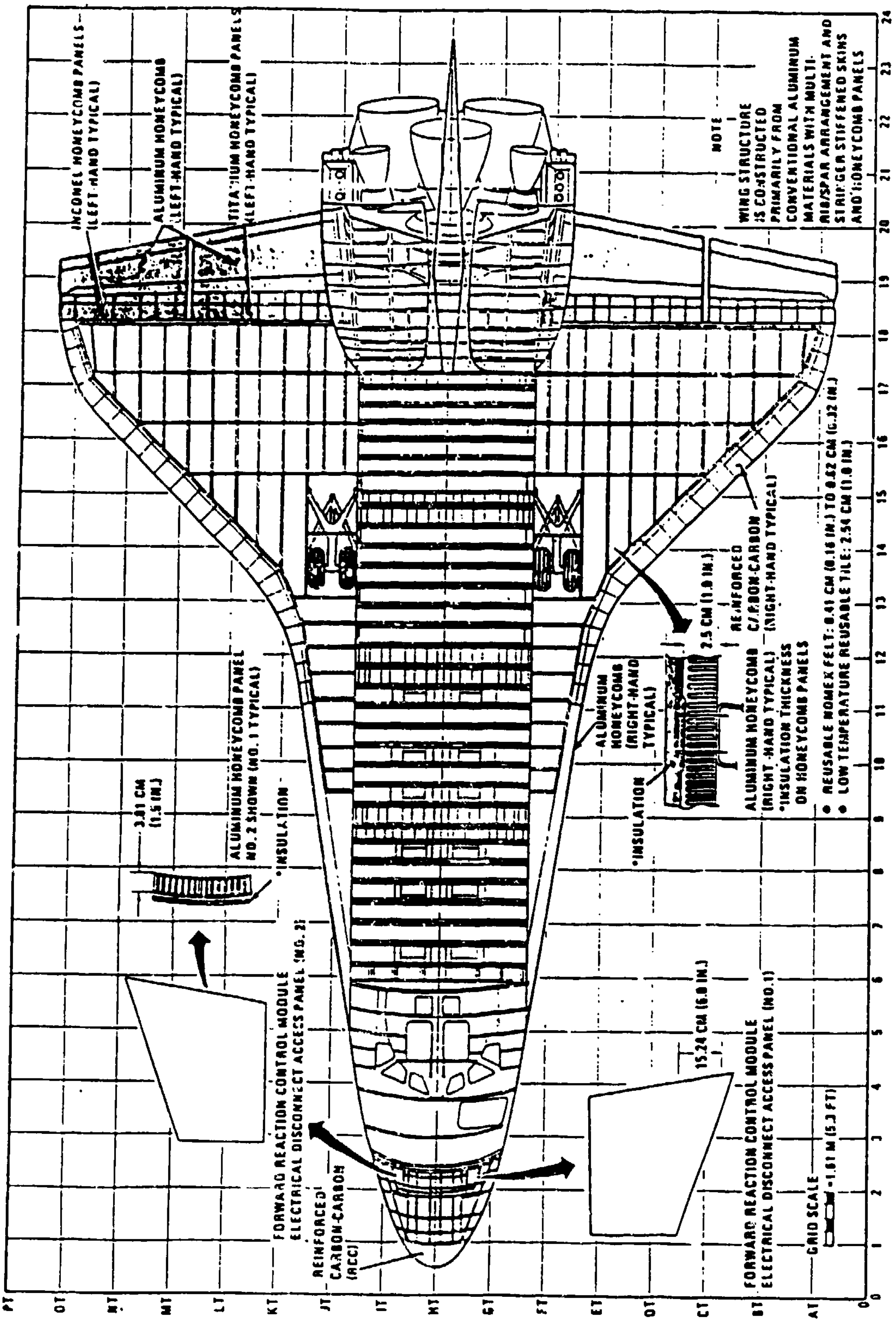


Figure CI(S)-19 Orbiter Structure, Top View

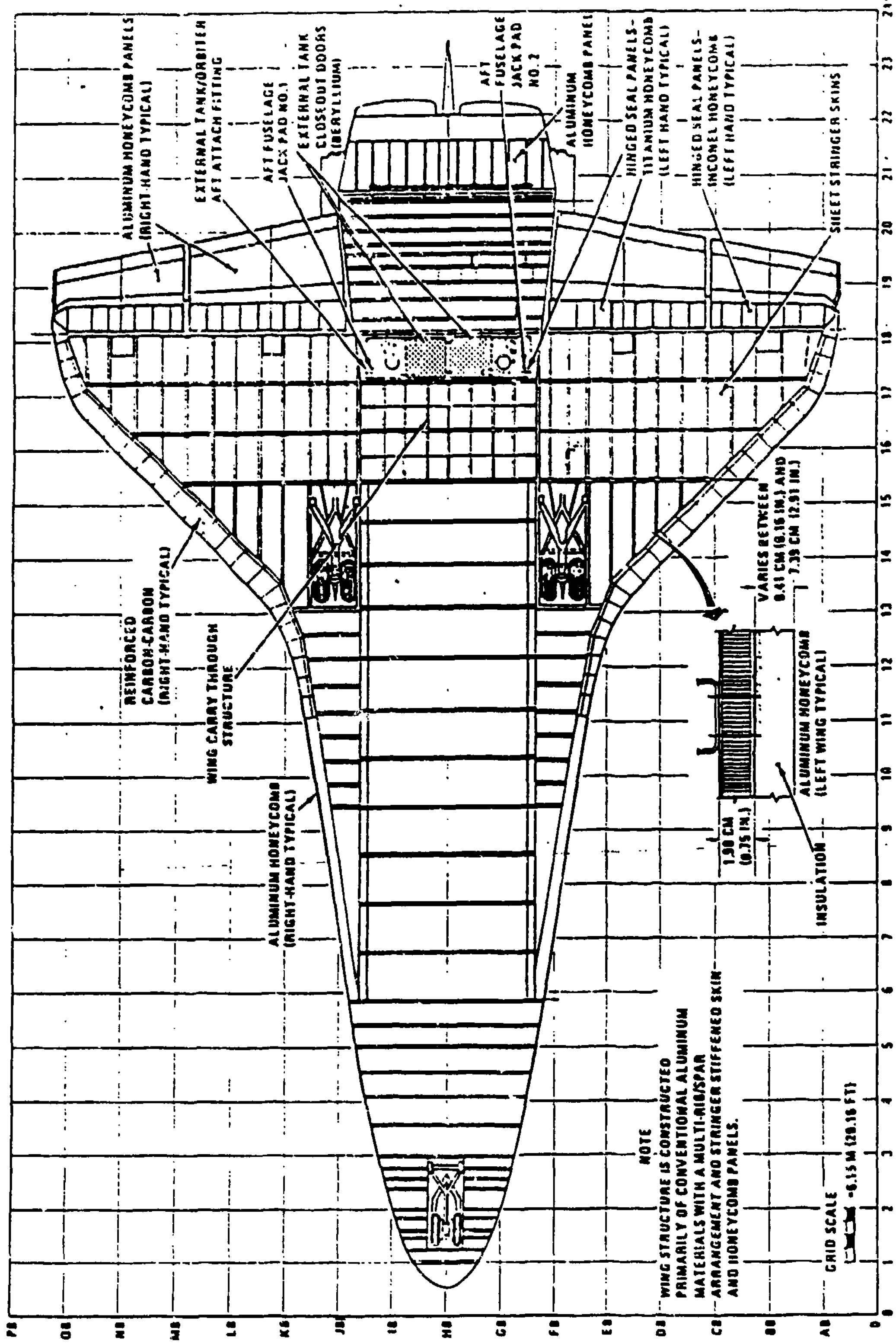


Figure CI(S)-20 Orbiter Structure, Bottom View

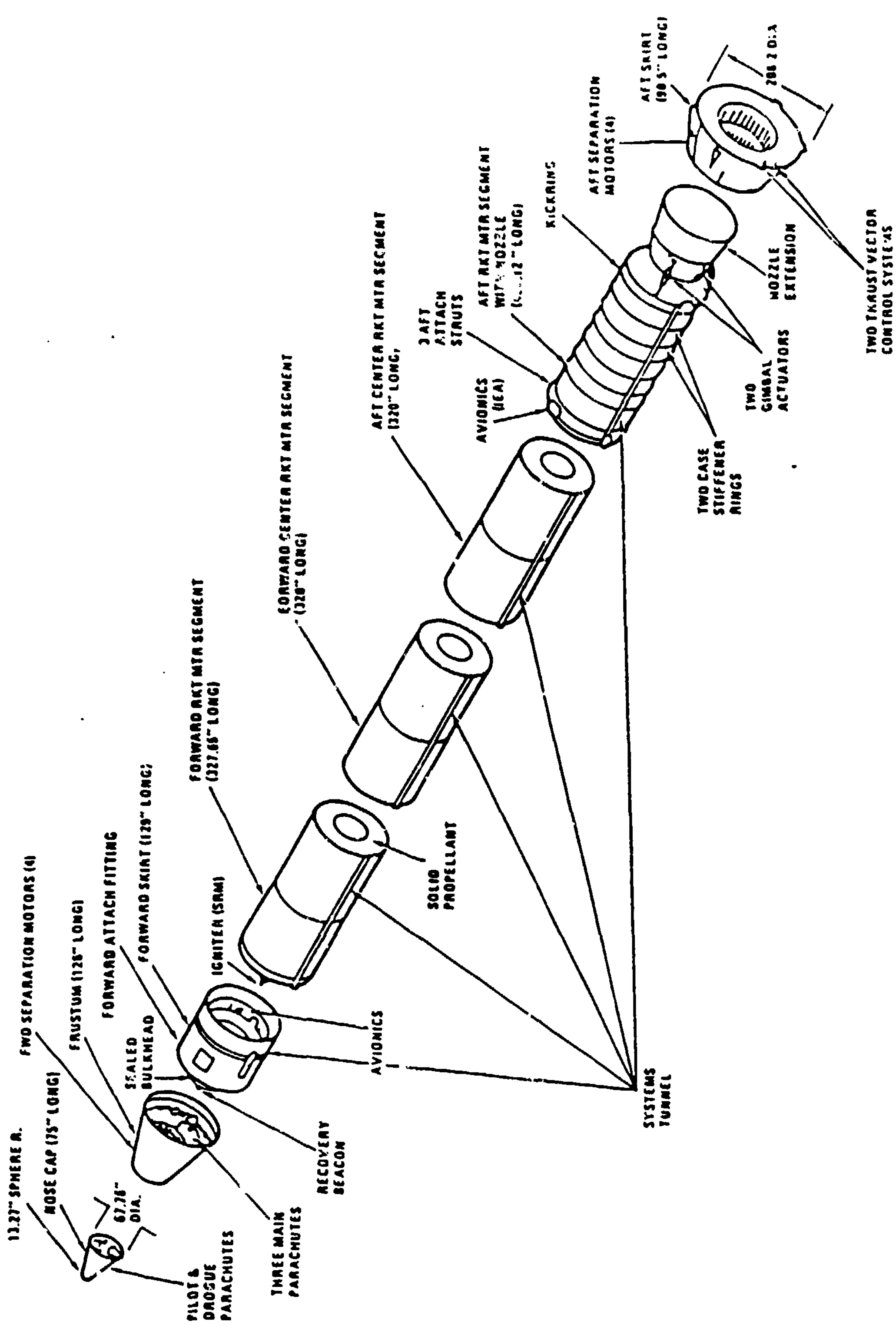


Figure CI(S)-21(a) SRB Parts - Overall Structure

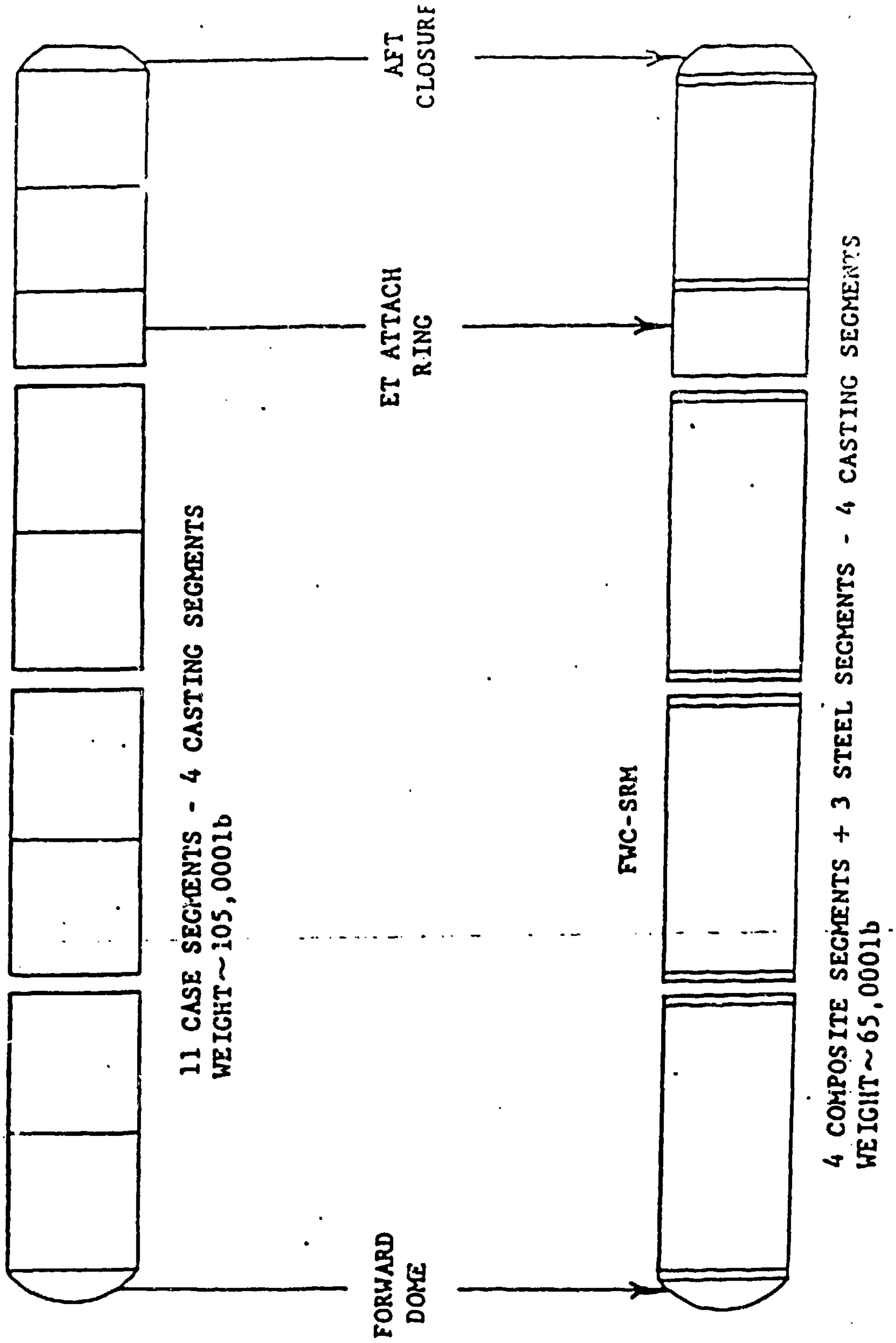


Figure CI(S)-21(b) Steelcase SRM & Filament Wound Case SRM

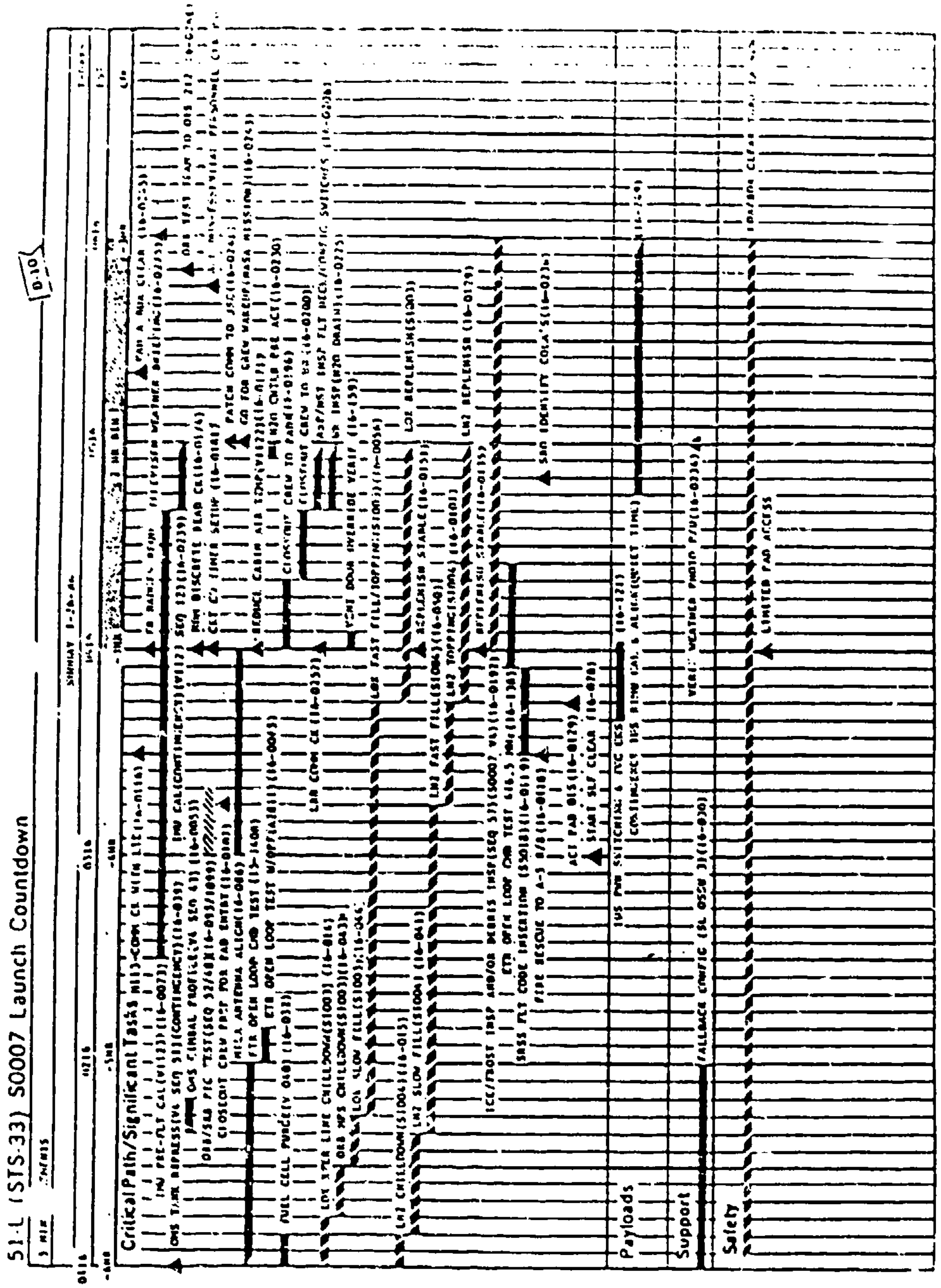


Figure CI(S)-22 Timeline of Propellant Loading & Launch Countdown Activities

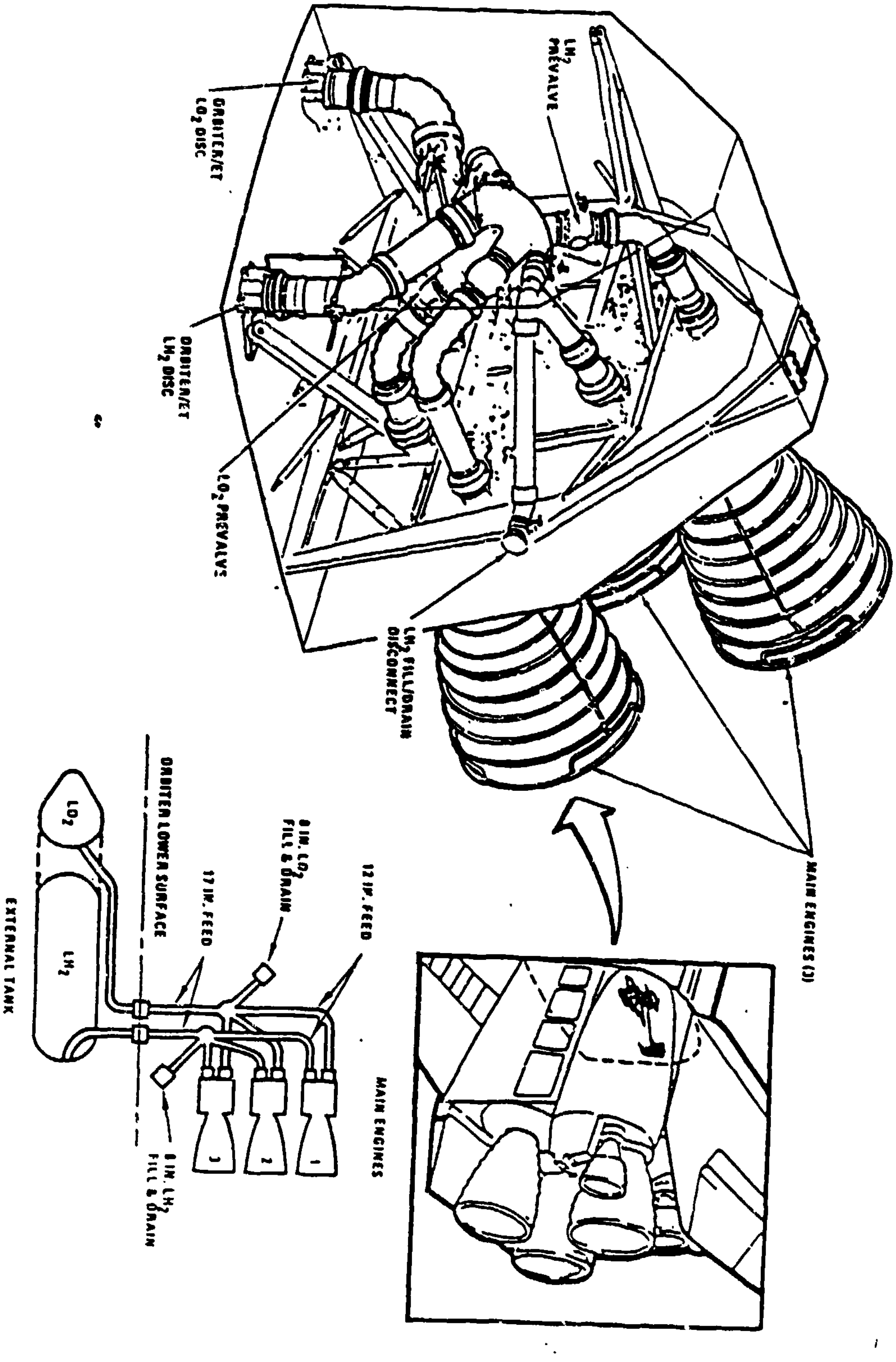


Figure CI(S)-23 Space Shuttle MEs

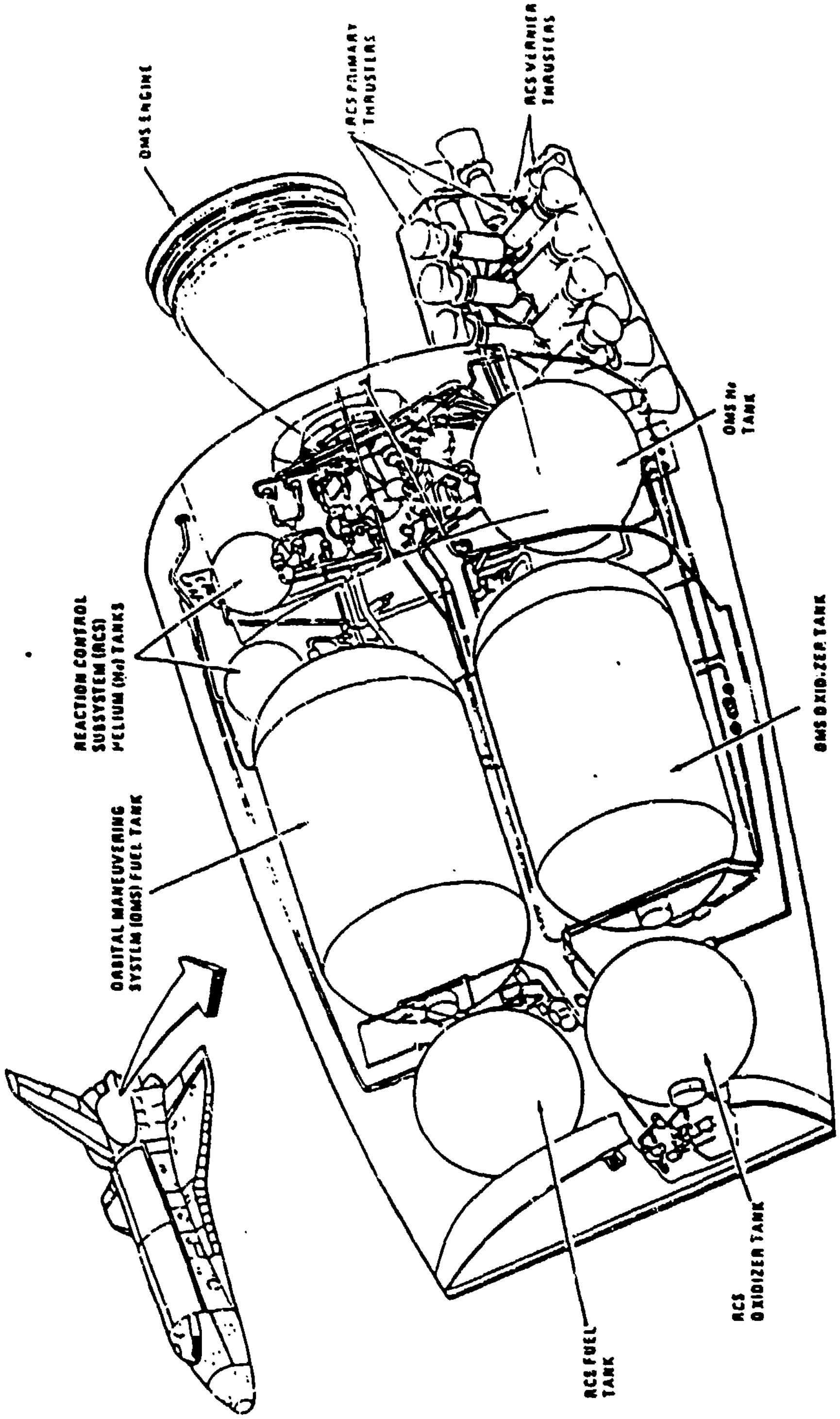


Figure CI(S)-24 Orbital Maneuvering Subsystem Engine Pod

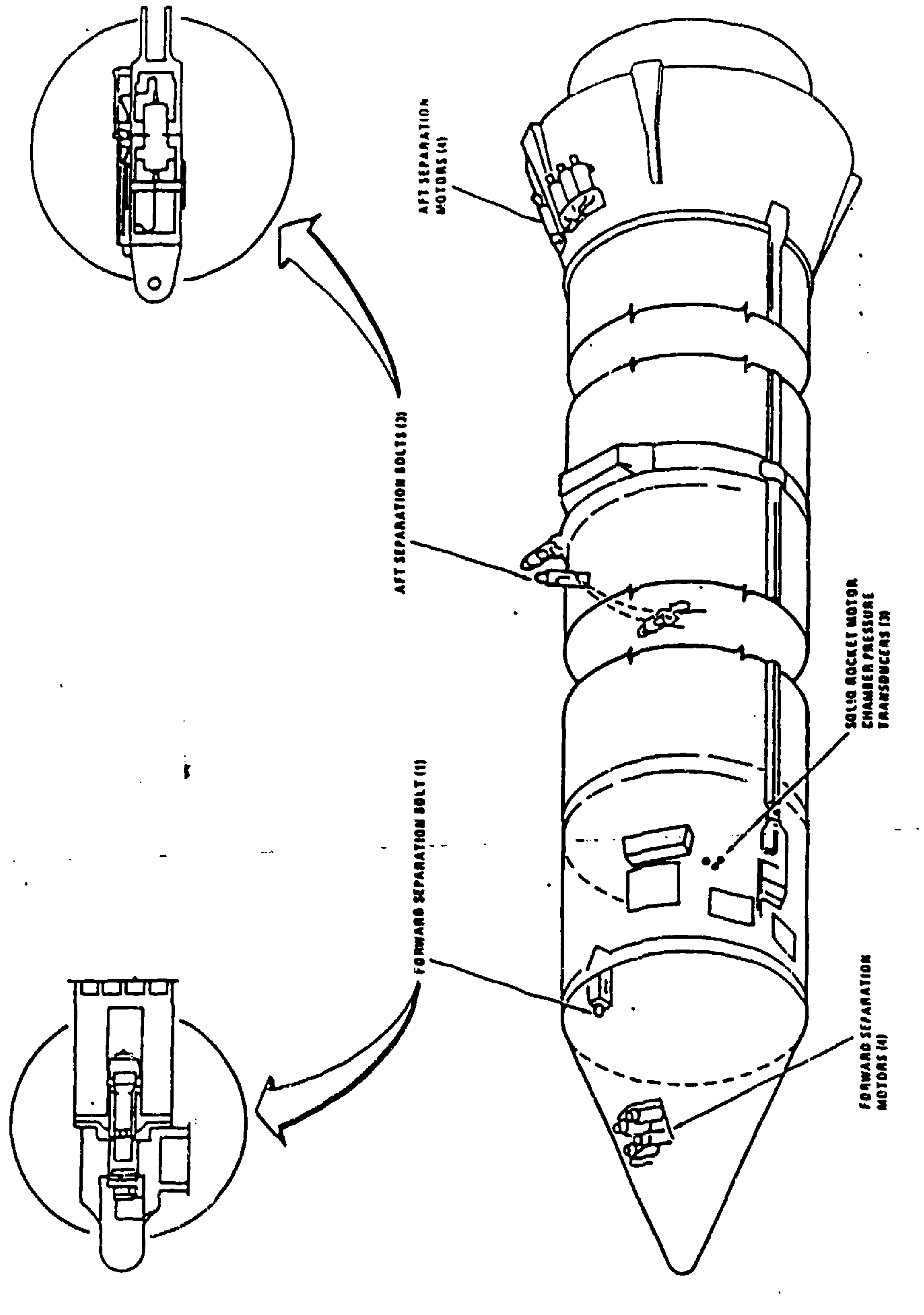


Figure CI(S)-25 Shutt_e SRB Separation System

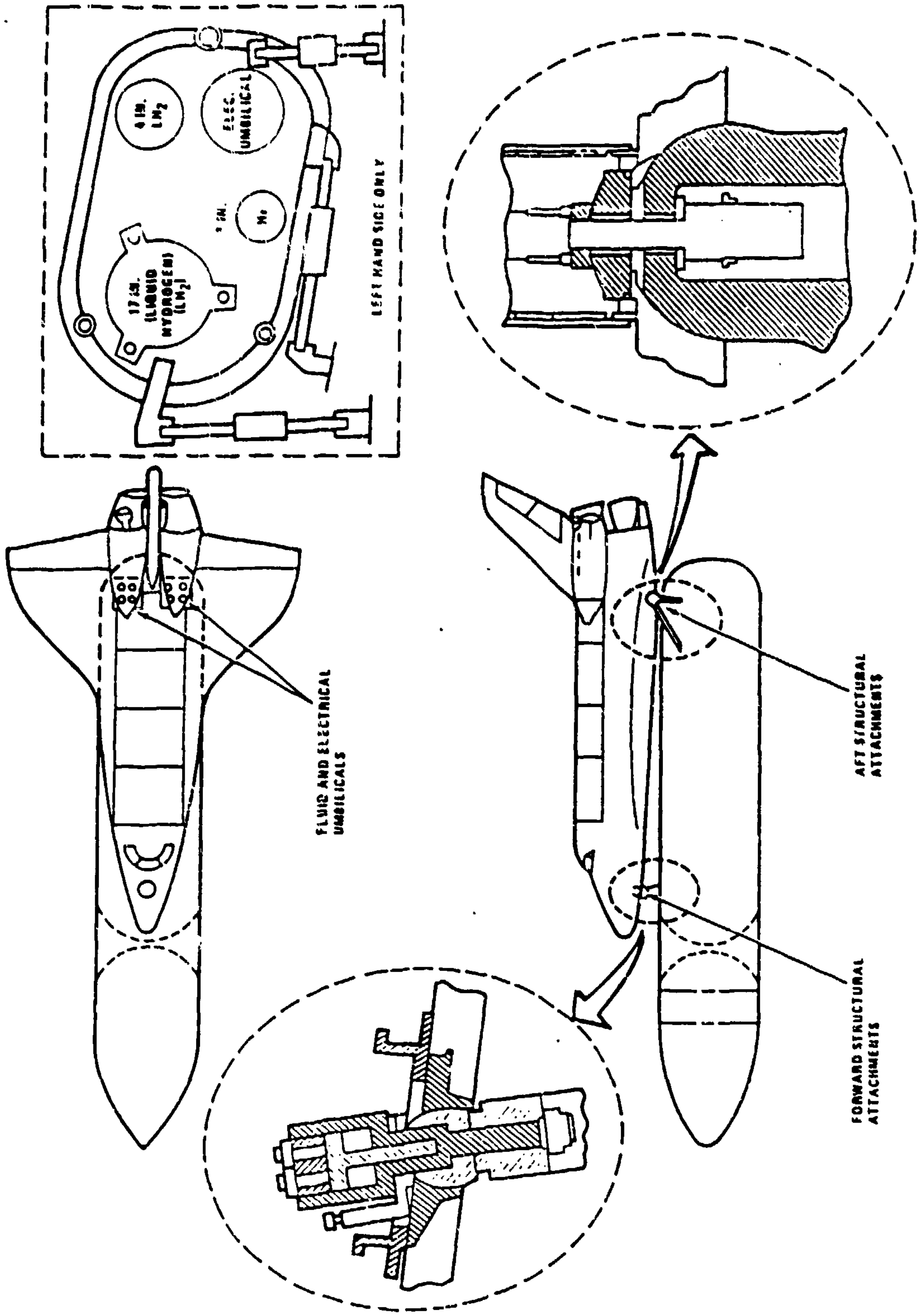


Figure CI(S)-26 Orbiter/ET Separation System

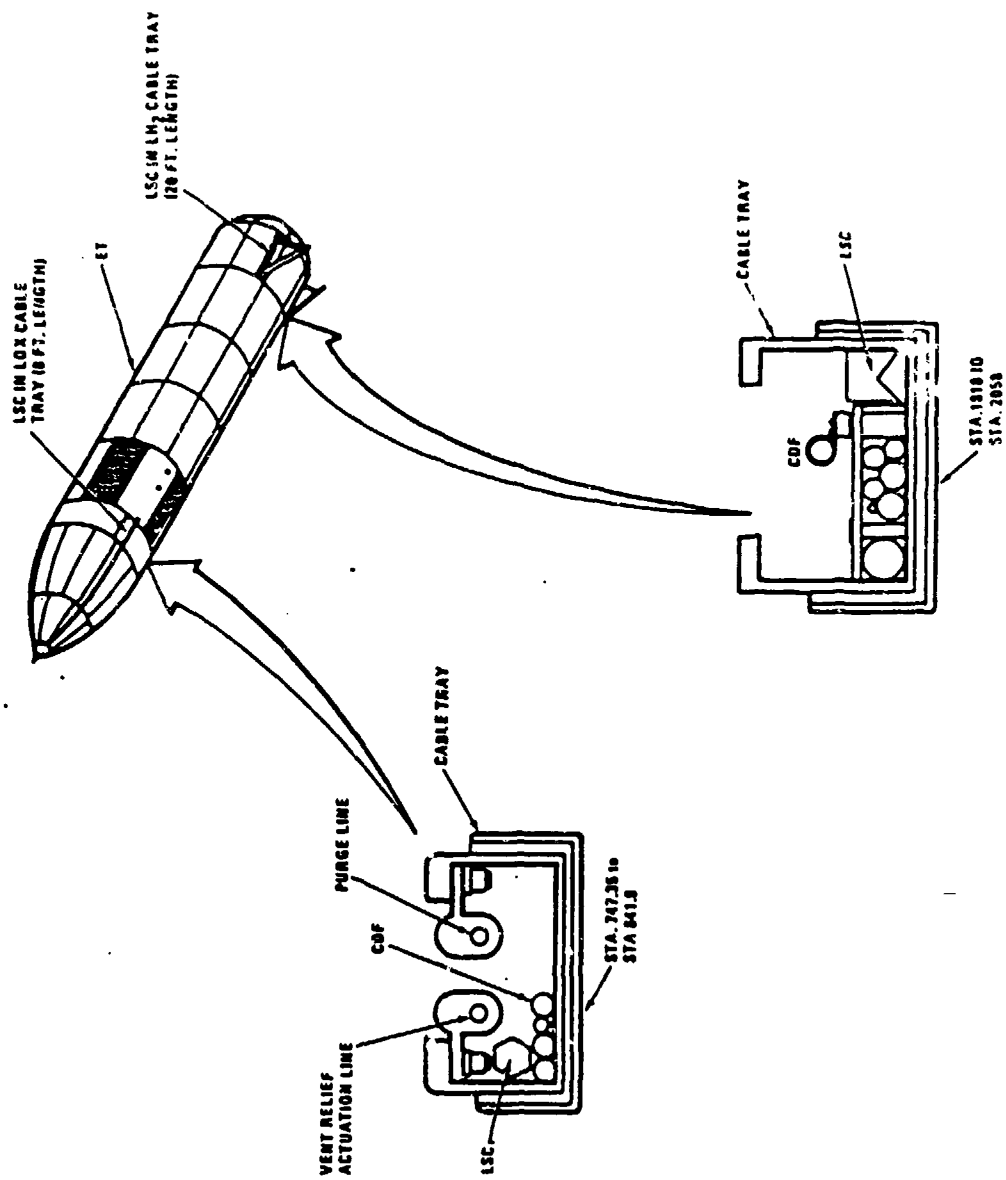


Figure CI(S)-27 Shuttle ET Command Destruct Charges

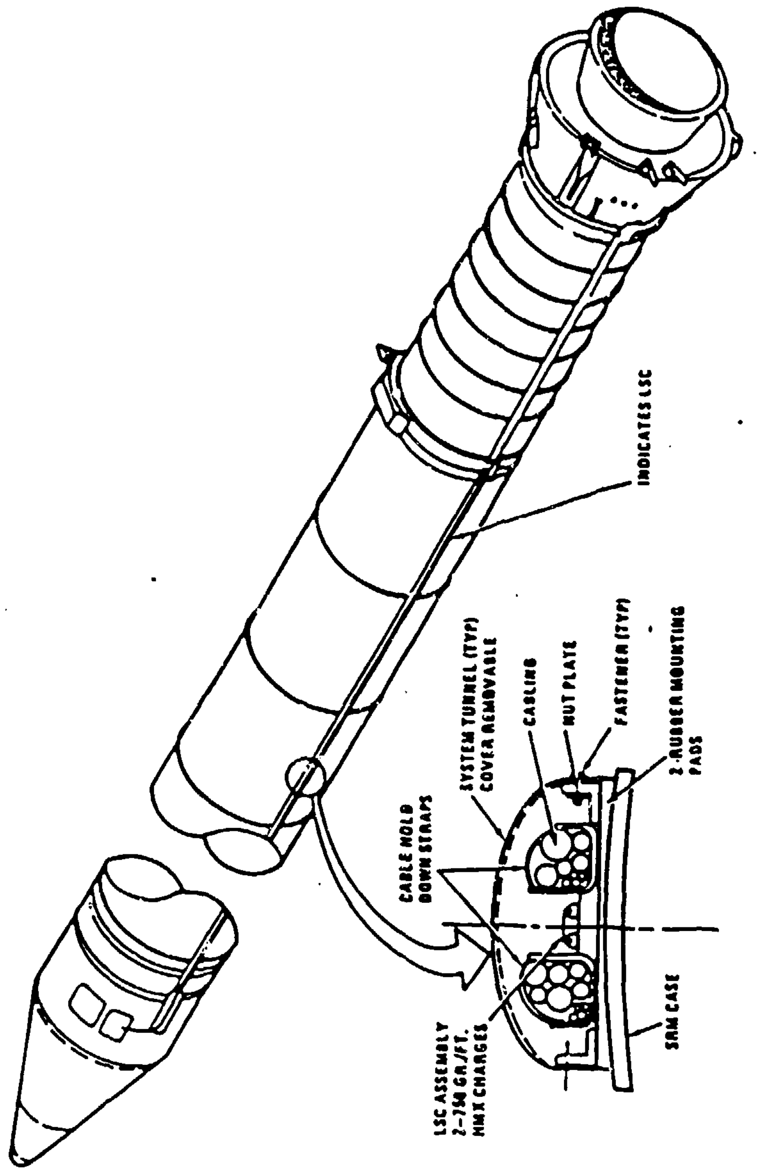


Figure CI(S)-28 Shuttle SRB Command Destruct Charges

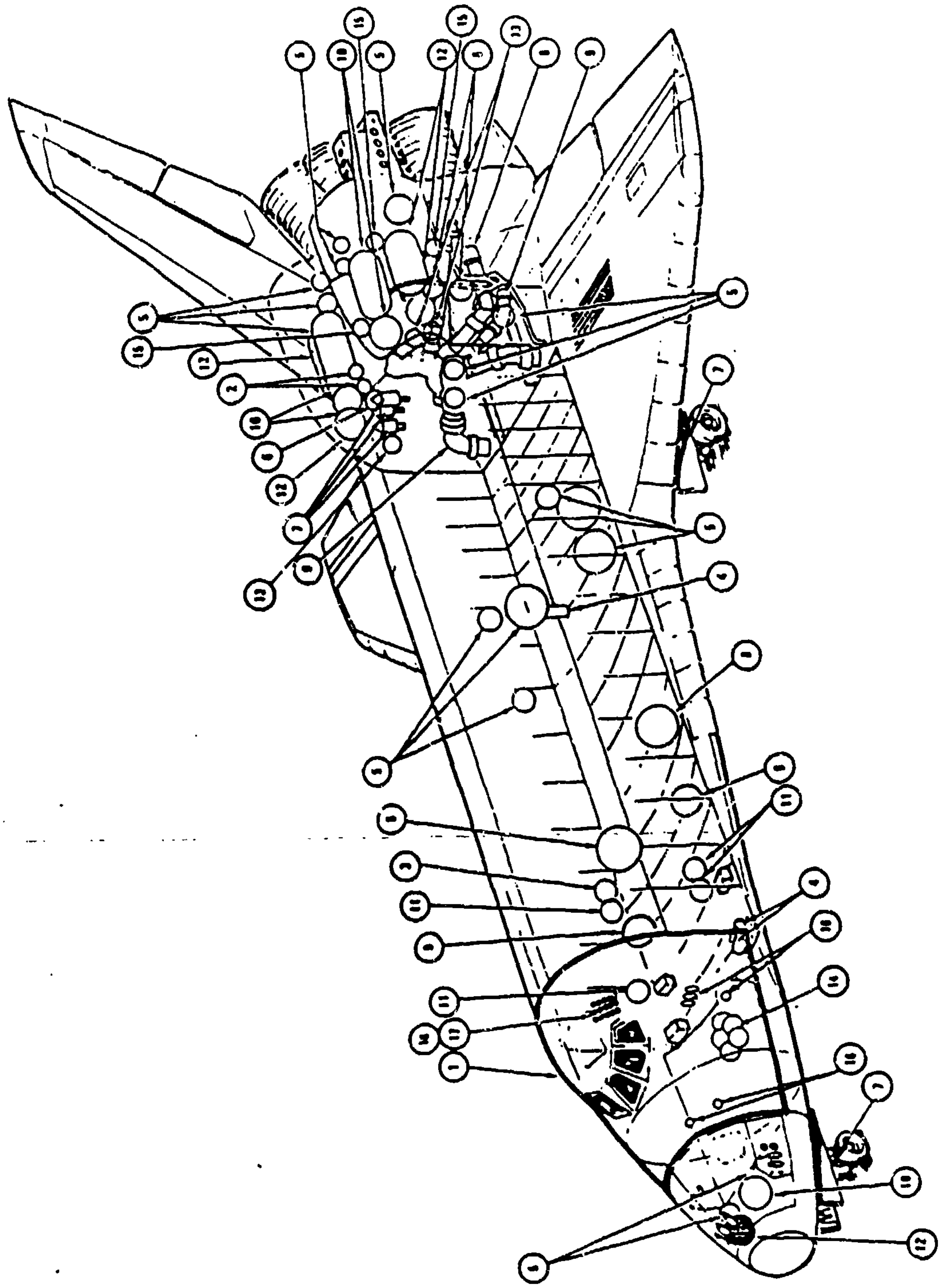


Figure CI(S)-29 Pressure Tanks in Orbiter
(Circled Numbers Correspond with Numbers in Table C1(S)-2

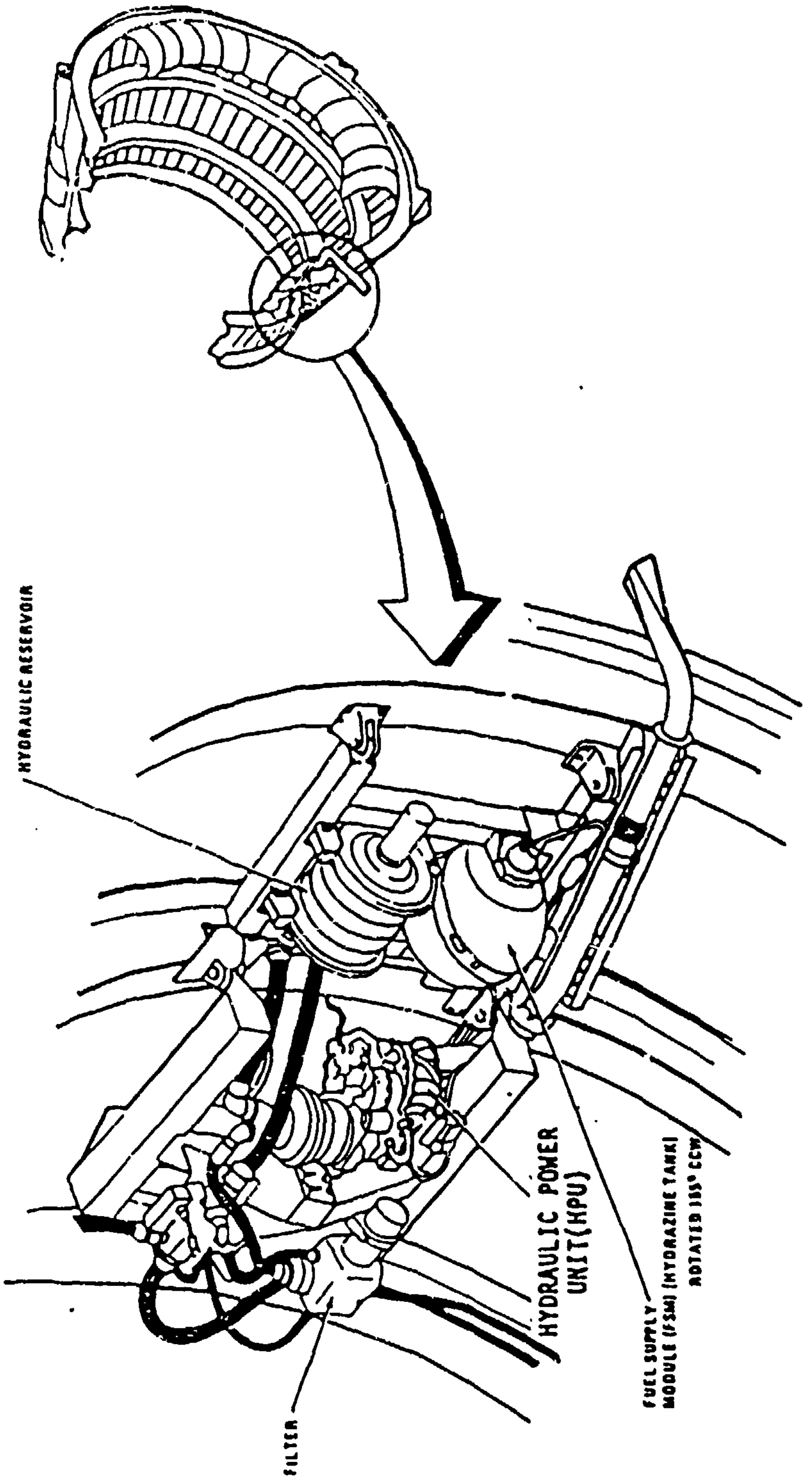


Figure CI(S)-30 Hydrazine Tank on a Shuttle SRB

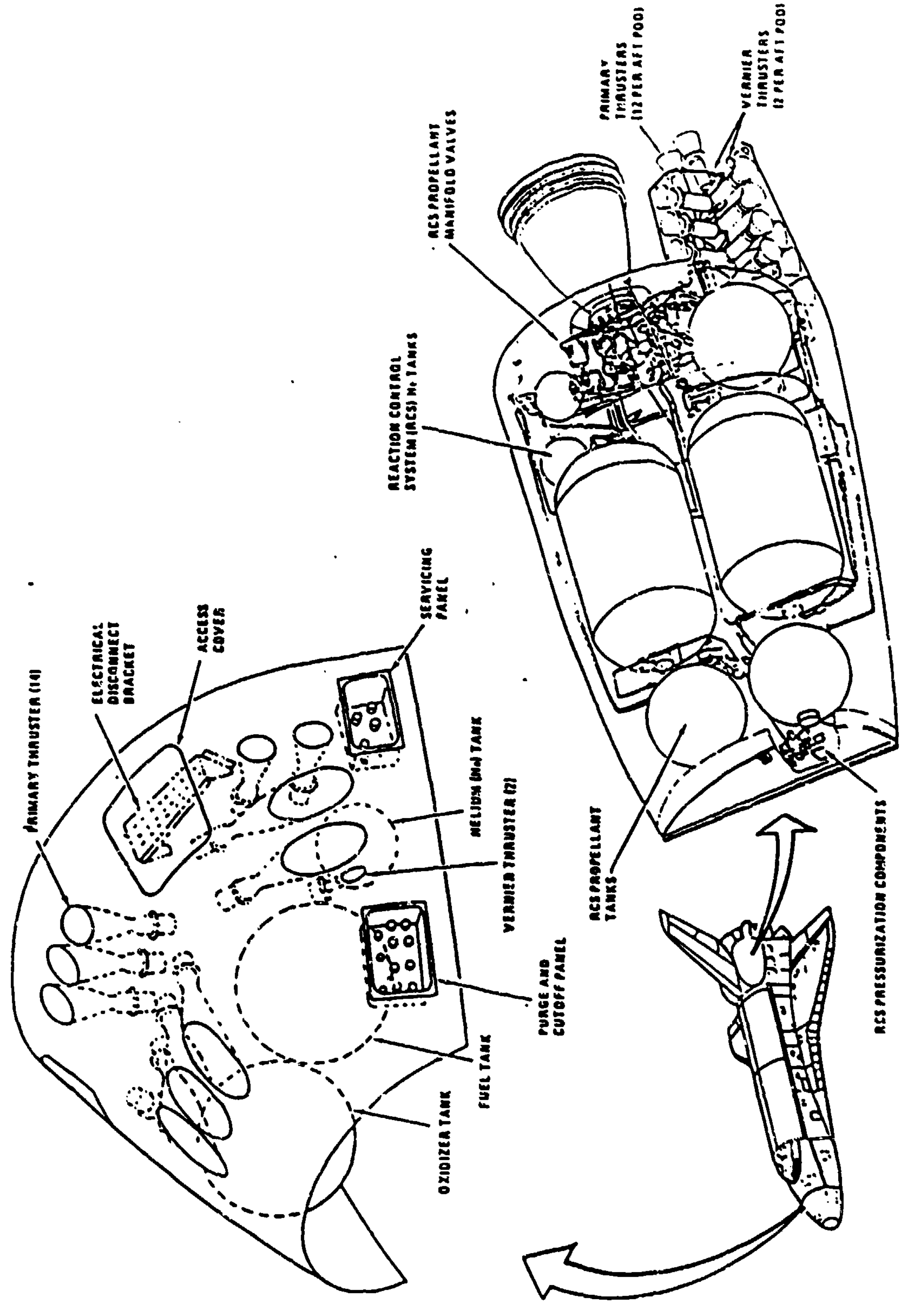


Figure CI(S)-31 Orbiter RCS

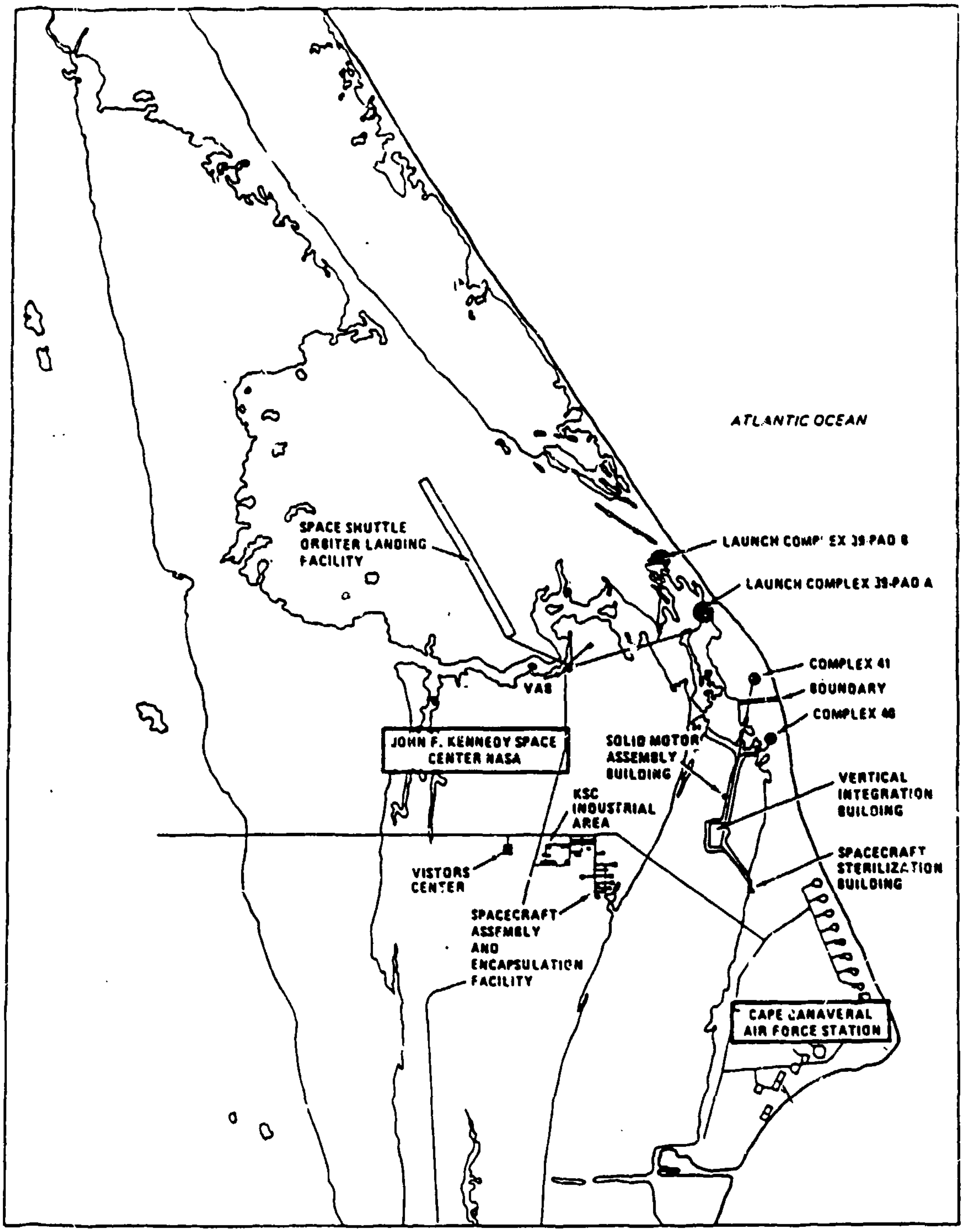


Figure CI(S)-32 Map of KSC and CCAFS

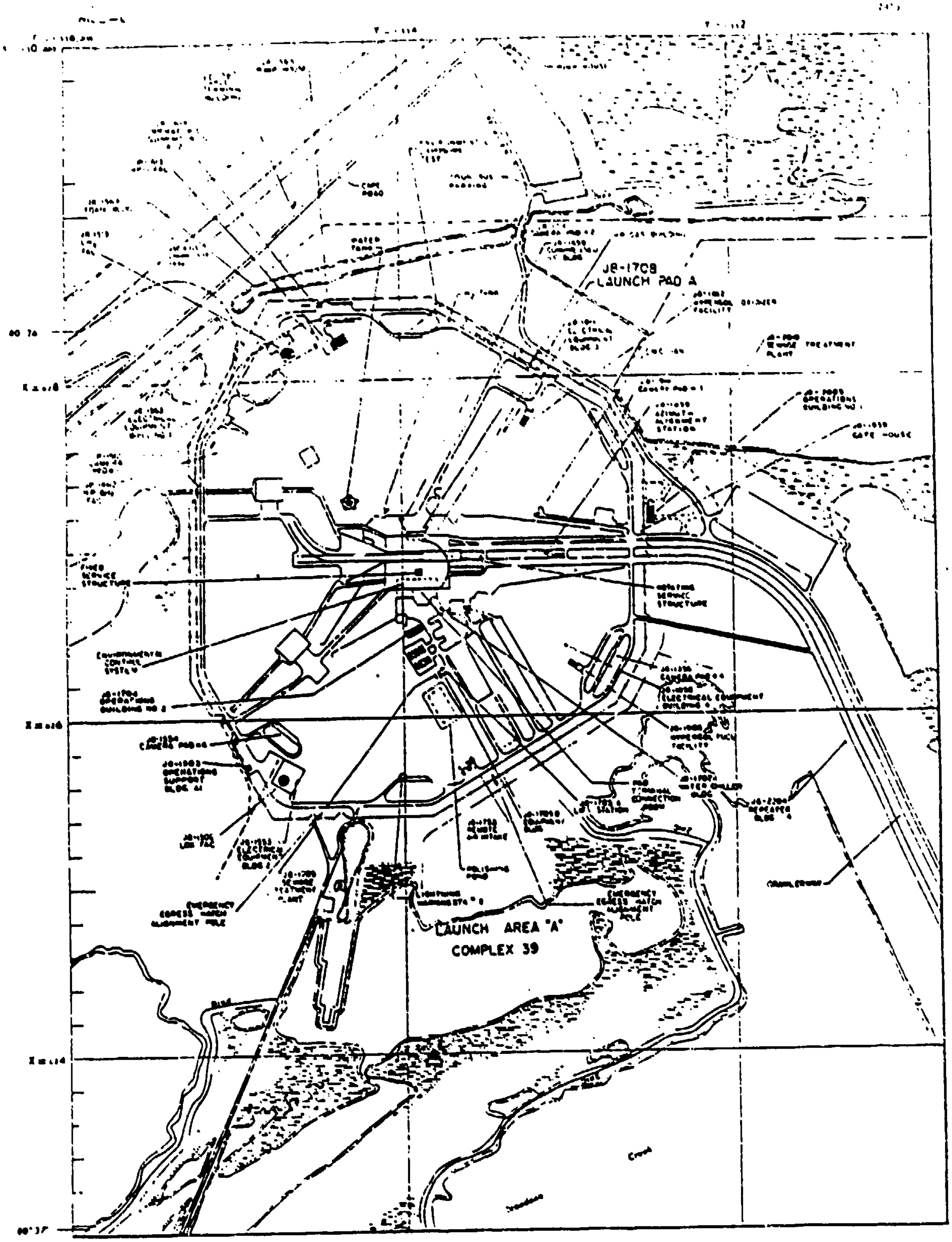


Figure CI(S)-33 Launch Pad 39A at KSC

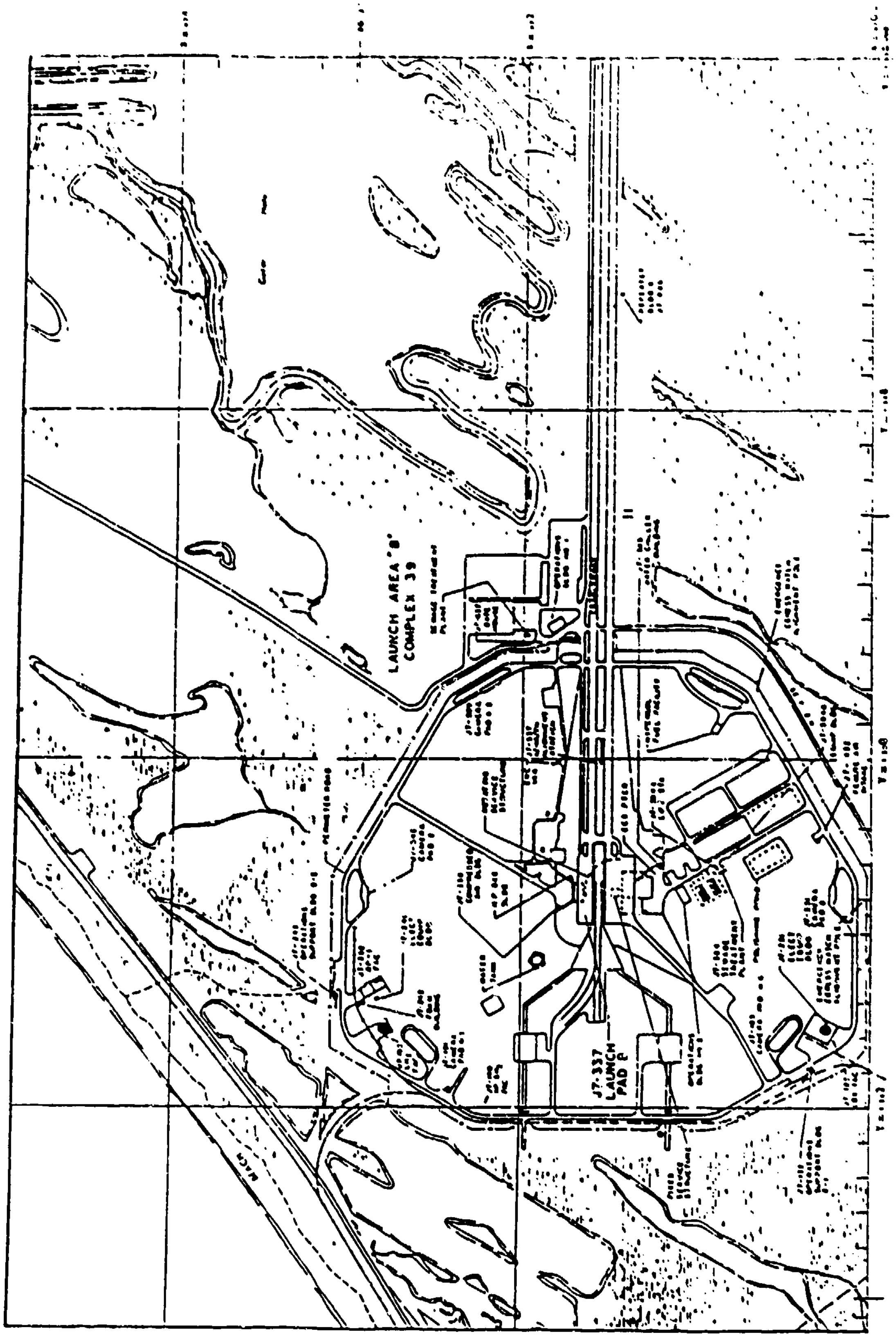


Figure CI(S)-34 Launch Pad 39B at KSC

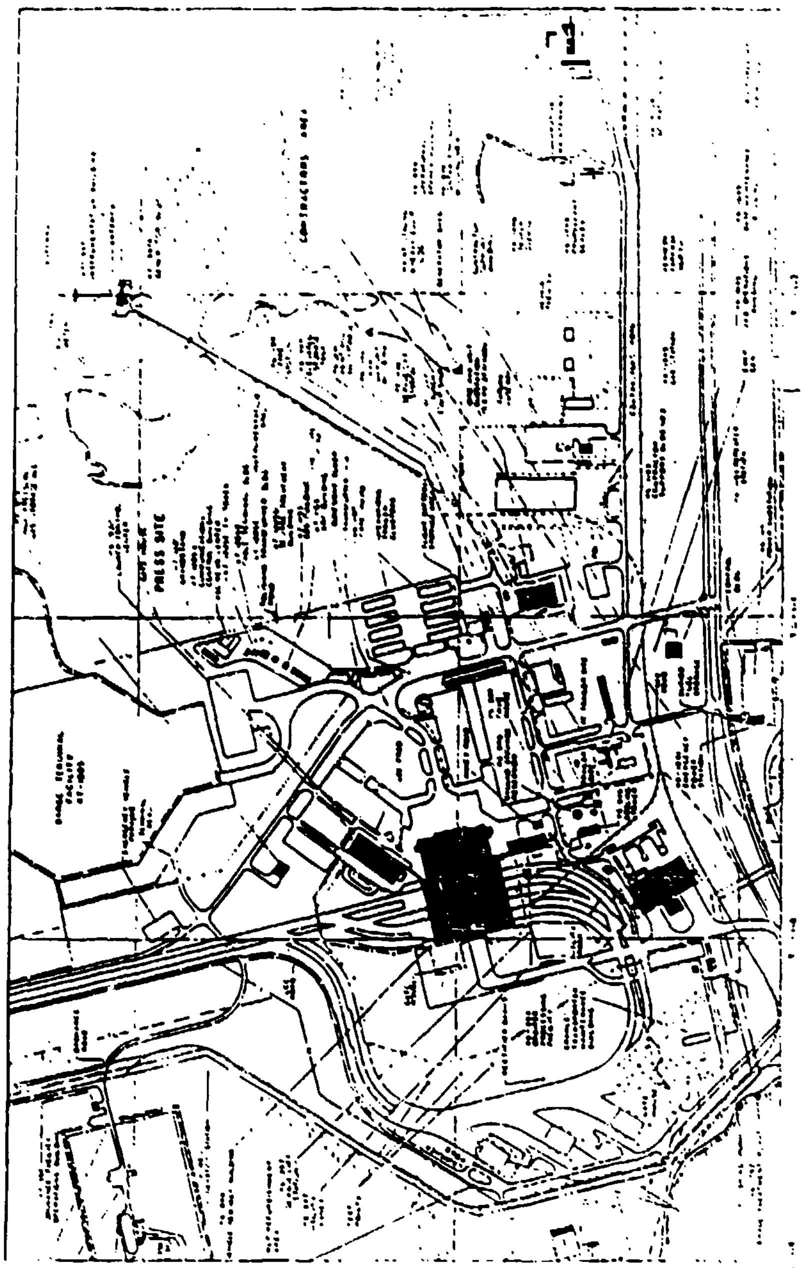


Figure C1(S)-35 Facilities Surrounding the VAB at KSC

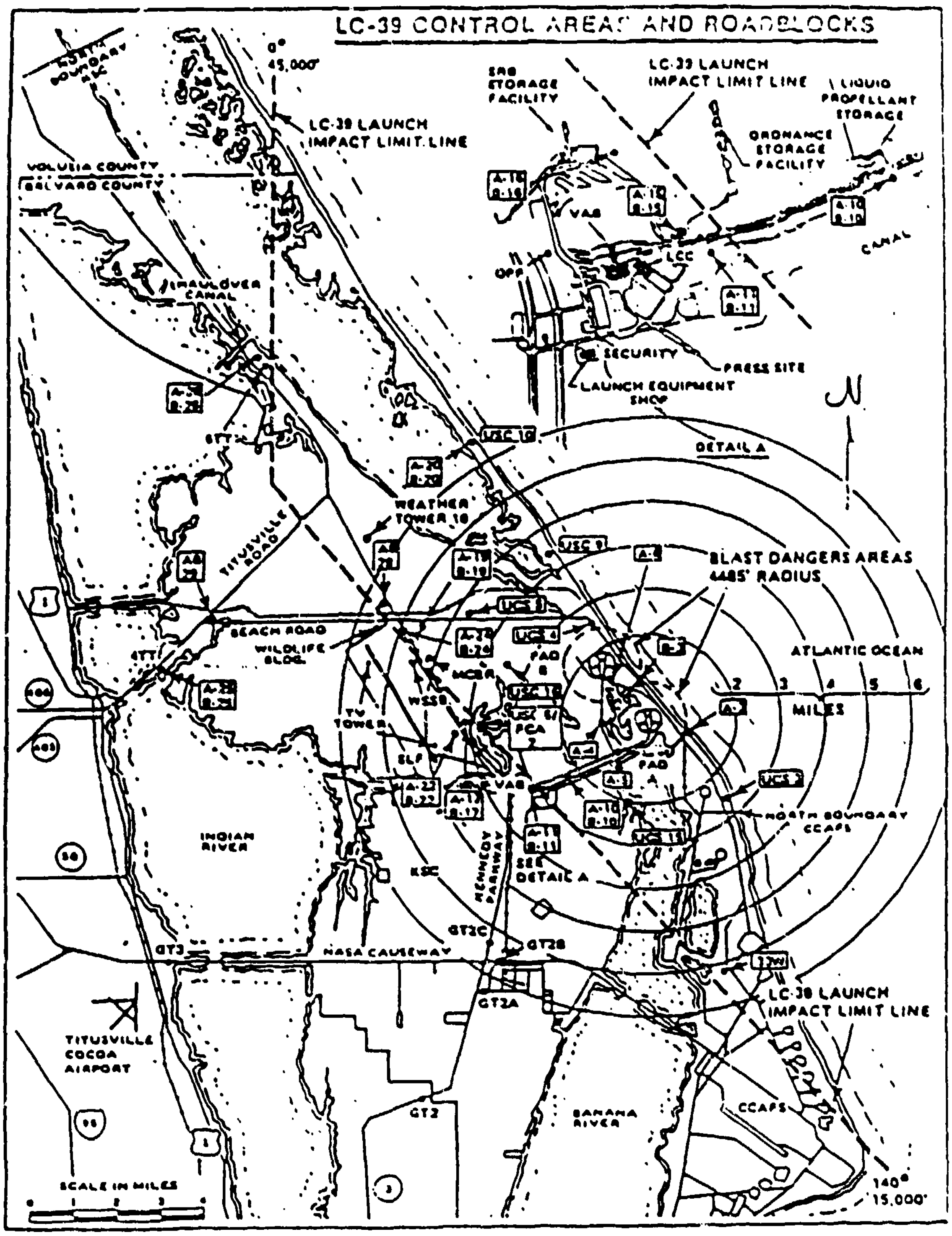


Figure CI(S)-36 Launch Pad Blast Danger Area
and STS Launch Impact Limit Line

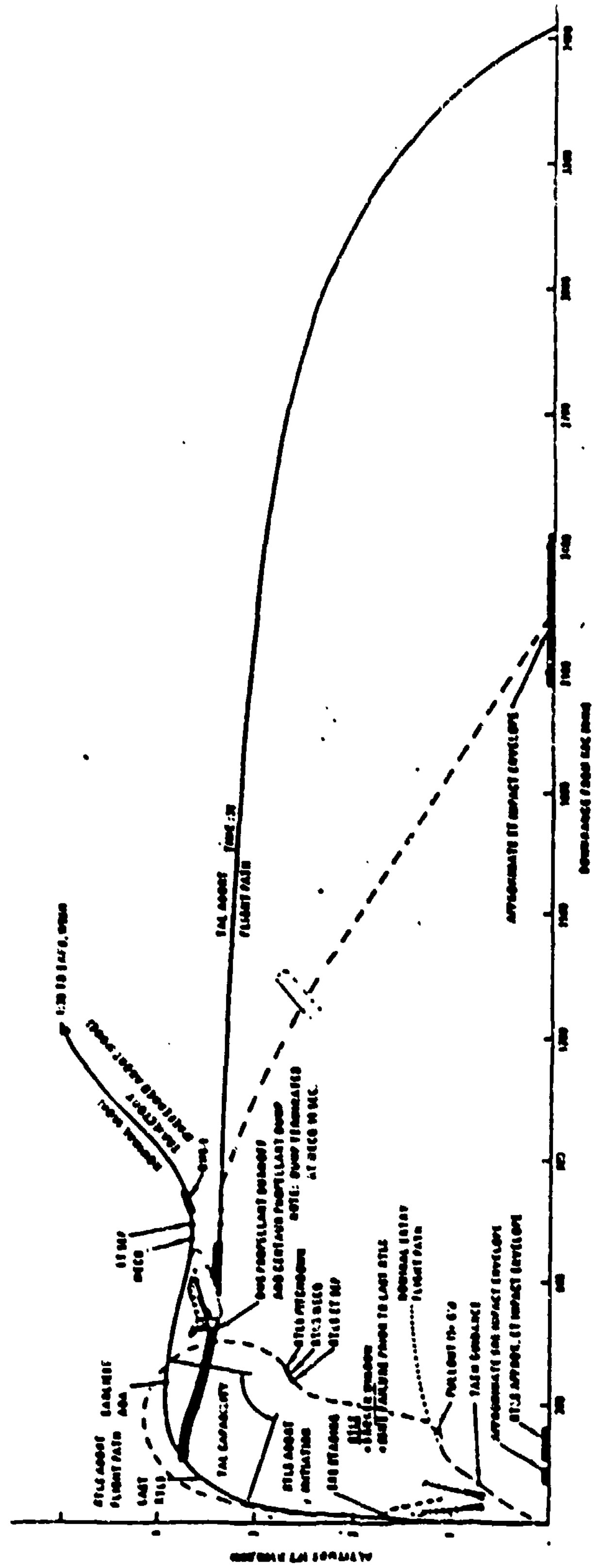


Figure CI(S)-37 RTLS & TAL Aborts Altitude Vs Range

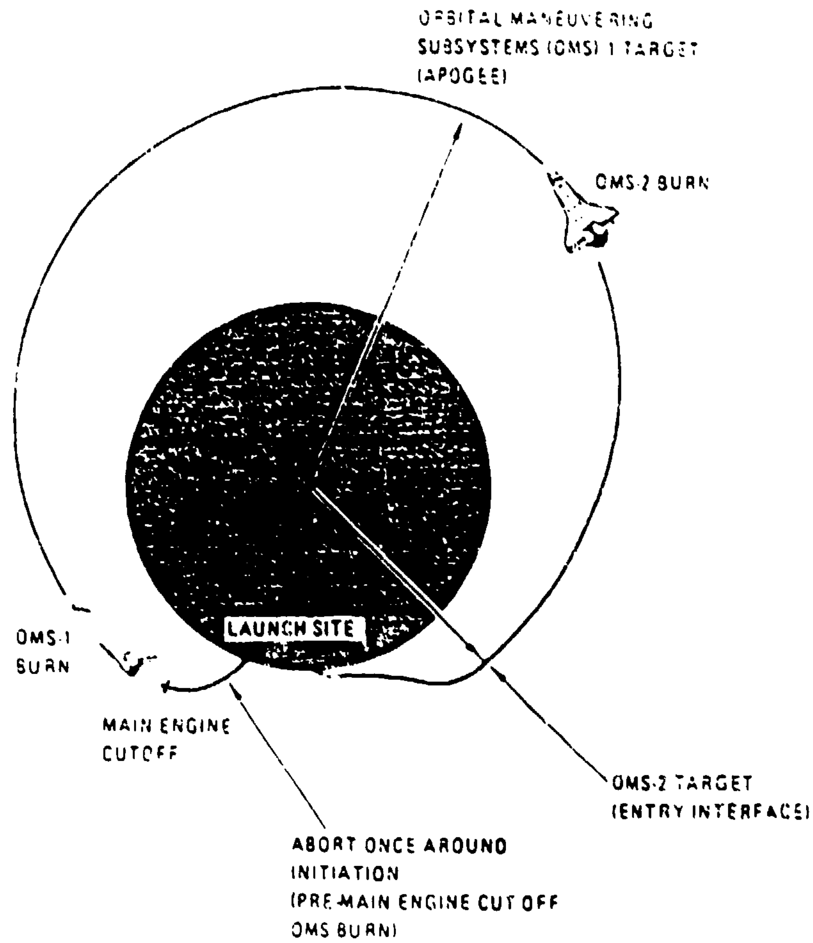


Figure CI(S)-38 Profile of AOA

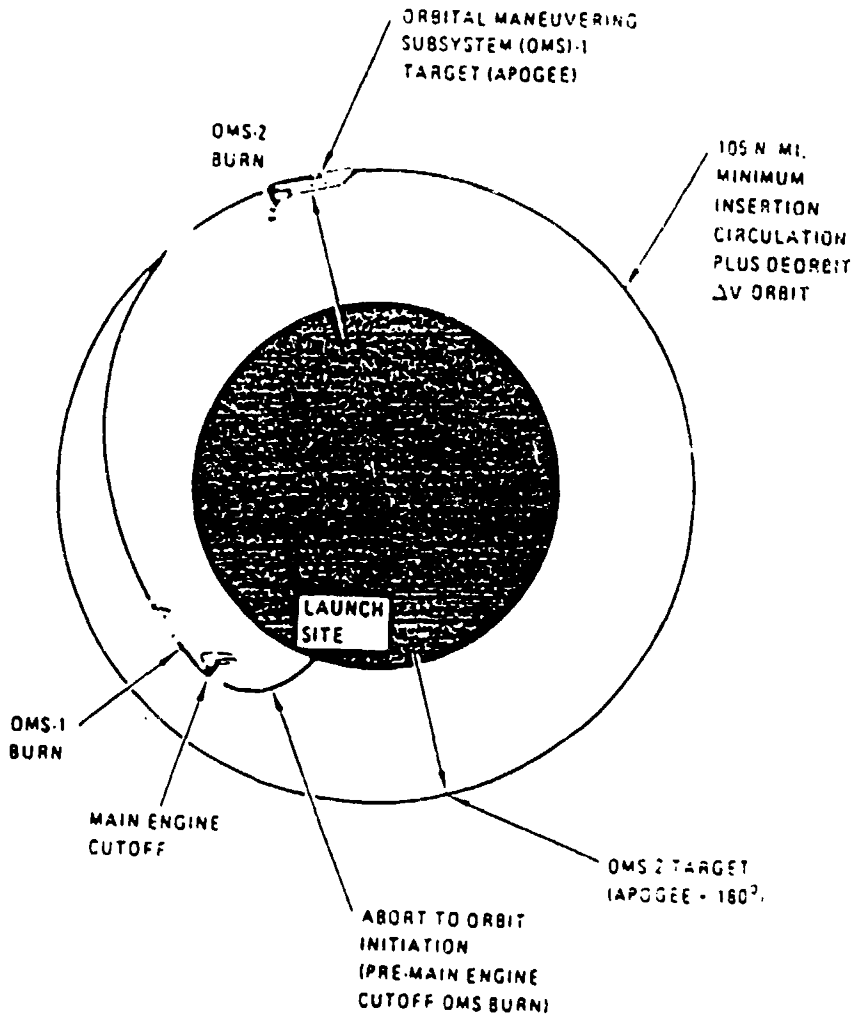


Figure CI(S)-39 Profile of ATO

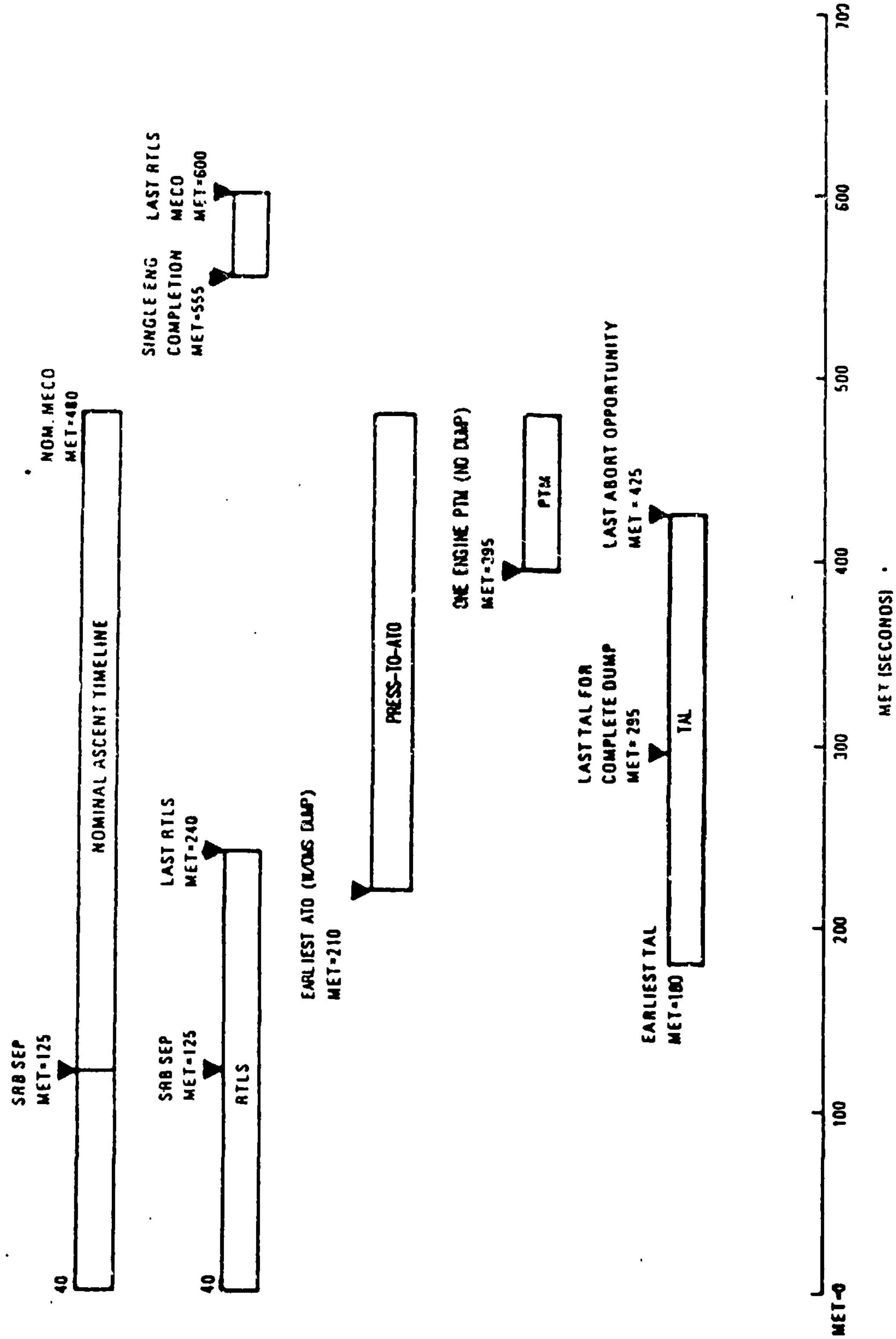


Figure CI(S)-40 Intact Abort Regions for Shuttle Missions

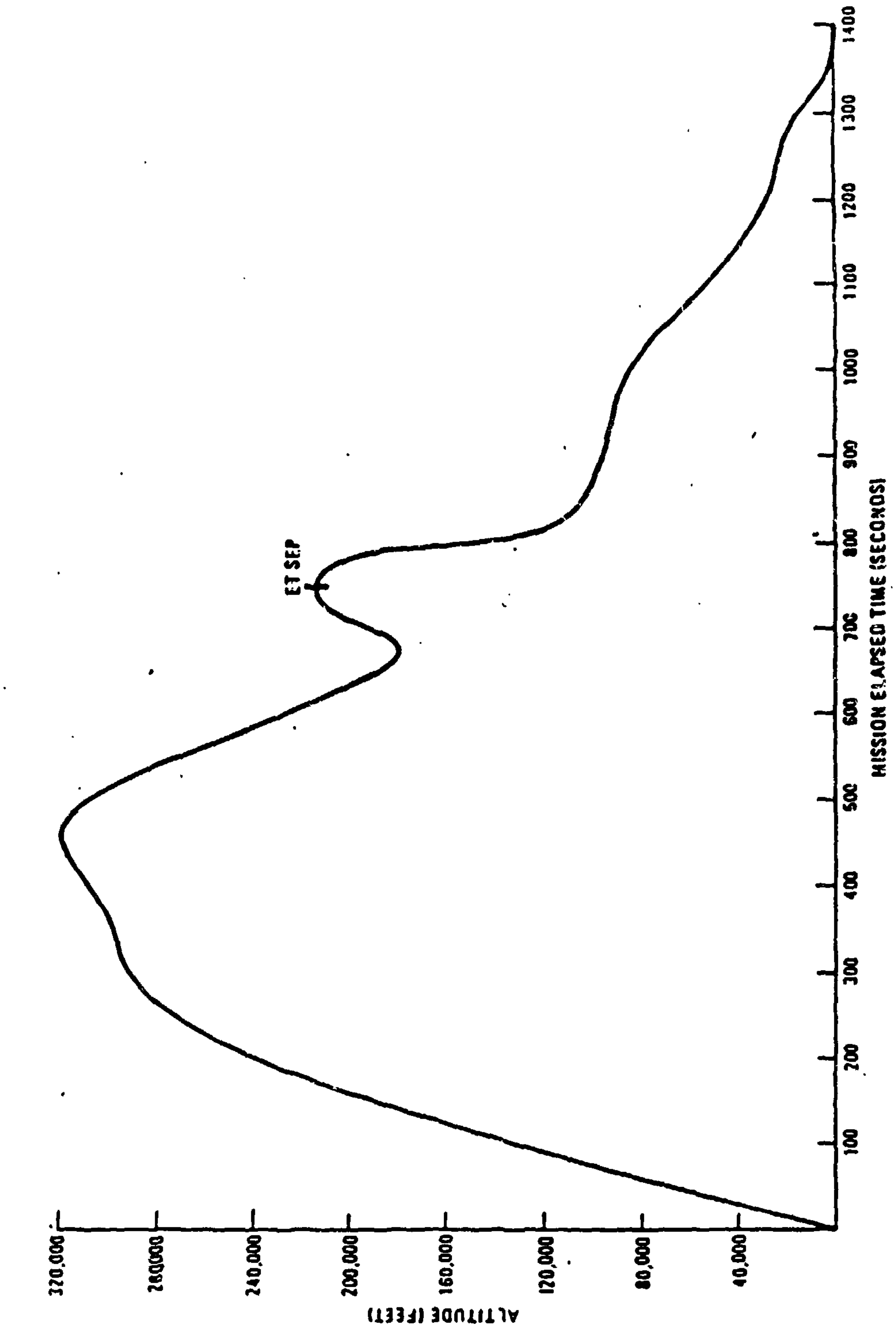


Figure CI(S)-41 Altitude Vs Time for Typical RTLS Type Abort

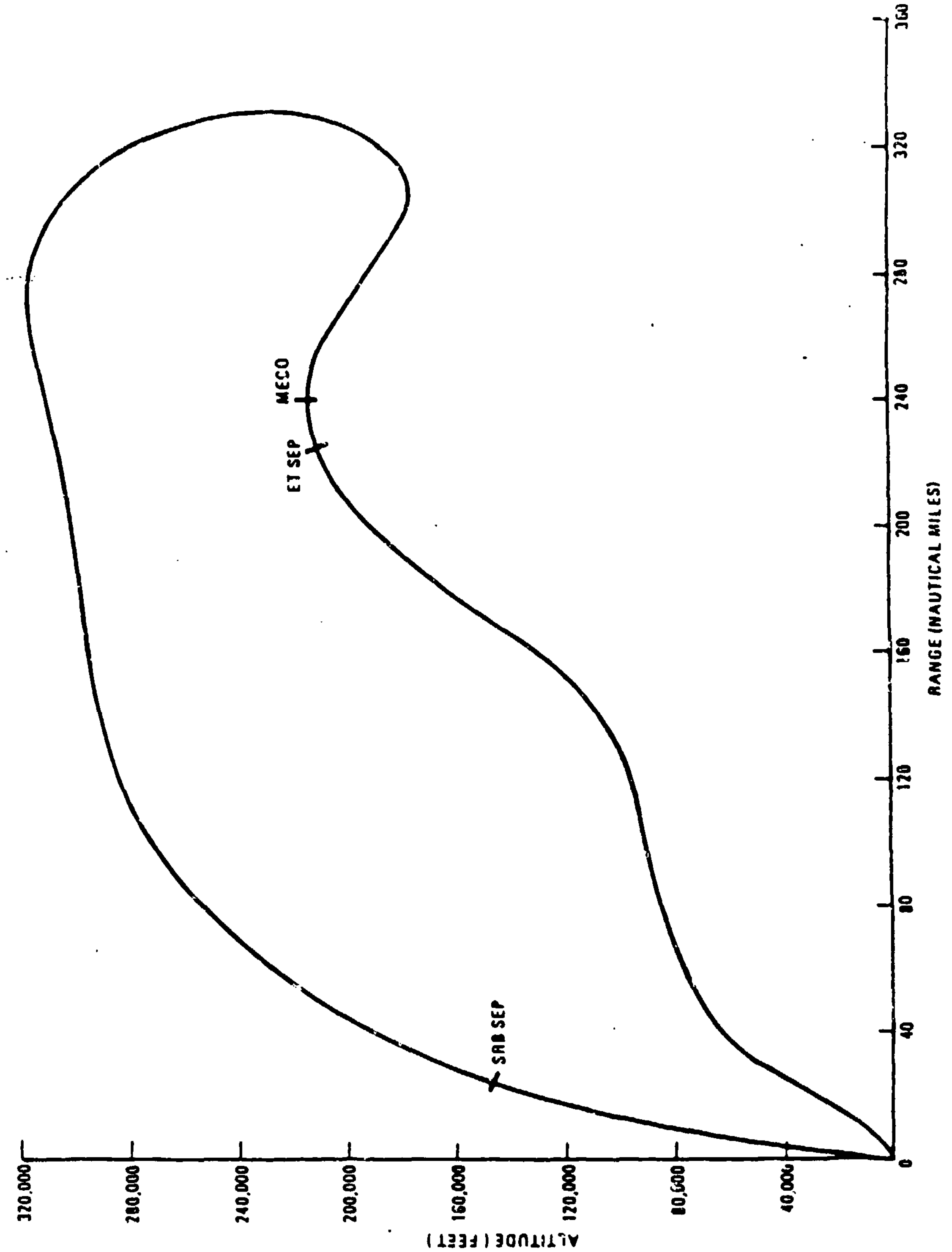


Figure CI(S)-42 Typical RTLS Abort Altitude Vs Range

Appendix C2
Titan 34D

APPENDIX C2
TITAN 34D

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APPENDIX C2
TITAN 34D

C2.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref No. 015, and the Accident Risk Assessment Report (ARAR) for Titan 34D/Transtage, MCR-82-071, Feb 1983, Martin Marietta Aerospace, Denver, Space Launch System Division.

C2.1 GENERAL DESCRIPTION

The T34D Common Core (T34K Standard Airborne Vehicle, CI No. T10D34K) is designed for maximum commonality between Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base (VAFB). The Common Core is configured with provisions for installing the unique airborne equipment of CCAFS (CI No. T04D34D) and two solid rocket motors. The VAFB configuration is known as T34D/Radio Guidance System (RGS) (SRM). Two configurations exist for CCAFS: the T34D/IUS and the T34D/Transtage.

The Titan 34D vehicle has evolved from the basic family of Titan Launch Vehicles: Titan III B, C, D, and E. The T34D is designed to use existing TIII proven design to the maximum extent and consists of a long core Titan Stage I and Stage II. A pair of f5 1/2-segment Solid Rocket Motors (SRMs) are strapped to Stage I to provide the initial thrust at lift-off.

Stages I and II of the Titan 34D vehicle use the same storable hypergolic liquid propellants. The fuel is Aerozine 50, a 50/50 mixture of hydrazine and unsymmetrical dimethylhydrazine (UDMH), and the oxidizer is nitrogen tetroxide (N_2O_4). The use of propellants storable at ambient temperature and pressure eliminates the holds and delays inherent in handling cryogenic propellants. This feature gives Titan the demonstrated capability of meeting critical launch windows within two seconds of any pre-established time. The hypergolic action of Aerozine 50 and N_2O_4 eliminates the need for an ignition system and related checkout and support equipment.

C2.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

In referencing various parts of the vehicle, two reference systems are used. They are the vehicle compartment designators and the vehicle's three-axis reference system. Figure C2-1 shows the alphanumeric compartment designators. The compartment numbers indicate the associated stage, and the letters are assigned from the top of the stage down. Vehicle station orientation is also shown. It should be noted that positive is in the downward direction. The vehicle three-axis reference system with respect to Launch Complex-40 (LC-40) is depicted in Figure C2-2. The outboard profile for T34D/IUS in the Cape Canaveral Air Force Station configuration is shown in Figure C2-3.

C2.2.1 Titan Stage 0 (Solid Rocket Motor)

Initial thrust for the Titan 34D vehicle is provided by two government-furnished United Technologies - Chemical Systems Division (UT/CSD) solid rocket motors. The solid rocket motors are also referred to as Stage 0. This stage consists of two identical, segmented, solid propellant rocket motors. These 120-inch-diameter motors are mounted 180 degrees apart on the Titan liquid-propellant, long-core vehicle. Each motor consists of a forward closure, an aft closure, and five and one-half segments (Fig. C2-4). The added 68-inch solid motor segment has replaced the structural spacer, allowing the SRM forward attachment to mate with the long-core vehicle at the same relative location as on the short-core vehicle. Other components include a single six-degree canted nozzle, an igniter, fore and aft solid propellant staging rockets, and a liquid-injection Thrust Vector Control (TVC) system. The TVC injectant, nitrogen tetroxide (N_2O_4), is carried in a tank mounted on the side of each motor and is pressure-fed into the nozzle exit section by gaseous nitrogen.

The motor case (both segments and closures) is constructed of D6ac steel that has been heat-treated to an ultimate strength of 195,000-220,000 psi. Each joint is a pin-and-clevis type held together by 240 cylindrical pins and is held in place by a retaining strap. An O-ring held within the joint maintains a propellant gas pressure seal.

Each center segment contains approximately 72,400 pounds of propellant consisting of powdered aluminum fuel, ammonium perchlorate oxidizer, and a binder of polybutadiene acrylic acid acrylonitrile (PBAN). The case-bonded propellant grain has a circular port, tapering 10 inches through the 10-foot length of the segment. The forward end has a smaller port. The purpose of this taper is to provide a 10-second controlled tailoff at the end of web-action time. The forward end of the segment is inhibited from burning by a rubber restrictor bonded to the propellant surface. Silica-filled butadiene acrylonitrile rubber insulation protects the motor case from exposure to combustion gases during motor operation. The insulation is thickest in the segment joint areas where there is no unburned propellant to protect the case walls.

The closures contain the same type of propellant as the segments, with the forward closure having mounting provisions for the solid propellant igniter. The forward closure has an eight-point star internal burning grain configuration instead of the cylindrical grain shape of the segments.

The forward closure Fig. C2-5, is 95 inches long and contains 39,100 pounds of propellant. The aft closure Fig. C2-6, contains approximately 20,300 pounds of propellant in a straight cylindrical bore configuration and projects 64 inches from the segment joint to a 57-inch-diameter boss for nozzle attachment.

The propellant burns along the entire central port of the SRM and on the aft of each segment between segments. The closures also burn on their ends with three inches of clearance left between segments to permit this burning. The igniter burns for approximately one second to fill the grain bore with hot gas and ignite the motor.

The SRM segment is described as follows:

SRM Segment

Manufacturer and Manufacturer's Part No.:
CSD, A04376-01 (A04376-03-01)

Type of Explosive and Weight:

Composite Propellant	73,400 lb (73,200 lb)

Propellant Composition (Percentage by Weight):

Ammonium Perchlorate	68%
Aluminum Powder	16%
PBAN	10%
Additives	6%

Propellant Characteristics:

Autoignition Temperature	475° for 1 hr
Volatile (toxic) Vapors	None
Static Electricity	Not Susceptible

Grounding Provisions:
No Special Provisions Other Than Static Ground Strap

Storage Requirements:

Relative Humidity	Uncontrolled
Temperature	40°F Min

Classification:
DOT Class B Solid Propellant
Military - Class 2 (1.3)

The SRM nozzle consists of a throat section and a two-piece exit cone assembly. The nozzle throat section is made of high-density graphite rings backed by a steel support shell and silica insulation bonded in a steel housing. The nozzle middle section consists of graphite and silica phenolic liners bonded to a steel outer shell and contains the thrust vector control injection ports. The exit section is an extension of the silica phenolic liner of the middle section except that its structural shell is an aluminum honeycomb sandwiched between steel for lighter weight. The three sections are bolted together forming an assembly approximately 11.5 feet long. Nozzle expansion ratio is 8:1, and the half-angle is 17 degrees.

The thrust vector control system used for the Titan 34D 120-inch solid rocket motor provides side forces on each motor in response to command signals from the Titan vehicle flight control computer. Nitrogen tetroxide (N_2O_4) is injected into the nozzle exit cone in each of four quadrants to provide side force in any direction normal to the flight control system which computes, based upon actual use, the amount of excess injectant fluid and provides the necessary valve commands to dump it.

The Thrust Vector Control (TVC) tank is 42 inches in diameter and approximately 22 feet in length with a total tank weight of 3817 pounds. As illustrated in Fig. C2-7, this is a single tank structure using a ullage blowdown system. The TVC tank has a nominal load of 8424 pounds of N_2O_4 and an initial pressure of 1030 psia that reduces to a minimum of 450 psia at SRM burnout.

The SRM liquid injection TVC system is capable of a vector angle of five degrees and a maximum side force of 110,000 pounds per motor. The TVC system has 24 valves located at an area ratio of 3.5:1. These valves are uniformly spaced on the periphery of the nozzle and operate in groups of six per quadrant. The valves modulate from zero position to full open and are controlled by electrical signals (0-10 volts) from the core vehicle. Nitrogen tetroxide is supplied to the valves through a toroidal manifold mounted above them. A single feed line transfers the TVC injectant from the tank to the nozzle distribution manifold.

Translation rockets are mounted perpendicular to the SRM centerline to provide the required translation force to move it free of the core engines upon release of the SRM-to-core attachments. Eight translation rockets are mounted on each SRM, four forward and four aft.

The translation rocket is a cylindrical solid-propellant motor comprising an aluminum case, a canted nozzle, and a case-bonded internal-burning seven-point star propellant grain, with burning on the aft end. The motor is six inches in diameter and 64 inches long.

The propellant charge consists of a Polybutadiene Acrylic Acid Acrylonitrile (PBAN) fuel binder, ammonium perchlorate oxidizer, and an aluminum additive. Nominal charge weight is 45 pounds. The rockets are supplied by United Technology Center and have Ordnance Item Number PD050003-501.

The Titan 34D solid rocket motors use a variety of ordnance devices to actuate certain internal systems and motor functions. The ordnance devices include motor igniter and motor, separation rockets, and destruct charges.

Ordnance devices will be received, stored, and inspected by either United Technologies Corporation (UTC) or the Air Force Eastern Test Range (AFETR) Range Contractor prior to being installed or assembled in the SRM. Work involving ordnance materials, propellant, and pressurized systems are performed in accordance with detailed procedures, as in all work performed on the solid motors. The procedures detail all requirements for performance of each task, including specific safety requirements if appropriate. The completed procedures become part of the records showing completion of each solid motor assembly.

All solid propellants used in the booster are Class 2 and are a fire hazard if ignited. They cannot be extinguished with normally available fire extinguishing equipment. If sufficiently confined in a closed container, the reacting propellant can overpressurize the container and create an explosion that may cause damage by both blast and projected debris. Partial confinement in a vessel, such as in an assembled motor, may result in propulsiveness rather than an explosion. Detonation of the propellant in the configurations present in the Titan 34D system has never occurred.

Prevention of propellant ignition is essential. Ignition of the propellant will result from direct contact with even a very small flame. It can be ignited by heat generated by frictional rubbing of the propellant and by heat generated from sudden impact between two surfaces of hard material (such as metal and metal or metal and concrete). Ignition by static electrical spark has never been experienced, and laboratory efforts to ignite it by this means have been unsuccessful.

Table C2-1 presents pertinent information on the ordnance devices.

Table C2-1
Titan 34D SRM Ordnance Items

ITEM	DESCRIPTION	CLASS[1]	CATEGORY[2]	QTY PER[3] FLIGHT	PURPOSE OF DEVICE
1	SRM Segment, solid grain	2 (1.3)	-	10	Propulsion
2	SRM half segment, solid grain	2 (1.3)	-	2	Propulsion
3	SRM Forward Closure	2 (1.3)	-	2	Propulsion
4	SRM aft closure solid grain	2 (1.3)	-	2	Propulsion
5	Igniter/ initiator	3 (1.2)	-	4	Ignite the SRM
6	S/A device, ignitor	3 (1.2)	A	2	Receive the ignition signal and ignite the igniter initiator
7	S/A device, destruct (motor)	3 (1.2)	A	2	Receive destruct signals and detonate destruct transfer fuses (harnesses)
8	LSC destruct (motor)	7 (1.1)	-	12	Cut linear strip from each segment and allow exhaust gases to escape and internal pressure to decrease rapidly
9	Jumper fuse, destruct (SRM/TVC)	3 (1.2)	-	20	Transmit detonation wave (destruct)
10	Transfer fuse, destruct (motor)	3 (1.2)	-	2	Provide explosive link to the LSCs from the S/A device
11	LSC destruct (TVC)	7	-	4	Cut linear strip from N204 tank

Table C2-1
Titan 34D SRM Ordnance Items
(Continued)

ITEM	DESCRIPTION	CLASS[1]	CATEGORY[2]	QTY PER[3] FLIGHT	PURPOSE OF DEVICE
12	Tube, explosive harness	3 (1.2)	-	2	Transmit detonation wave (destruct from motor destruct system to ?? of TVC destruct system
13	Staging rocket motor	3 (1.2)	-	16	Provide thrust to rotate and propel SRMs away from core at separation
14	Igniter, staging rocket motor	3 1.2)	A	16	Ignite staging rocket motor

(1) Military explosive classification per AFM 127-100
(2) Category A device - EEDs which by expenditure of own energy or initiation of chain of events may cause injury or death to people or damage to property
(3) Two SRMs per flight

C2.2.2 Titan Stage I

The Titan 34D Stage I airframe (Fig. C2-8) is an all-aluminum structure designed to contain propellants, provide support for SRMs and Stage I subsystems, and support the upper stages. It also contains attachments for the solid rocket motors and distributes thrust loads from all propulsion systems.

The Titan Stage I airframe structure consists of an extended fuel tank with its longeron and skirt sections, an extended oxidizer tank and its skirt sections, and an engine heat shield. The combined extension of the fuel and oxidizer tanks is approximately 68 inches longer than the TIIIC configuration.

The fuel tank assembly consists of an extended fuel tank with forward and aft skirt assemblies and an engine mount truss at the apex of the tank conical bottom. The fuel tank wall also serves as the vehicle's exterior skin. The tank proper is an all-welded aluminum unit with stringer and frame-reinforced walls, conical bottom, and an elliptical dome top. Special extruded frames join the tank walls to the top and bottom. Through the center of the Titan Stage I fuel tank is a 13-inch-diameter conduit which allows passage of the oxidizer feed line to the engines.

The oxidizer tank assembly includes the extended oxidizer tank with forward and aft skirts. The forward tank structure is similar to the fuel tank's forward dome. However, the oxidizer tank bottom is an inverted dome rather than a cone.

C2.2.2.1 Propellant Feed System - The propellant feed system for Titan Stage I is illustrated in Figure C2-9. Both fuel and oxidizer lines terminate at a set of electrically operated reclosable prevalves. Propellants are loaded into the vehicle tanks through manually operated disconnects above the prevalves. The Titan Stage I engine feed system also includes a set of toroidal accumulators in the fuel feed lines and oxidizer metal bellows accumulators in the oxidizer feed lines. These accumulators are designed to dampen pressure surges to the turbopump assembly and thus reduce the longitudinal vehicle oscillation "POGO" effect between the engine and the airframe.

Propellant tanks must be pressurized to maintain sufficient inlet pressure to the engine pumps for proper pump operation. The tanks are pressurized with gaseous nitrogen prior to engine start. The engine autogenous (self-generating) system supplies pressurized gas to the tanks at a controlled rate to make up for the removal of propellant from the tanks. The fuel tank is pressurized by gas from the turbine inlet, and the oxidizer tank is pressurized by oxidizer that has been heated to a gaseous state by heat exchangers in the turbine exhaust stack.

C2.2.2.2 Stage I Engine (LR87AJ-11) - Titan Stage I propulsion is provided by two government-furnished Aerojet liquid rocket engines, (LR87AJ-11). The individual engines, designated Subassembly 1 and Subassembly 2, are designed to operate simultaneously under a single control system.

A summary of nominal engine performance data is given below:

Altitude Thrust	529,000 lb
Altitude Specific Impulse	301.1 sec
Total Flow Rate	1,727 lb/sec
Oxidizer Flow Rate	1,135 lb/sec
Fuel Flow Rate	592 lb/sec
Mixture Ratio	1.91
Operating Cycle	165 sec
Expansion Ratio	15:1

The engine is hydraulically balanced and requires no thrust controls. It is preset to operate at a certain level (i.e., consume propellant at a fixed rate) by the use of orifices. The steady-state level is determined by balance orifices in the propellant discharge lines and cavitating Venturis in the gas generator bootstrap lines. The propellant flow rate established by the discharge line orifices is a function of both upstream and downstream pressures. The cavitating venturi establishes a propellant flow rate that is sensitive only to up-stream pressure, maintaining a constant flow rate over a wide range of downstream pressures. This controlled propellant flow rate to the gas generator results in a stabilized turbine speed.

The Titan Stage I engine consists of the following major components and subsystems: pump suction (inlet) lines, turbopump assemblies, pump discharge lines, thrust chamber valves, gas generator systems, thrust chambers, autogenous pressurization system, control and instrumentation harnesses, and engine frame.

Suction lines duct the fuel and oxidizer from the propellant tank lines to the turbopump assemblies, each of which is driven by a turbine rated at over 5000 horsepower. The fluid pressure is increased through the pumps by over 1000 psi to force the propellant through the discharge lines to the injector and into the combustion chamber. A portion of this propellant flow is routed to the gas generator to drive the turbines to maintain pump operation. Combustion in the thrust chambers produces gas at pressures over 800 psia and temperatures over 5000°F. This gas is expanded through a convergent-divergent (DeLaval) nozzle and exhausted at supersonic velocity to produce thrust. Thrust vector control (pitch, yaw, and roll) is achieved by pivoting the thrust chambers independently on gimbal bearing mounts. The gimbal action of the thrust chambers is provided by hydraulic actuators operating in response to signals from the launch vehicle flight control system.

An electrical control harness carries signals to start and shut down the engine, and an instrumentation harness carries information from the various engine transducers to the vehicle for transmission of engine performance data to ground stations.

Prior to intended use of the LR87AJ-11 rocket engine, prevalves in the propellant tank lines immediately above the engine interface prevent propellant from entering the engine system. This allows early loading of propellant tanks before launch while still protecting the engine systems from propellant contamination. During the countdown, an arming signal is supplied to the launch vehicle, which signal opens the prevalves and allows fuel and oxidizer to fill the engine.

Opening of the electrically-operated prevalves places the engine in the fill and bleed condition as shown in Figure C2-10. Both fuel and oxidizer fill the engine above the thrust chamber valves because of the static pressure of the propellants in the tanks above the engine. Air entrapped in the oxidizer lines travels through 3/8-inch flex lines up into the oxidizer tank. Air removal from the fuel lines must be complete, as fuel hydraulic pressure actuates the thrust chamber valves at engine start. Smooth operation of the valves depends upon a hydraulically hard system. Therefore, the fuel system is bled full rather than allowing entrapped air to bubble out under gravitational forces as in the case of the oxidizer lines.

The fuel-operated valve actuation system consists of a rod and piston mechanically linked to the Thrust Chamber Valves (TCV) held closed by springs and opened by fuel pressure and acts as a pilot valve to the TCV actuator.

While in the bleed position the Pressure Sequencing Valve (PSV) diverts the fuel into and through the closing side of the TCV actuator, through a 1/4-inch stainless steel vent line to an overboard manifold mounted on the PSV, and out an overboard drain line through a check valve which serves only to protect the PSV from contamination. A bleed orifice located in the drain line and PSV manifold connection controls the bleed rate to approximately 1200 cubic centimeters per minute per subassembly. As long as the engine remains in the fill and bleed condition, fuel is bled overboard in the manner and at the rate described. Minimum bleed duration prior to engine start is 30 seconds.

Start Sequence is shown in Figure C2-11. After completing the bleed cycle, the engine is ready for operation. Titan Stage I ignition occurs approximately 112 seconds after lift-off during SRM burn. The start signal, Fire Switch 1 (FS-1), applies 28 Vdc to the initiator charges of the solid propellant start cartridges mounted on the turbine inlet manifold of each subassembly and initiates separation of the exit closure from the ablative skirt. The start cartridge solid propellant ignites and supplies gas to the turbines, causing them to accelerate. The turbine shaft of each subassembly is connected through a gear train to the fuel and oxidizer pump so that pump operation also begins. Because the thrust chamber valves are closed, no propellant flows, and pump acceleration produces only an increasing pressure in the discharge lines and valve actuation system.

C2.2.3 Titan Stage II

The Titan Stage II airframe structure (Fig. C2-8) consists of the fuel tank, its skirt sections, and the oxidizer tank and its skirt sections.

The fuel tank assembly consists of three major sections: the fuel tank, fuel tank aft skirt, and fuel tank forward skirt. The fuel tank is an all-welded unit with a machined barrel section, a dome bottom with engine mount attachment, a dome top, and an oxidizer passage conduit.

The oxidizer tank is similar in construction to the fuel tank assembly. The tank barrel is approximately 21 inches long and consists of four machined panels. All machining is on the inner surface and has the same diagonal cross-grid pattern as the fuel tank skins. The aft dome assembly is similar to the aft dome of Titan Stage I oxidizer tank. The center cap has a six-inch-diameter outlet. The aft skirt assembly is similar in design to the aft skirt of the Titan Stage I fuel tank. The skirt is approximately 55 inches long and is reinforced by 36 stringers and three ring frames. Attachment points are provided in the interface structure for mounting of four retrorockets.

Three solid propellant retrorockets are located around the aft end exterior of Stage II. They provide the necessary force to retard the forward motion of the Titan Stage II at Transtage separation.

The major components of the electrical system (i.e., Stage II Inadvertent Separation Destruct System (ISDS), static inverter, Redundant Inertial Measurements Unit (RMU), lateral acceleration sensing system) are located on a truss assembly in the Stage II airframe compartment (2B).

C2.2.3.1 Propellant Feed System - The propellant feed system for Titan Stage II is illustrated in Figures C2-12 and C2-13. Both fuel and oxidizer lines terminate at a set of electrically operated, reclosable prevalves. Propellants are loaded into the vehicle tanks through manually operated disconnects above the prevalves.

C2.2.3.2 Stage II Engine (LR91AJ-11) - Except for being somewhat smaller, the Titan Stage II engine is similar in construction and operation to a single subassembly of the Titan Stage I engine. This engine is designed to produce approximately 101,000 pounds of thrust at altitude. Pitch and yaw thrust vector control is achieved by pivoting the thrust chamber on a gimbal bearing mount. Roll control is provided by ducting turbine exhaust through a swiveled roll control nozzle.

A summary of engine performance data is given below:

Vacuum Thrust (Including Roll Nozzle)	101,000 lb
Vacuum Specific Impulse (Including Roll Nozzle)	318.7 sec
Total Flow Rate	316 lb/s
Oxidizer Flow Rate	203 lb/s
Fuel Flow Rate	114 lb/s
Mixture Ratio	1.79
Operating Cycle	225 sec
Expansion Ratio (Nozzle)	49.2:1

As indicated in Figure C2-14 and C2-15, the Titan Stage II engine operates identically to one subassembly of the Titan Stage I engine. Bleed-in of the Titan Stage II and Stage I engines occurs at the same time, because all prevalves in the vehicle are opened simultaneously. The Titan Stage II ignition occurs simultaneously with the Titan Stage I shutdown command. The Stage II engine shutdown signal to the Pressure Sequencing Valve Override (PSVOR) is issued by majority-voted liquid level detectors in the bottom dome of the propellant depleting tank.

C2.2.3.3 Stage II Retro Rocket - The rocket motor consists of the following: a thin-walled steel chamber loaded with a case-bonded solid propellant; an aft bulkhead with graphite insert for the nozzle throat; a steel coupling, which holds the plastic exit cone to the bulkhead; a nozzle closure; a pressure tap; a head cap, which serves to mount the igniter; and a cartridge-loaded Pyrogen-type igniter (Fig. C2-16). Three rocket motors are used. Each motor is mounted in a canister, oriented parallel to the missile centerline and installed so that the nozzle exhausts in the forward direction. The three rockets are fired simultaneously to provide the desired separation velocity.

The Ordnance Item Number and Manufacturer are Rocket Motor, PD40S0001-519 by Thiokol Chemical. The motor is loaded with 4.5 pounds of Thiokol PTE8035 propellant. Its composition is 14% Polybutadyne acrylic acid (PBAA), 70% ammonium perchlorate, and 16% aluminum powder. Storage specification is for normal storage for Class B explosive.

C2.2.4 Airborne Hydraulic Systems

The Stage I, II, and III hydraulic systems provide the engine actuators with hydraulic fluid under pressure in-flight and during Programmable Aerospace Ground Equipment (PACE) ground checkout of the flight control system. Electrical signals from the flight control computer are sent to the servovalves in the actuators. These electrical inputs control the flow of hydraulic fluid to the actuator pistons. The pistons are connected to the engine thrust chambers and gimbal the thrust chambers to change the attitude of the vehicle.

Stage I Hydraulic System - The Stage I hydraulic system is located in Compartment 1C. The system consists of a pressure manifold, a regulating unit (accumulator reservoir), four linear actuators, a turbine-driven pump, and electric-motor-driven pump, a filter, and the necessary hydraulic lines.

During fill, flush, bleed, and checkout operations and during part of the launch sequence, electrical power from the transient bus is supplied to the motor-driven pump (Figur   17). The pump draws fluid from the accumulator reservoir. The pump then pumps the fluid at 3000 psi out of the outlet port through a line to the filter, then to the pressure manifold, and into the Accumulator side of the regulating unit. The fluid return lines from Actuators 2_1, 3_1, and 4_1 return to the reservoir. Fluid returning from actuator 1_1 is routed through the turbine-driven pump to the reservoir. Returning the fluid from Actuator 1_1 through the turbine-driven pump permits bleeding this pump during servicing.

Stage II Hydraulic System - The Stage II hydraulic system components are located in the Stage II engine compartment (2C). This system positions the second stage engine and the roll nozzle in response to command signals from the flight control computer. The Stage II components are similar to Stage I. The system consists of the linear actuators, the regulating unit (accumulator-reservoir), a filter, a turbine-driven pump, an electric-motor-driven pump, and the necessary hydraulic lines.

The electric-motor-driven pump in Stage II is operated during checkout and during fill, flush, and bleed operations. It receives power from the transient bus. The pump draws fluid from the reservoir section of the regulating unit (Fig. C2-18). The hydraulic fluid is then sent, under pressure, through a filter to the main engine actuators and the roll actuator. Fluid is returned from the main engine actuators to the reservoir section of the regulating unit. The roll actuator fluid returns to the regulating unit through the turbine pump. This permits bleeding of the pump during servicing.

C2.2.5 Inertial Guidance System

The Inertial Guidance System (IGS) guides the various stages of the Titan vehicle on a trajectory and controls Stage III velocity so that the payload arrives at the desired location in space traveling at the desired velocity.

To guide the vehicle on this trajectory, the IGS computer sends analog dc steering signals to the Digital Flight Control System (DFCS) actuators and the SRM thrust vector control system. During flight the computer also sends discrete commands to the airborne electrical system to control such things as engine cutoff and engine start signals.

The guidance operation (Fig. C2-19) is accomplished only during powered flight and assumes that the payload will follow a predictable path upon separation from the airframe. In brief, the guidance equipment measures vehicle acceleration from which it computes present velocity and position. Measurements are made from an inertial reference, making unnecessary a ground link for guidance. The computer determines steering signals necessary to attain the proper vehicle attitude and delivers engine cutoff signals when the proper velocity has been reached. Because the orbital portion of flight is predictable, the guidance system need only bring the payload to a point in space which, at the proper attitude and velocity of the vehicle, will result in insertion into the correct orbit.

C2.2.6 Flight Termination System

The Flight Termination System (FTS) provides the Range Safety Officer (RSO) with the capability to shutdown the core engines or shutdown the core engines and destroy the Titan vehicle should it become necessary. The FTS will also automatically destroy stages of the vehicle if they should inadvertently separate.

The FTS consists of two systems: The Command Shutdown and Destruct System (CSDS) and the Inadvertent Separation Destruct System (ISDS). Either system has the capability to destroy that portion of the vehicle to which it is electrically connected. The CSDS has the additional capability to shutdown the core engines without destruction. The CSDS responds to commands transmitted from the ground while the ISDS is activated when certain electrical paths are interrupted as a result of inadvertent separation of stages in flight.

An ISDS is incorporated in each SRM, in Stage I and in Stage II, to automatically destroy these stages should they inadvertently separate from the Space Launch Vehicle. No ISDS is required for the Transtage because the RSO can use the command transmitter to send the destruct signal to the command receivers.

C2.2.7 Ordnance Items

A listing of Titan 34D ordnance items is given:

1. Stage I Destruct System - One destruct safe-arm initiator, PD64S0336-515. Three strands of primacord MMS-N170 Type I, Form B. Six boosters, 60E7-1 (one on each end of primacord strands). Two bidirectional destruct charges, PD60S0135-503.

2. Stage II Destruct System - One destruct safe-arm initiator, PD64S0336-515. Three strands of primacord MMS-N170 Type I, Form B. Six boosters, 60E7-1 (one on each end of primacord strands). Two bidirectional destruct charges, PD60S0135-503.

3. Stage I Engine Start System (Aerojet) - Two engine start cartridges, 380349. Two engine start cartridge initiators, 380162.

4. Stage II Engine Start System (Aerojet) - One engine start cartridge, 384386. One engine start cartridge initiator, 380162.

5. Stage I to Stage II Attachment and Separation System. Twenty-four separation-nut pressure cartridges, PD60S0129-507.

6. Stage II to Transtage Attachment and Separation System. Sixteen separation nut pressure cartridges, PD60S0129-507.

7. Stage II Retro Thrust System - Three retro rockets, PD50S0001-519. Three retro rocket igniters, PD50S0001-503.

8. SRM to Core Forward Attachment and Release System - Eight separation nut pressure cartridges, PD60S0129-507.

9. SRM to Core Aft Attachment and Release System - Four explosive bolts, PD26S0022-011.

10. SRM to Core Translation System - Supplied by UTC.

11. Attitude Control System Valve Actuation - Two ACS Start Valve Cartridges, PD60S0129-507.

C2.3 MISSION SCENARIO

The Titan 34D/IUS is designed to direct/inject spacecraft/payloads into various orbits. The minimum required Titan 34D/IUS payload capability is 4120 pounds delivered to geosynchronous orbit. The Titan 34D minimum class booster performance capability to low-Earth orbit is 32,900 pounds throw weight. For Cape Canaveral Air Force Missions (CCAFS) missions, this represents the combined Inertial Upper Stage (IUS) and payload weight.

The weight delivered to each type of orbit is a function of payload mission and design constraints. Mission constraints include launch window, tracking and data requirements, thermal inputs, and so on; design constraints include structural design loads, fairing jettison time, and guidance system look angle.

Delivery of satellites to geosynchronous orbits requires a circular altitude of 19,320 nmi in the Earth's equatorial plan. The Titan 34D booster must achieve a minimum park orbit of 80x95 nmi in order for the orbit to be sustained until the desired equatorial crossing is reached for placement of the satellite in a preselected hemisphere. A transfer maneuver is then initiated by the IUS, and coast is continued until geosynchronous altitude is achieved. At this time, final plane change and orbit circularization maneuvers are executed.

C2.3.1 Mission and Performance Requirements

Specific Titan 34D/IUS mission and performance requirements are finalized after the payload mission and design constraints have been identified and after a mission flight plan has been generated that satisfies these constraints.

The T34D/IUS is integrated and launched at the Integrated-Transfer-Launch (ITL) Area of Cape Canaveral Air Force Station. Under the ITL concept, different parts of the vehicle are assembled and checked out in different areas and transferred to the point where they are integrated with the rest of the vehicle as illustrated in Figure C2-20.

C2.3.2 Flight Control System Operation

Initial vehicle rollout to launch azimuth is provided by a Flight Programmer and timed open-loop roll torquing program and is fine-tuned by the IUS guidance system. The basic vehicle trajectory shaping is provided by a time-based pitch program generated in the Titan Flight Control System (FCS). Approximately 80 seconds after lift-off and prior to Stage I engine ignition, the IUS Guidance System will provide closed-loop yaw/roll steering. After Stage 0 separation, the IUS guidance system provides closed-loop steering in the pitch, yaw, and roll axes.

The T34D/IUS analog Flight Control System (FCS) provides stable vehicle control by accepting, scaling, mixing, and filtering attitude and rate acceleration sensor inputs and issuing thrust vector commands. In-flight gain and dynamics changes are programmed as a function of stage and flight time in order to satisfy the vehicle stability requirements.

The FCS shall sequence in-flight events required to satisfy the open-loop trajectory shaping requirements, maintain vehicle stability, and command certain flight critical functions, such as staging and engine start. (Fig. C2-21)

At SRM ignition an Aerospace Ground Equipment (AGE) discrete will uncage the three-axis reference system (TARS) gyros which function as displacement gyros providing the flight control vehicle attitude reference. Simultaneously, the AGE will also issue a discrete to the flight control computer to start the first of five SRM TVC injectant fluid dump levels. TVC dump improves performance by augmenting solid thrust and decreasing vehicle weight. The remaining four dump levels are sequenced by Stage 0 gain change discretes.

At liftoff, the flight control computer is in the initial Stage 0 gain and dynamic state. Additional gain and dynamic changes are then sequenced in flight as required to maintain vehicle stability margins.

During the middle portion of Stage 0 flight, the Lateral Acceleration Sensing System (LASS) loop of the FCS is closed to provide structural load relief through the maximum dynamic load (Max Q) region of the atmosphere. After transition through the region, the LASS channel is disabled. Near the end of Stage 0 flight, the FCS enables the ordnance power bus to provide primary power for all subsequent staging functions. The first flight-critical function sequenced by the FCS is Stage I engine start. After a fixed time, the Staging Timer and Flight Programmer B redundantly issue discretes to command Stage 0/I separation.

The Stage I engine shutdown, Stage I/II separation, and Stage II engine start signals are issued simultaneously when either of the Stage I engine thrust chamber pressure switches senses a reduction in chamber pressure.

The physical Stage I/II separation (staging connector disconnect) sends a discrete to the FCS to change the computer gains and pitch rate and to start the second time bases of Flight Programmer A and the Staging Timer. A fixed time after separation, the Flight Programmer and the Staging Timer issue the prime and backup Payload Fairing (PLF) unlatch and separation signals.

When the IUS guidance system determines that the vehicle has achieved predetermined orbital parameters, it issues a discrete which shuts down the Stage II engine and safes the Stage II destruct systems. After a predetermined time delay, the discrete for Stage II retrofire is issued by the IUS guidance system. At this point, the Titan portion of the mission is completed.

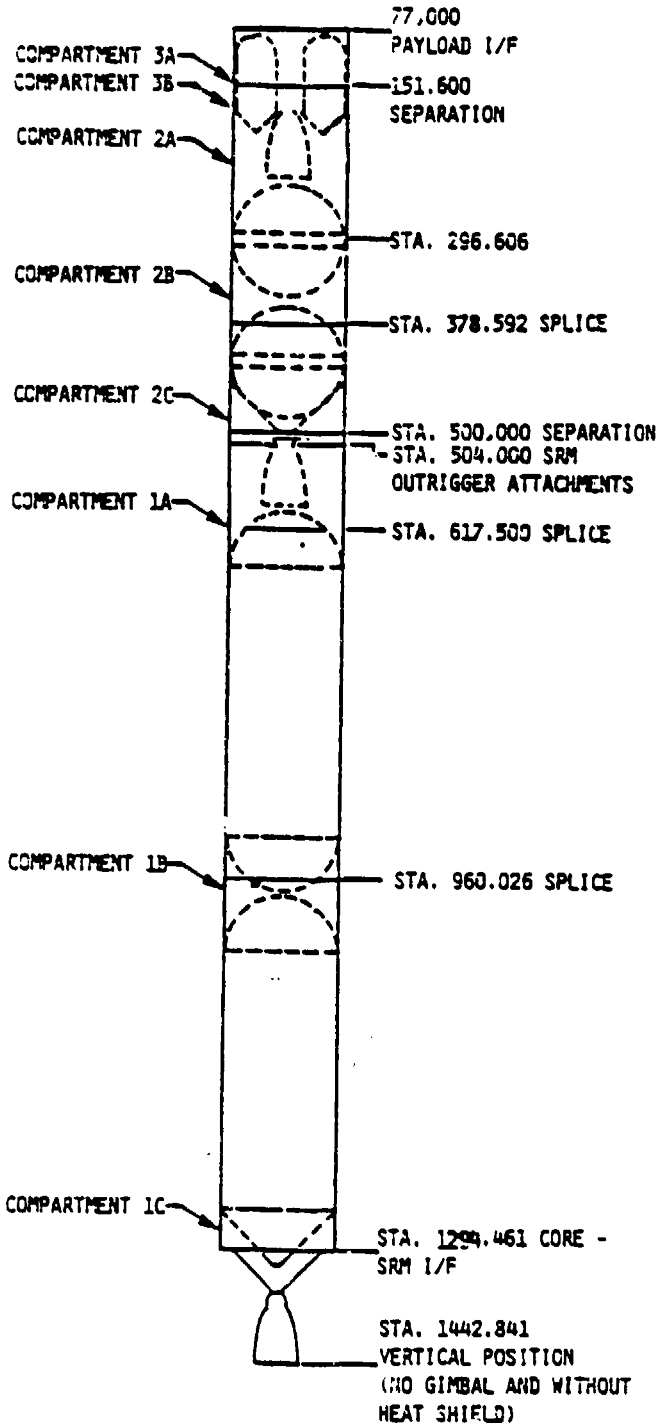


Figure C2-1 T34D Core - Compartment/Station Designation

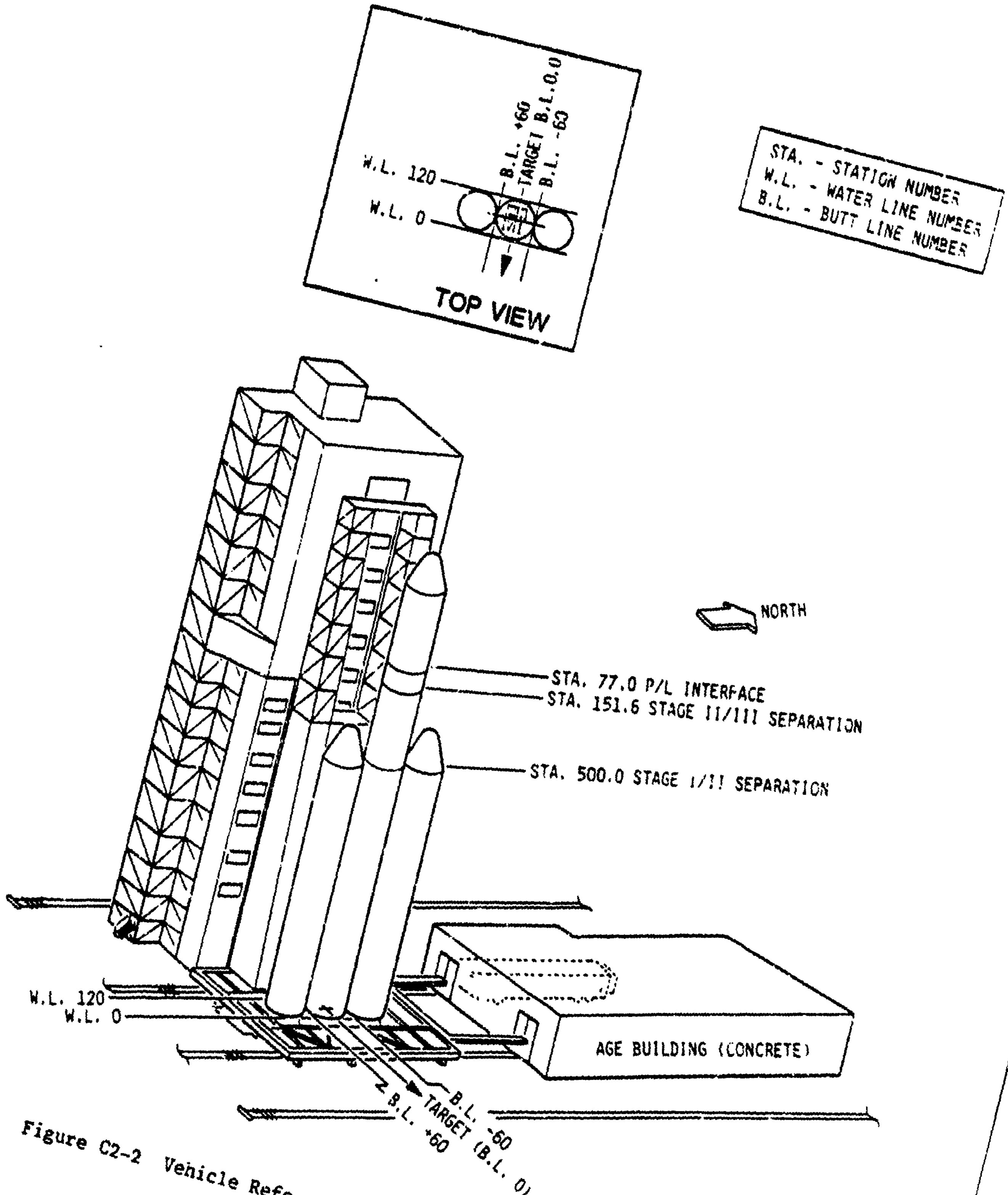


Figure C2-2 Vehicle Reference System on Launch Complex 40

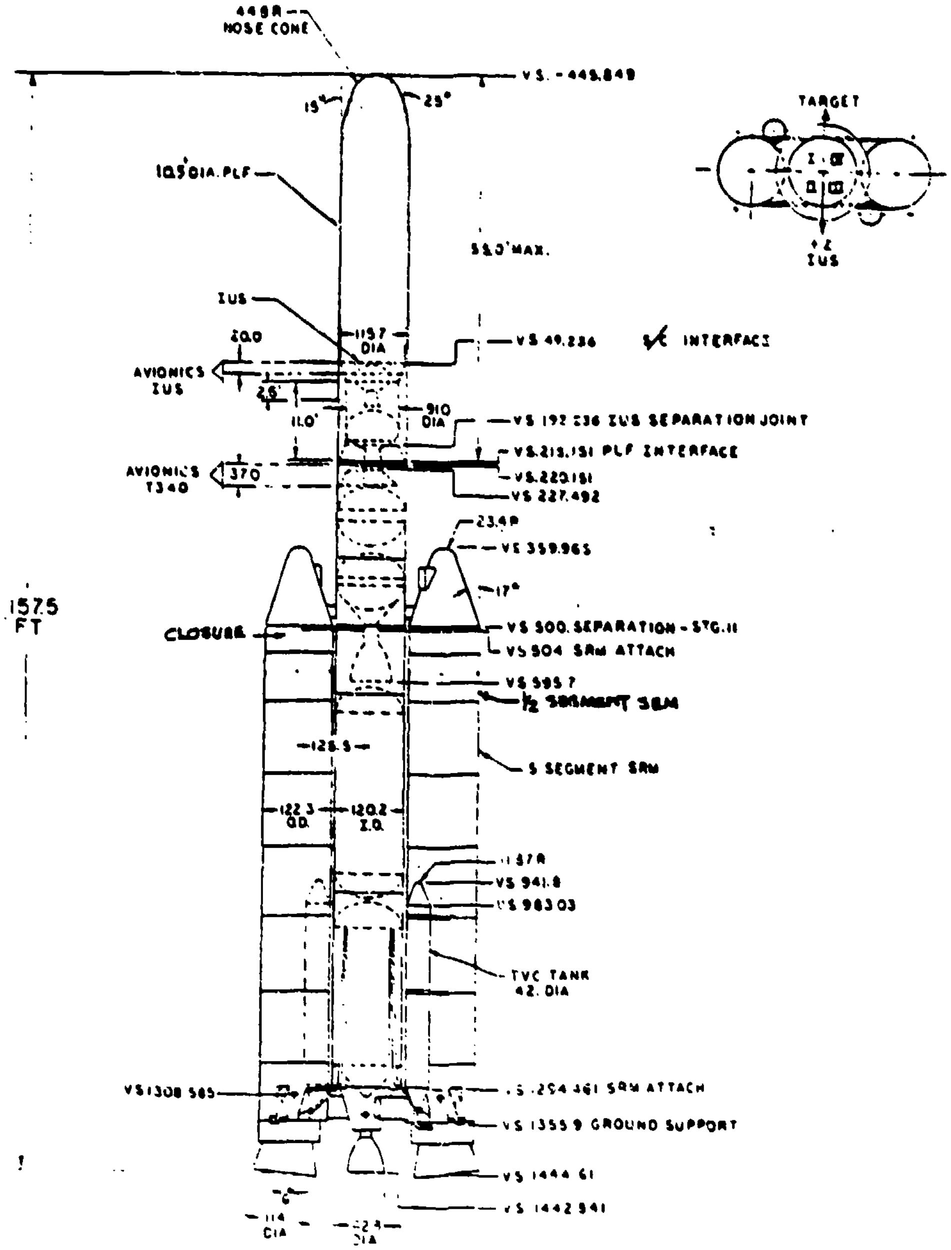
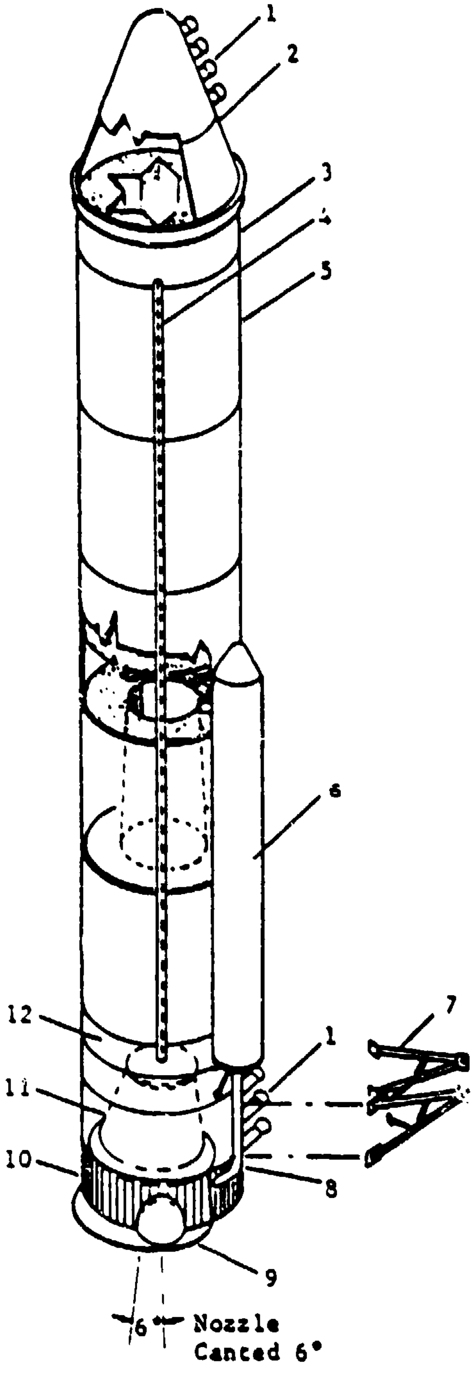


Figure C2-3 Titan 34D/IUS Outboard Profile (CCAFS)



1. Stage Separation Rockets
2. Skirt Fairing
3. Forward Closure
4. External Conduit
5. Segment ($5^1/_2$ Req'd)
6. Thrust Vector Control Gaseous Nitrogen and Nitrogen Tetroxide Tank
7. Stage Zero Support Frame
8. Thrust Vector Control Propellant Feed Line
9. Nozzle Assembly
10. Heat Shield
11. Aft Skirt
12. Aft Closure

Figure C2-4 Solid Rocket Motor Components

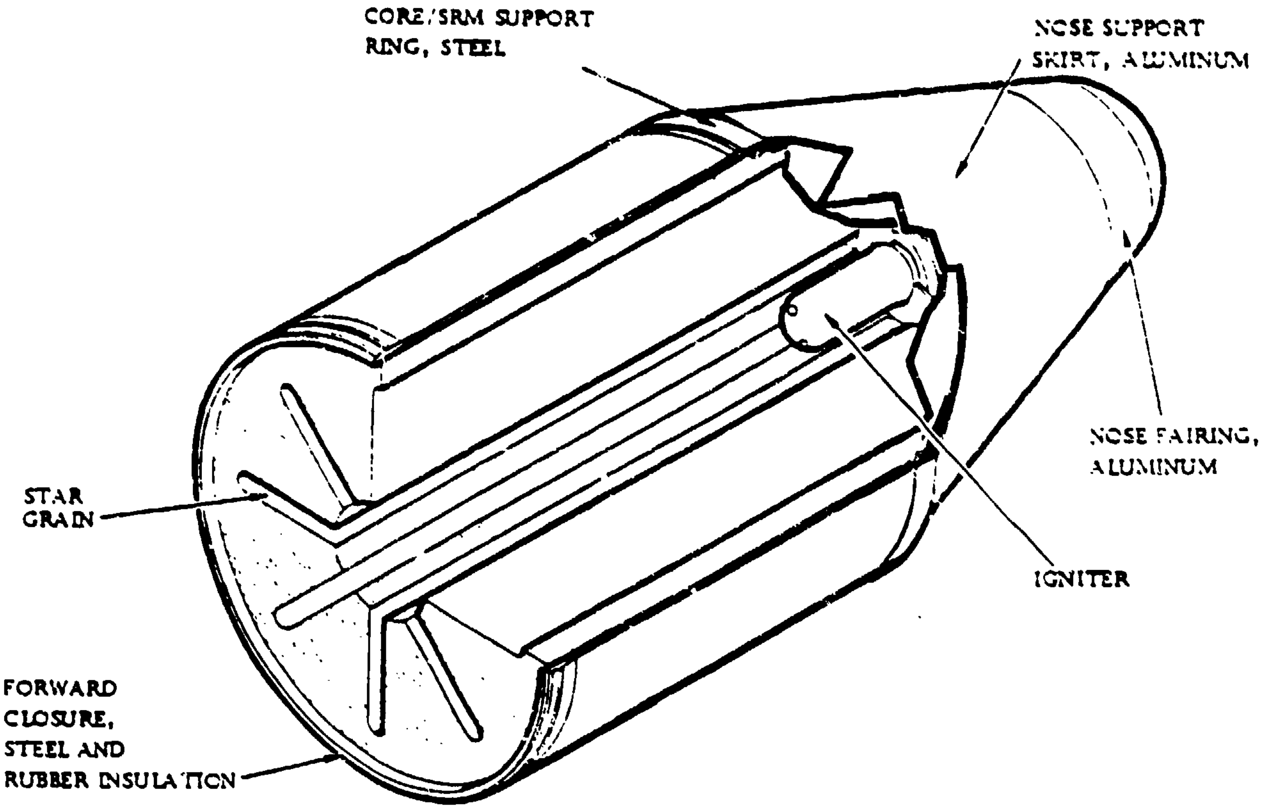


Figure C2-5 SRM Forward Closure

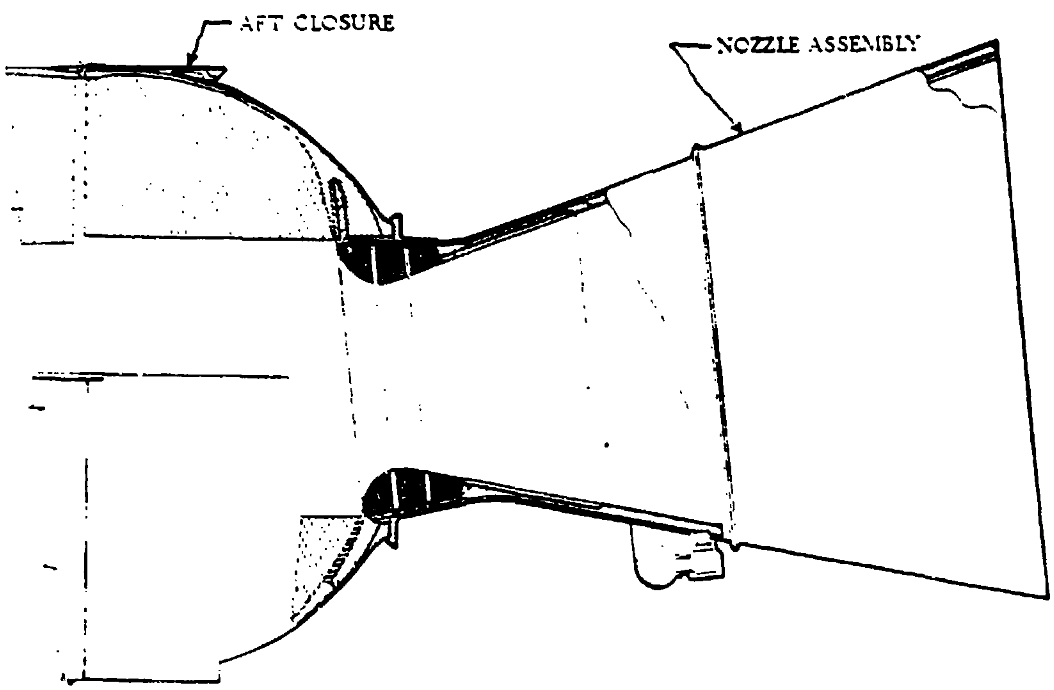


Figure C2-6 SRM Aft Closure Assembly

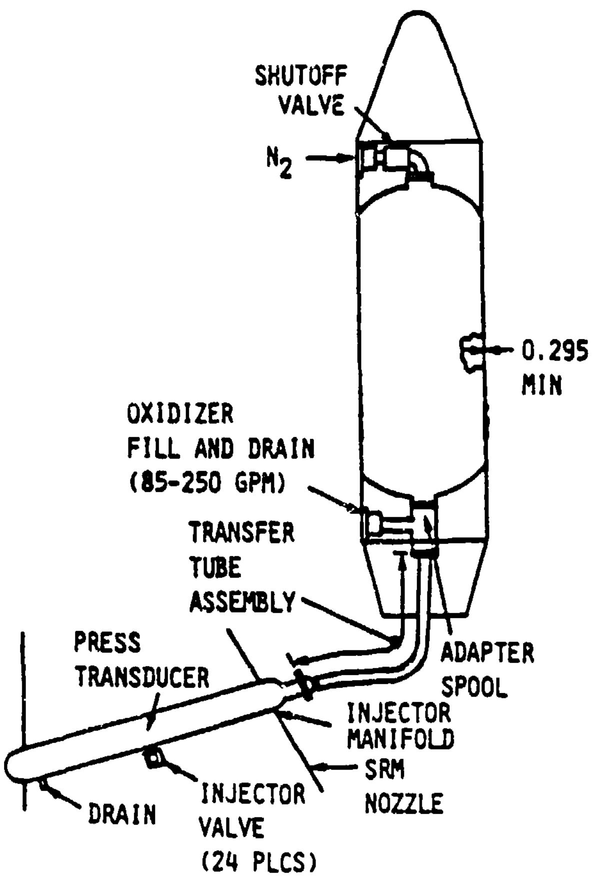


Figure C2-7 Thrust Vector Control Tankage

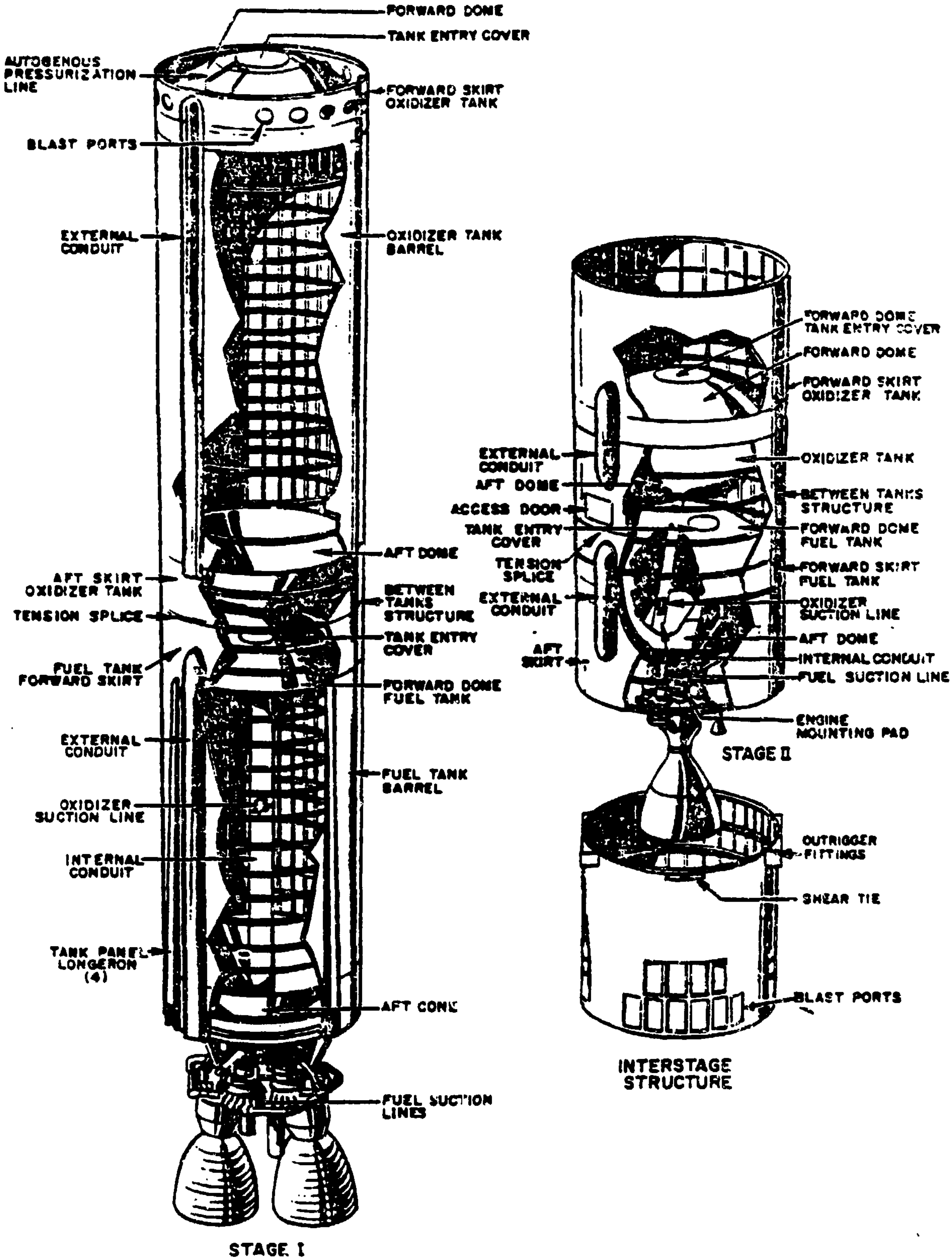


Figure C2-8 Titan 34D Core Airframe Structure

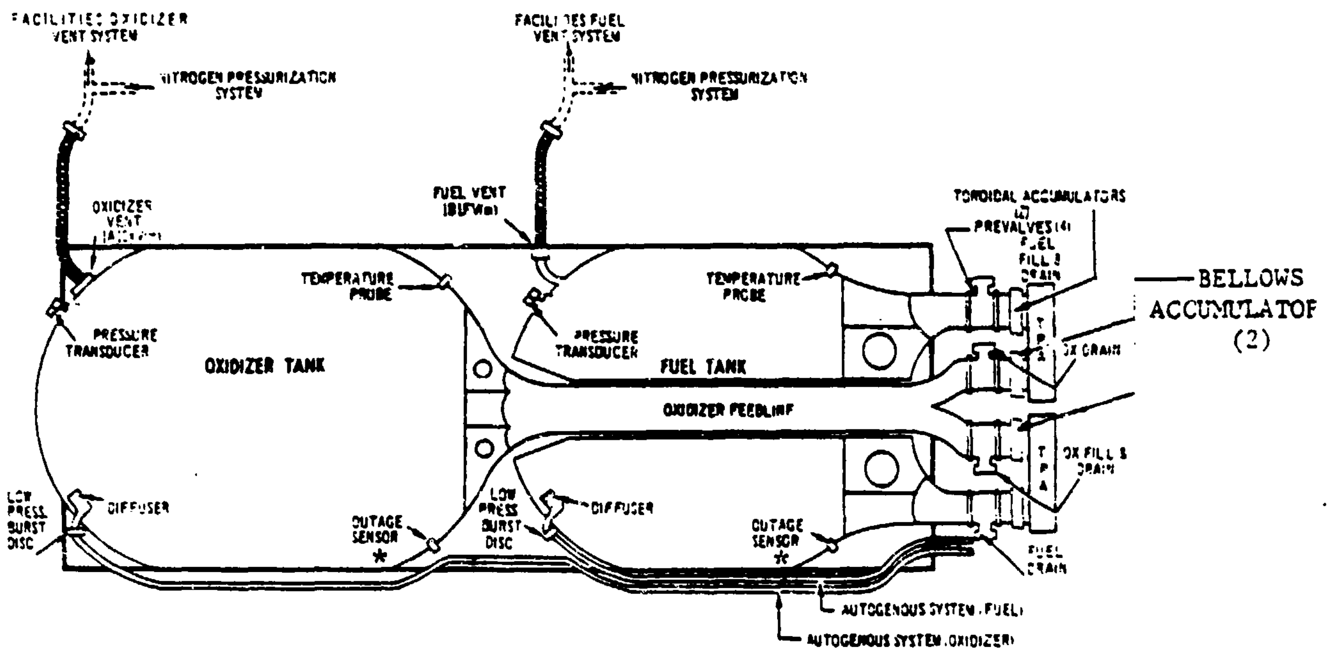


Figure C2-9 Stage I Airframe Attachments

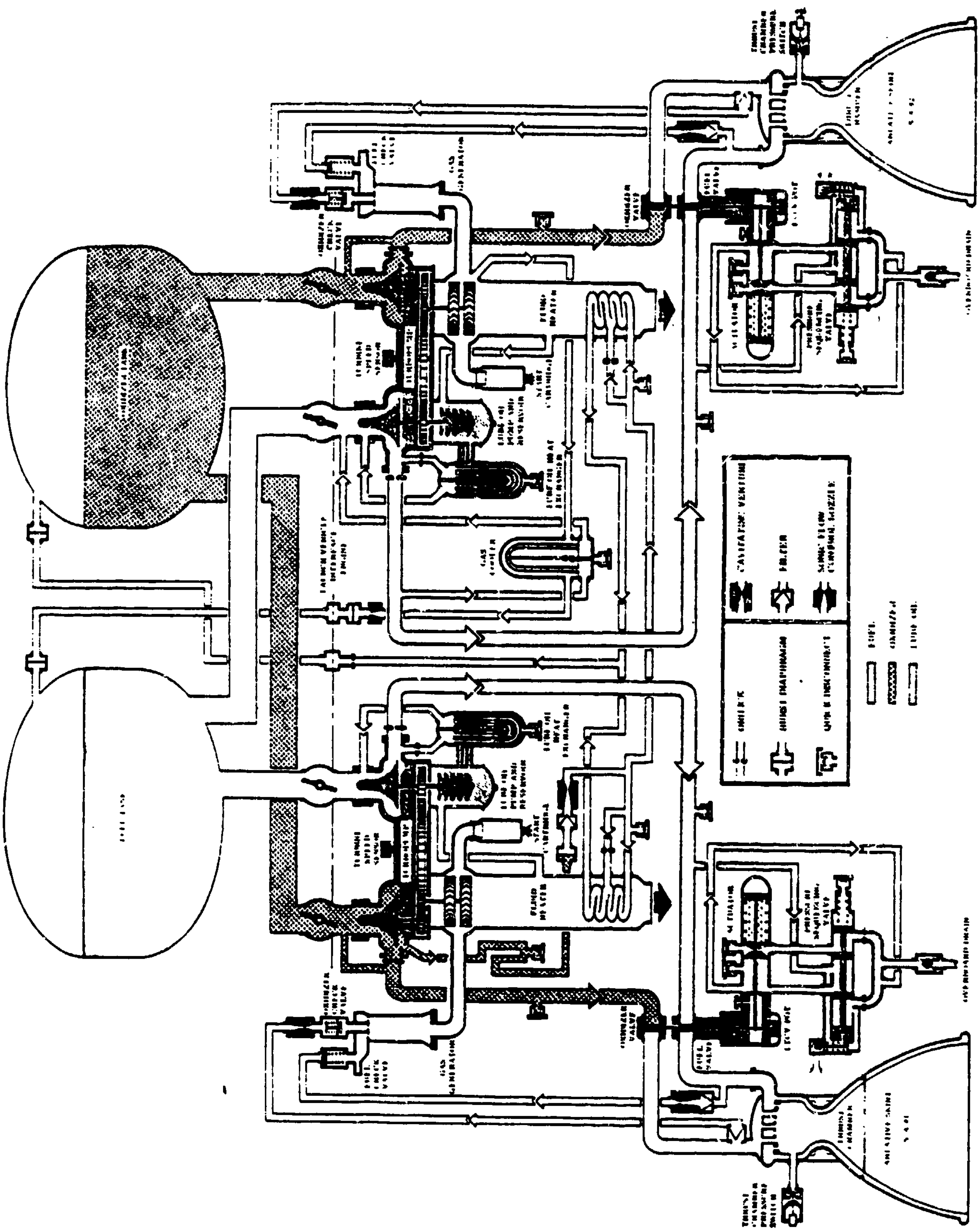


Figure C2-10 LR87AJ-11 Propellant Fill and Bleed Schematic

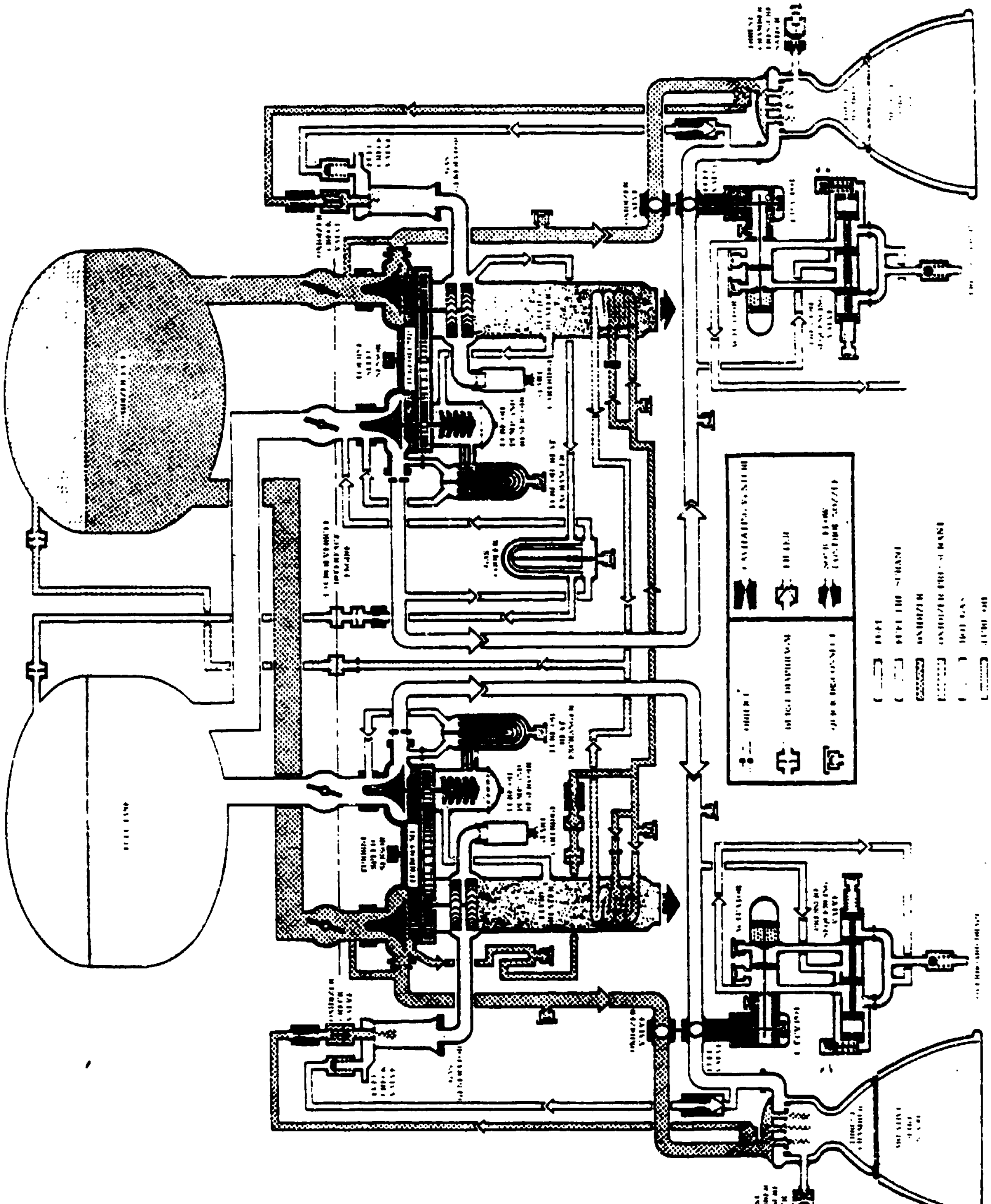


Figure C2-11 LR87AJ-11 Engine Start Sequence

* Plugged ports

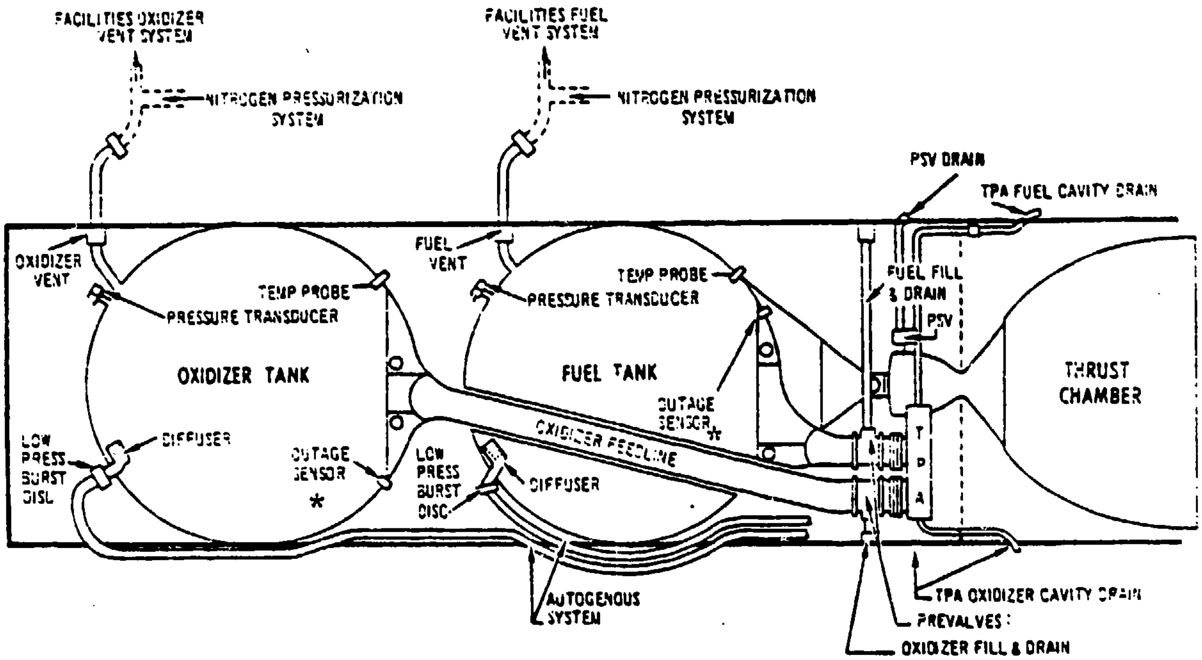


Figure C2-12 Stage II Airframe Attachments

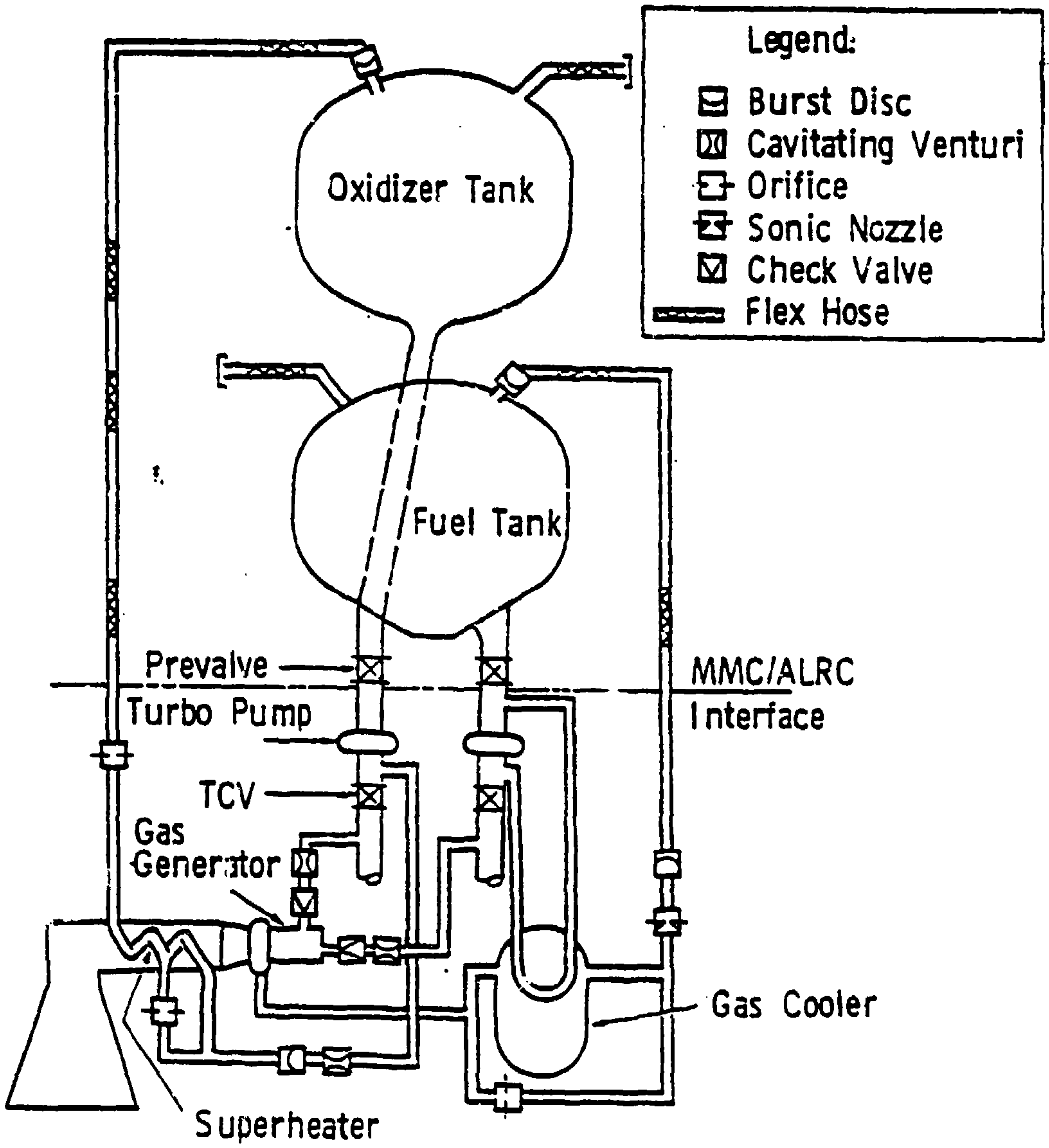


Figure C2-13 T34D Stage II Pressurization and Propellant Feed System Schematic

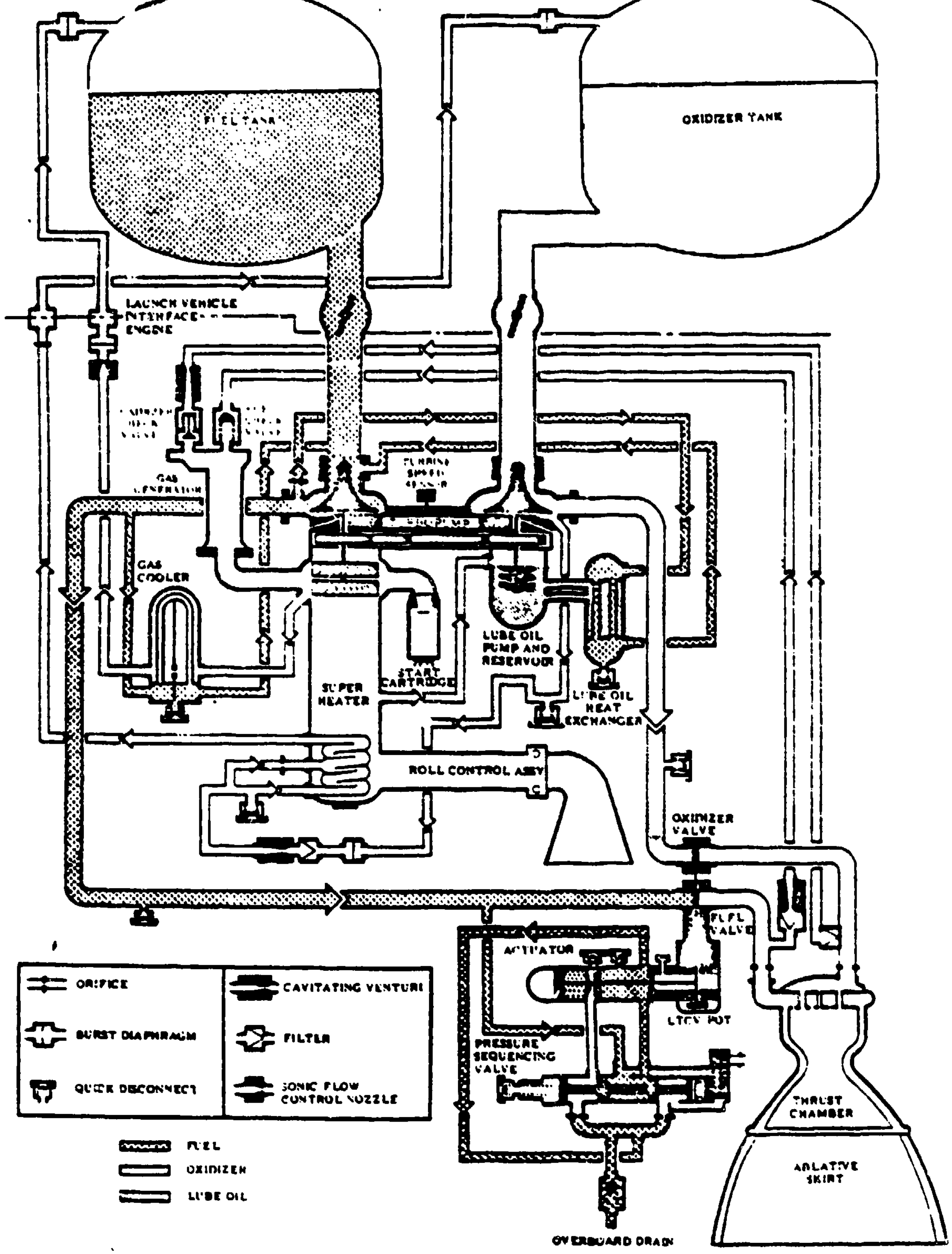


Figure C-14 Stage II Engine Fill and Bleed Sequence

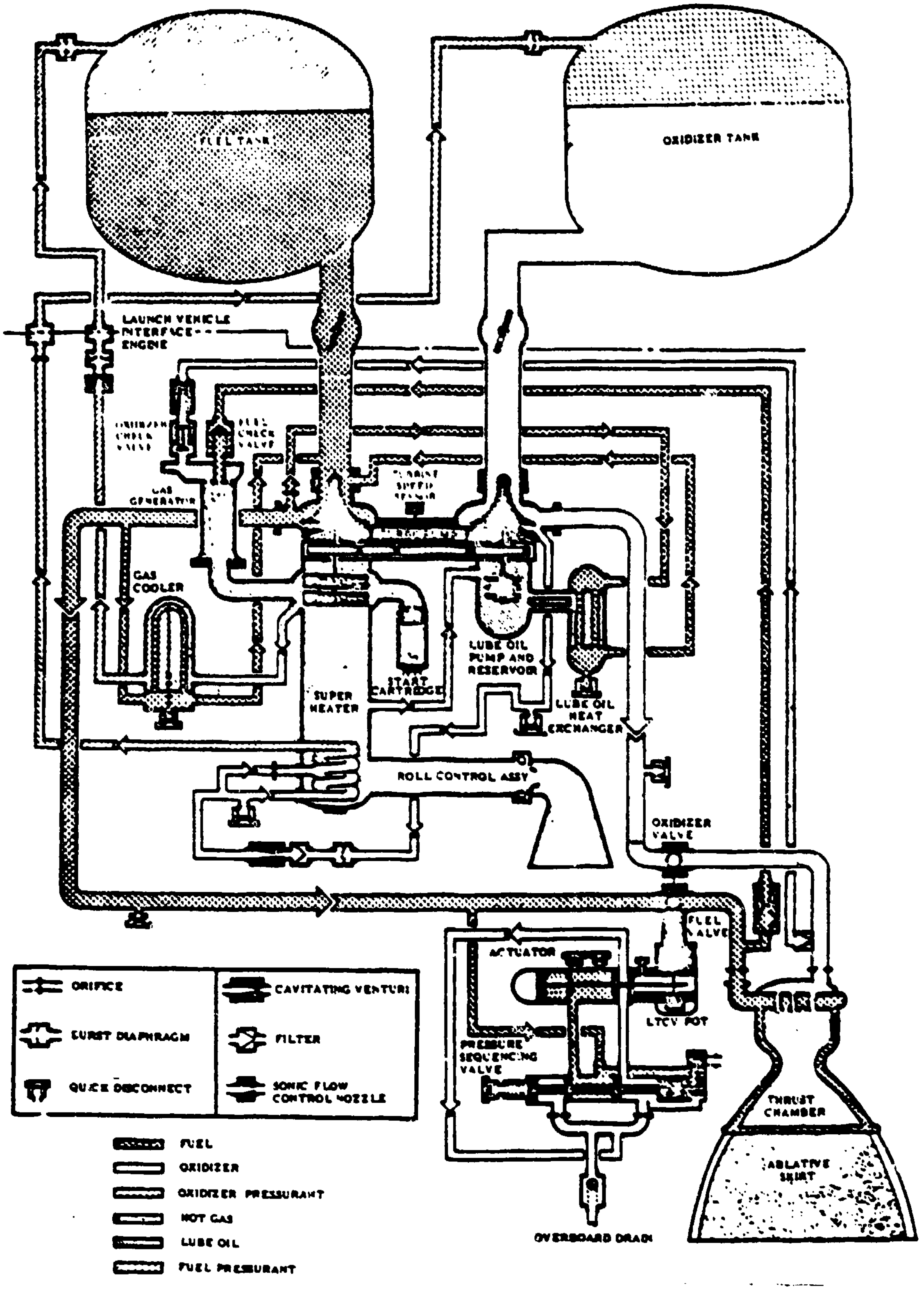


Figure C2-15 Stage II Engine Start Sequence

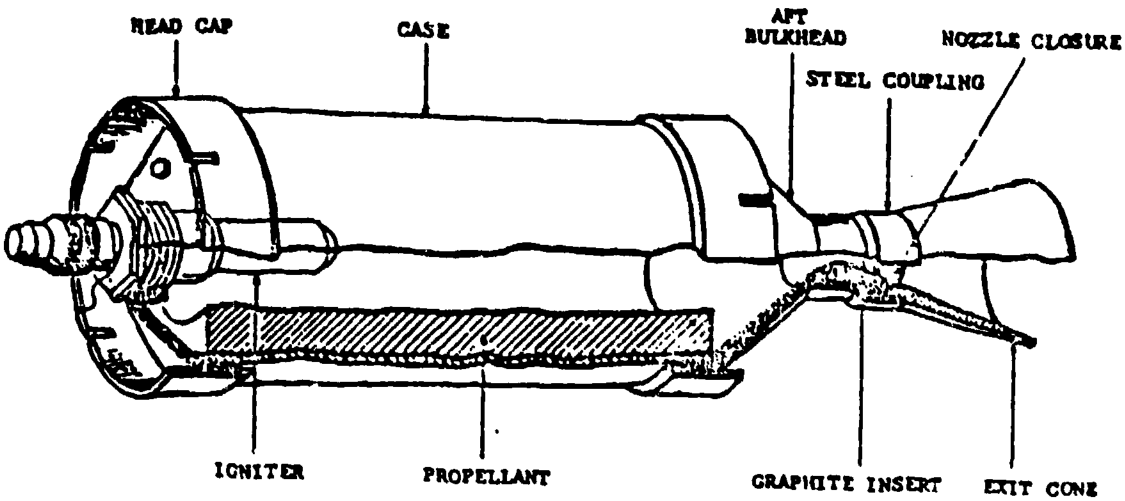
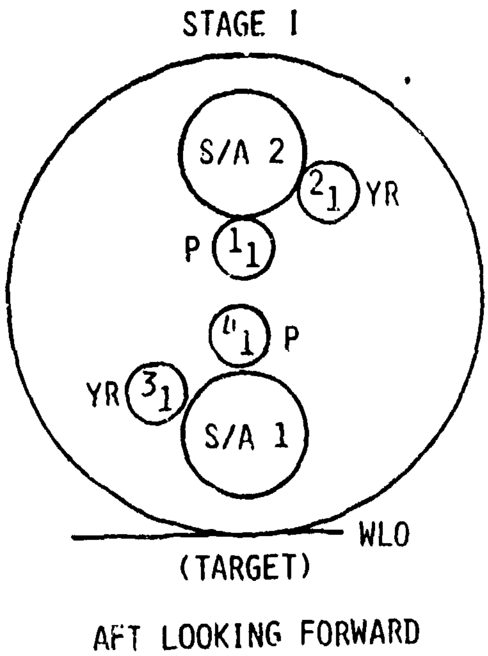


Figure C2-16 Stage II Retro Rocket



SYSTEM PRESS = 3000 PSI
ACTUATOR PISTON AREA = 10 IN^2
ACTUATOR STROKE = ±1.1 IN
TURBINE PUMP CAPACITY = 15 GPM
ACCUM VOLUME = 11 CU IN
RES VOLUME = 100 CU IN
MOTOR PUMP CAPACITY = 3.0 GPM

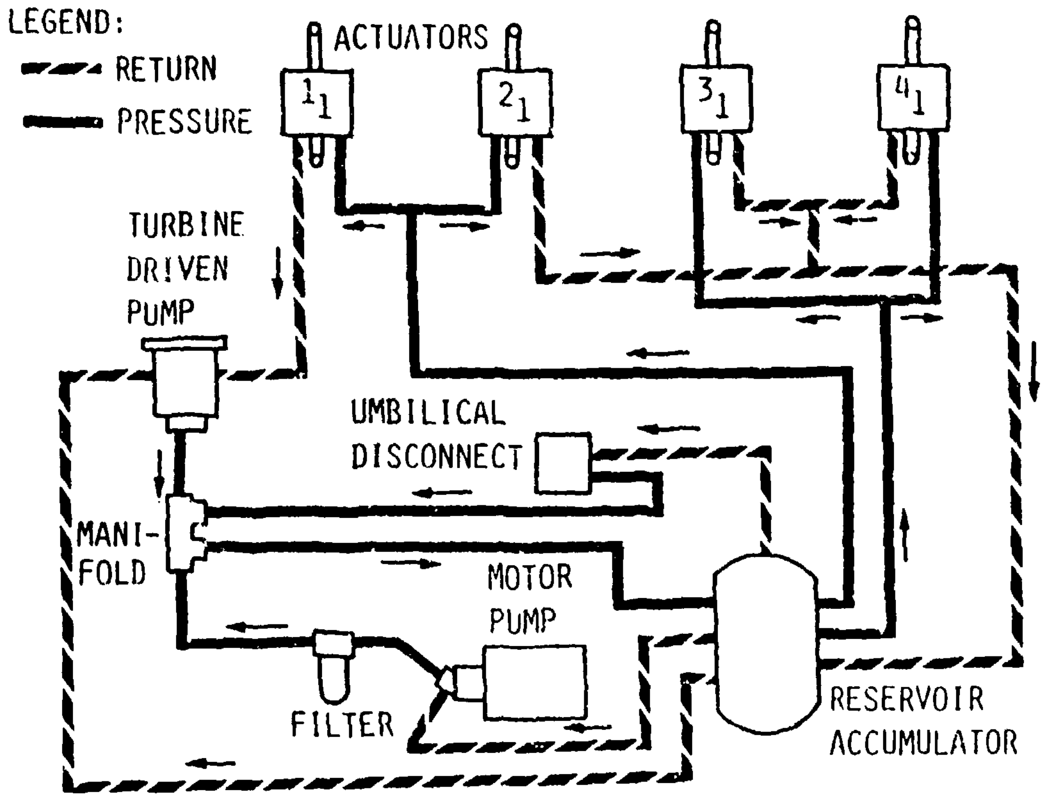
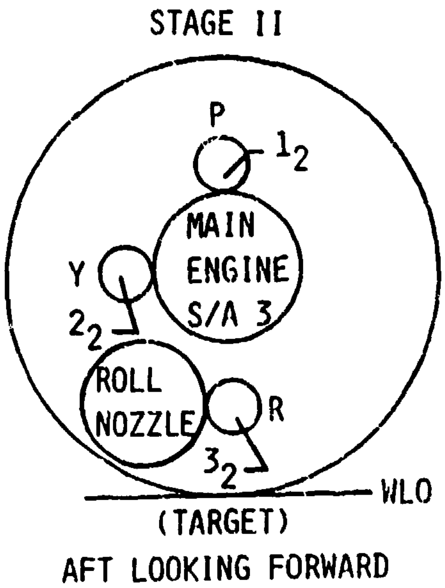


Figure C2-17 Stage I Hydraulic System



SYSTEM PRESSURE = 3000 PSI
ACTUATOR PISTON AREA = 2.55 IN ^{2} (P&Y)
= 0.415 IN ^{2} (R)

ACTUATOR STROKE = $\pm$0.498 IN (P&Y)
= $\pm$1.400 IN (R)

TURBINE PUMP CAPACITY = 5.4 GPM
MOTOR PUMP CAPACITY = 3.0 GPM
ACCUMULATOR VOLUME = 11 CU IN
RESERVOIR VOLUME = 100 CU IN

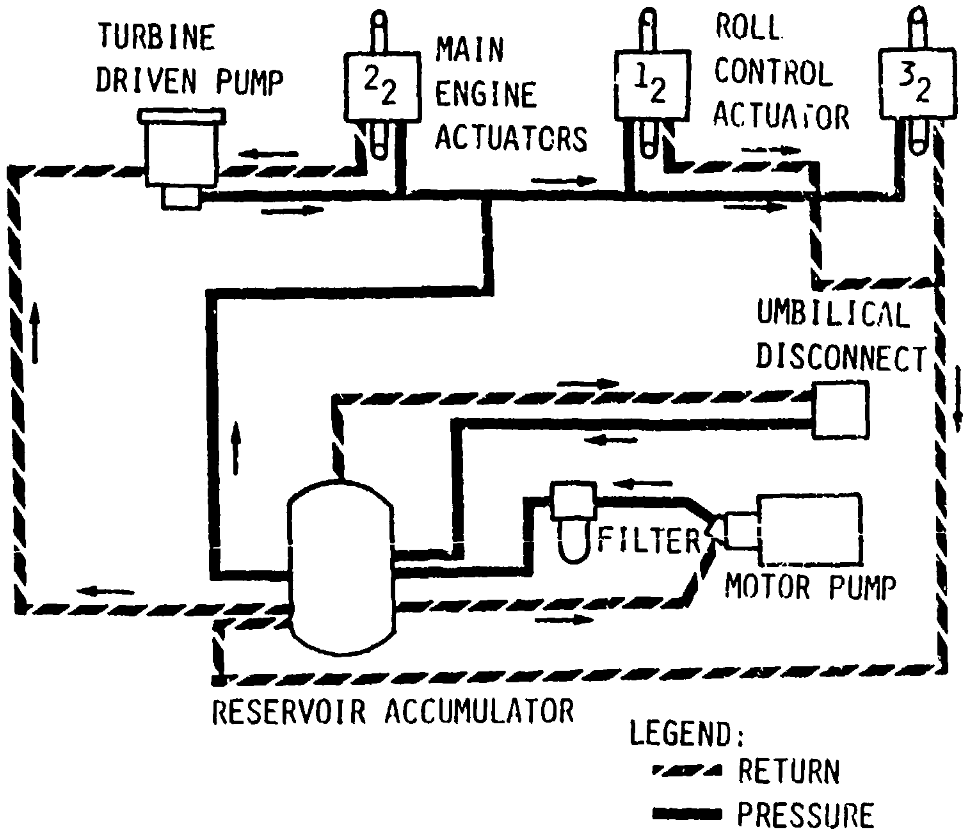


Figure C2-18 Stage II Hydraulic System

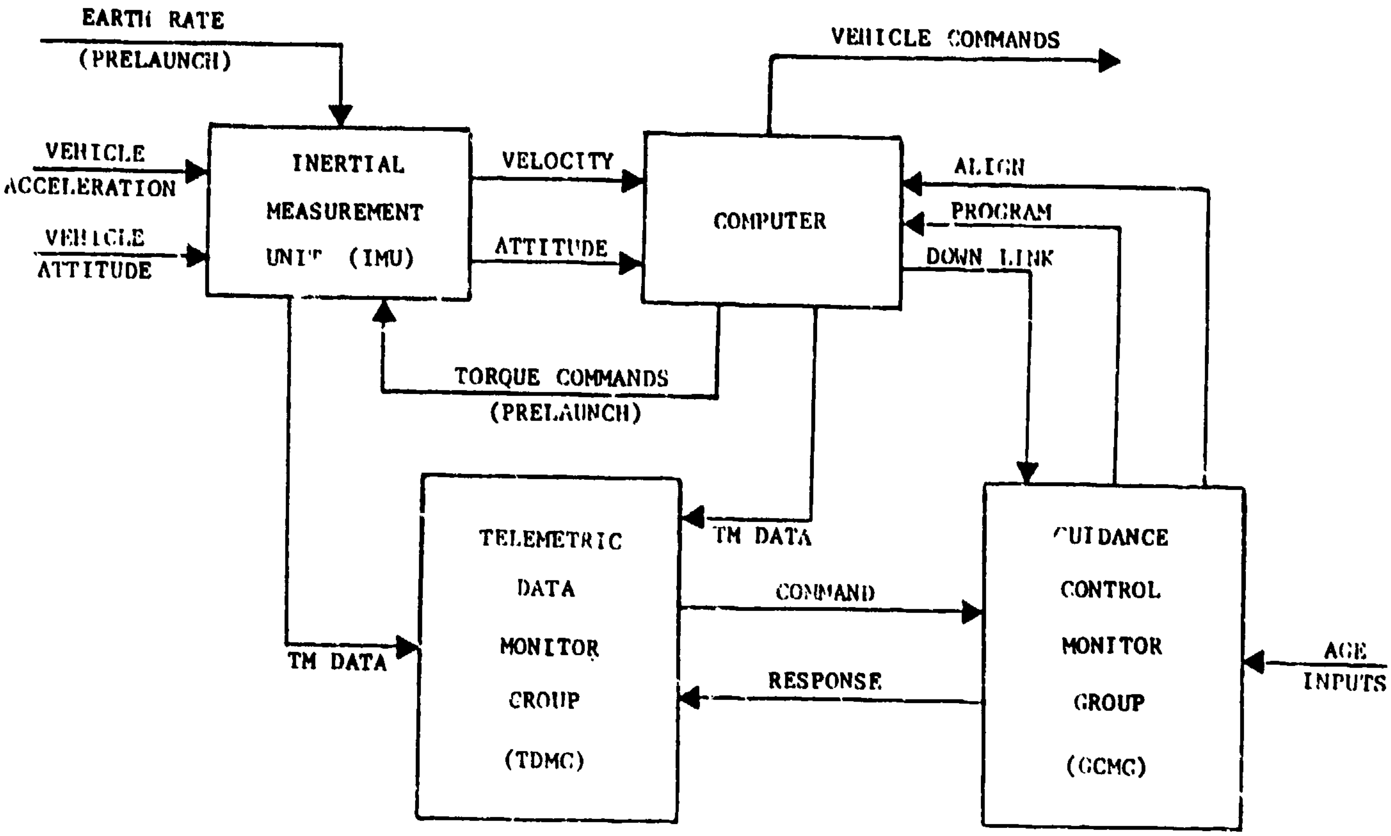


Figure C2-19 T34D/TS Inertial Guidance System

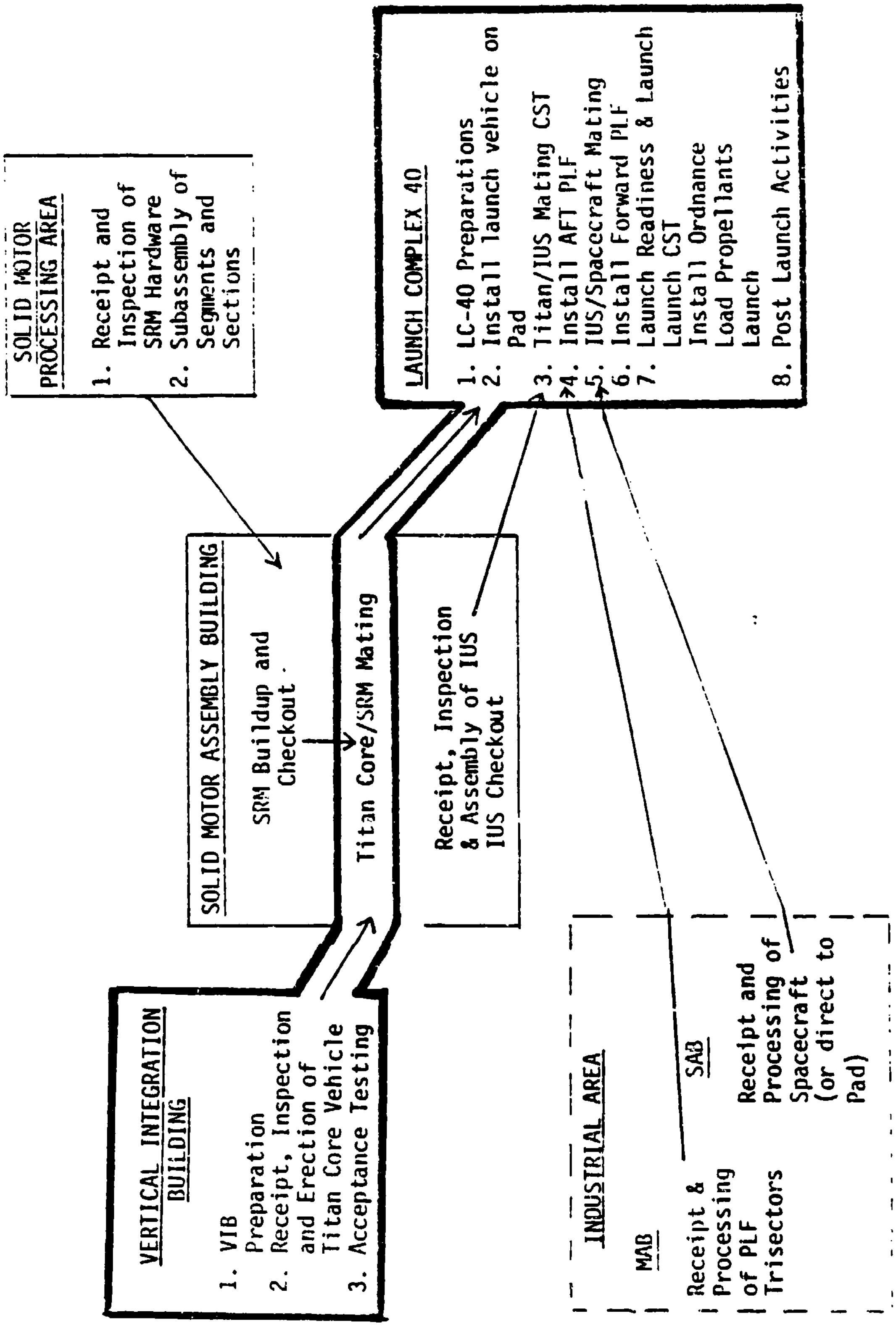


Figure C2-20 T34D/IUS Integrated-Transfer-Launch Processing

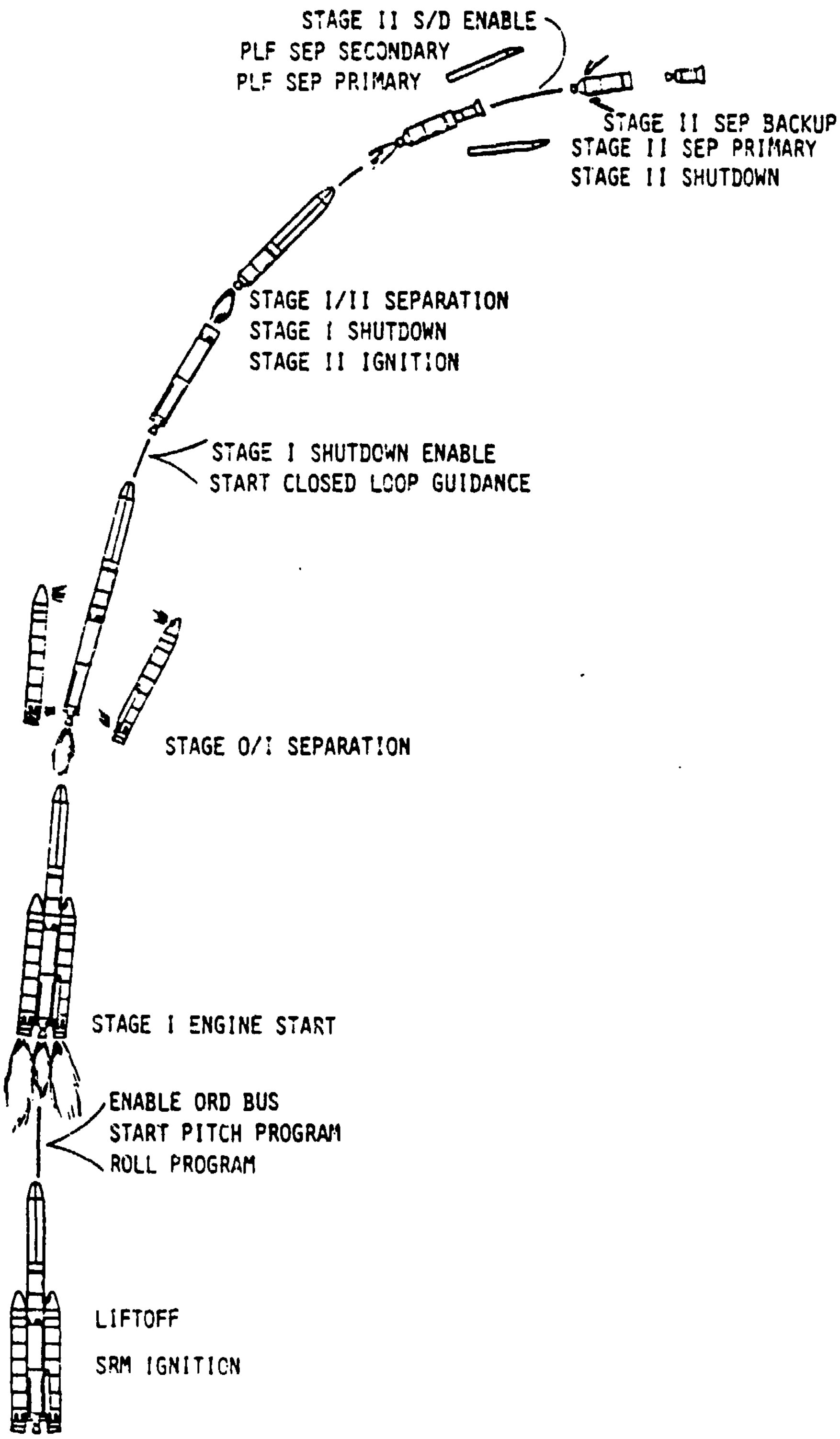


Figure C2-21 Titan 34D/IUS Ascent Sequence Profile

Appendix C3
Atlas F (HGM 16F)

APPENDIX C3
ATLAS F (HGM16F)

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APPENDIX C3
ATLAS F (HGM16F)

C3.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography document, Ref. 244.

C3.1 GENERAL DESCRIPTION

The HGM16F Missile Weapons System is composed of an integrated system of equipment, personnel, and facilities to support, maintain, test, check out, and launch the HGM16F Missile. Supply and maintenance support are provided by Systems Support Manager Prime Air Materiel Areas and by the contractors as required. Additional logistical support is provided by the parent airbase of the facility. An operational strategic missile squadron consists of 12 launch complexes located at varying distances from the parent airbase and one squadron maintenance area located at the parent airbase. Additional launch complexes (one each) are Operational System Test Facility - 2 (OSTF-2), 576-D, and 576-E, located at Vandenberg Air Force Base (VAFB), California. (See Table C3-1)

C3.1.1 Launch Complex

A typical launch complex consists of two underground structures (launch control center and silo). In addition, the OSTF-2 Launch Complex also contains an instrumentation building and utility building (See Fig. C3-1). The structures within the launch complex contain the equipment required to perform tests, checkout, and launch functions on the missile. The launch complex is capable of performing propellant loading exercises (wet countdowns), simulated countdowns, and tactical countdowns. Training launches may be performed, in addition, from the Vandenberg complexes only. In propellant loading exercises, the missile and launch countdown equipment are cycled through the countdown sequence except for starting the rocket engines. The exercise is aborted in the final phase of countdown.

C3.1.2 Missile

The missile is shown in Figure C3-2. It consists of three major sections: a nose section, a tank section, and a booster section. The nose section consists of the re-entry vehicle and adapter and houses the payload of the missile. It also contains circuits and components required to accomplish re-entry vehicle separation, arming, and fuzing.

The tank section consists of a thin-walled stainless-steel monocoque structure which maintains its rigidity by pressurization and is separated into two tanks by an intermediate bulkhead. A sensor still-well assembly of the propellant utilization system is located within each tank. The forward end of the tank section contains provisions for mounting the re-entry vehicle. The aft end contains provisions for mounting components of the hydraulic and propulsion subsystems. Equipment pods are located externally on the tank section to house missile electrical equipment (battery and inverter), flight control system equipment, retarding rockets, guidance system equipment, propellant utilization computer

Table C3-1 HGM16F Missile Weapon System Description

EQUIPMENT	PARTICULARS
HGM16F Missile:	
Length	Approximately 80 feet
Diameter	Tank section: 10 feet, tapering to 70.5 inches (48 inches with re-entry vehicle adapter) Booster section: 10 feet, flaring to 16 feet
Propellant capacities:	
Fuel tank	Approximately 12,000 gallons
Liquid oxygen tank	Approximately 19,000 gallons
Propellants	RP-1 fuel and liquid oxygen
Propulsion system	Five rocket engines: Two booster engines: 330,000 pounds thrust (sea level) One sustainer engine: 57,000 pounds thrust (sea level)
Guidance system	Inertial guidance
Range	Greater than 5500 miles
Launch complex:	Underground hardsite: One silo One launch control center
Silo:	
NO. of crib levels	8
Depth	173.5 feet
Diameter	52 feet
Purpose	Provides facilities for test, checkout, countdown, and launch of the HGM16F Missile
Launch control center:	
NO. of floor levels	2
Height	33 feet
Diameter	44 feet
Purpose	Provides personnel facilities, communications equipment, and equipment for checkout, countdown, and launch control

assembly, and instrumentation equipment. Two decoy pods contain penetration devices. These devices are jettisoned at preset intervals during flight to cause false indications in detection equipment attempting to plot the course of the missile. The forward end of the tank section contains a bulkhead and access plate to allow entry into the oxidizer tank. Also mounted on the bulkhead are the liquid oxygen boiloff valve and the re-entry vehicle adapter.

The booster section contains assemblies of the propulsion, pressurization, and hydraulic subsystems used during the booster stage of missile flight. It also contains rise-off disconnect fittings to connect hydraulic, pneumatic, and propellant lines to the missile from ground supply sources. The booster section is coupled to the tank section by four special staging disconnect couplings. Thrust is transmitted from the booster section to the tank section by a thrust ring welded to the aft end of the tank section. The booster section is jettisoned during the staging portion of missile flight. Staging is accomplished approximately 120 seconds after missile launch.

C3.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C3.2.1 Propulsion System

The missile propulsion system consists of two booster engines, two vernier engines, one sustainer engine, a vernier solo tank system, and associated subsystems. All engines are of the single start type and use RP-1 fuel and liquid oxygen. The combined thrust of the five engines at sea level is approximately 389,000 pounds.

The engines are started during countdown by the launch control system. For starting and progression into mainstage operation, the following sequence of events occurs: The booster and sustainer engines each uses a solid propellant gas generator and are controlled by a fuel pressure ladder sequence. During start, an engine start signal fires the solid propellant gas generator initiators, forcing hot gases produced by the burning of the solid propellant to the turbopump turbines. (See Fig. C3-3). This provides the initial drive to the turbopumps. The flow of hot gases from the solid-propellant gas generators causes the pumps to force liquid propellants from the missile tanks directly to the thrust chambers and liquid propellant gas generators. Electrical signals and fuel pressure-control the various propellant valves. Combustion is initiated by hypergolic combination in all thrust chambers. Pyrotechnic igniters, electrically fired, ignite the propellants as they flow into the liquid propellant gas generators. The got gases produced by the liquid gas generators sustain mainstage operation after starting and until the engines are shut down.

The vernier engines start shortly after the booster and sustainer engines reach mainstage (approximately 3.5 seconds) and operate on propellants supplied from the sustainer engine. Pressurized spherical vernier solo tanks contain and supply the small amounts of fuel and oxidizer necessary for vernier solo operation.

When the rocket engines enter the ignition stage, the ENGINE START indicator on the commit patch will illuminate green. From this point, all sequencing and operation of the booster and sustainer engine components are governed by a fuel pressure ladder sequence and, therefore, are independent of the launch control center.

The ability to shut down the propulsion system with direct circuitry is maintained until umbilical separation at launch. However, after ENGINE START, only a malfunction will shut down the propulsion system; no manual control is possible. When the propulsion system develops sufficient thrust to lift the missile from the launcher platform, a signal is sent to eject umbilicals and illuminate the MISSILE AWAY indicator on the commit patch of the launch control console.

After certain programmed flight conditions are met, signals from the missile guidance system cut off the booster engines and separate the booster section from the missile. The sustainer engine is similarly cut off at a later time, but remains attached to the missile. Vernier engines continue to operate on propellants from the vernier correction phase; the vernier engines are cut off; and the missile is separated from the re-entry vehicle by a signal from the missile guidance system.

Each booster engine is a fixed-thrust engine consisting of a solid-propellant gas generator and liquid-propellant gas generator driven turbopump, and a lightweight, tubular-wall, gimbal-mounted thrust chamber. The turbopump is fastened to a support bracket that incorporates mounting points for attachment to the missile and thrust chamber gimbal. The various control valves, gas generator, and interconnecting ducting and electrical cables are fixed in position to the turbopump and thrust chamber.

The turbopump delivers liquid oxygen and fuel to the thrust chamber and consists of two centrifugal pumps driven by a high-speed turbine through a reduction geartrain. The initial power to turn the turbopump and bring it up to sufficient speed to supply propellants to the thrust chamber and the liquid propellant gas generator is supplied by a solid-propellant gas generator. The high-velocity gas that provides the sustained power for the turbine is supplied by the liquid-propellant gas generator. The gas generator consists of a combuster, where liquid propellants are burned, and the valves that control propellant flow. The propellants are supplied under pressure and are originally ignited in the liquid propellant gas generator by pyrotechnic igniters. The resultant combustion gas is directed to the turbopump turbine impellers. The turbopump gears and bearings are lubricated with oil supplied under pressure from an oil pump. The oil supply tank mounted on the thrust chamber is pressurized with helium to ensure adequate oil flow to the oil pump.

The booster thrust chamber is bell-shaped. Tubes run lengthwise from the top of the thrust chamber to the bottom of the skirt, forming a cylindrical combustion chamber converging into a narrow throat and then diverging to a wide-mouthed expansion chamber. During countdown, inert fluid is injected into the booster thrust chamber by the inert fluid injection system. The inert fluid reduces the pressure surge that occurs when the booster engines are started. Before fuel is fed into the combustion chamber, it passes through the tubes and cools the walls of the chamber. A circular injection plate at the top of the thrust chamber introduces both fuel and liquid oxygen into the combustion chamber. Fuel for ignition passes through an ignition fuel valve and a hypergolic fluid container to six orifices in the injector plate. The propellants are ignited in the combustion chamber by contact of hypergolic fluid and liquid oxidizer. A gimbal mount transmits thrust loads, developed in the thrust chamber, to the missile. The gimbal mount permits both yaw and pitch axis-thrust chamber movement.

The sustainer engine is an integral, gimbal-mounted, fixed-thrust unit installed on the thrust cone of the missile fuel tank. The engine consists of a liquid-propellant gas generator, turbopump, thrust chamber, and control and lubricating components. The sustainer solid-propellant gas generator, liquid-propellant gas generator, and turbopump are similar to the corresponding booster components. The turbopump supplies propellants to the vernier engines and to the sustainer engine during operation. The sustainer thrust chamber is similar to the booster chamber, except that the sustainer combustion chamber is smaller.

The two vernier engines are bi-thrust (two-rated thrust level) units installed on the missile as separate and complete engines. The vernier thrust chambers are double-walled, with the inner and outer walls being separated by a solid spiral coil that forms a passageway for fuel flow. The passageway allows fuel to flow from a fuel inlet manifold at the exit nozzle of the chamber to the injector housing at the head, cooling the inner wall of the chamber. A pneumatically actuated propellant valve controls propellants to the thrust chambers. The thrust chambers are mounted on gimbal shafts that turn in response to hydraulic actuators.

The vernier solo tank subsystem includes the vernier solo oxidizer and fuel tanks and the necessary vernier control components. The subsystem provides propellants for vernier engine solo operation and control of the vernier propellant valves after sustainer cut-off. After the sustainer engine reaches mainstage, the vernier engines are started. The vernier engines then use propellants supplied from the main missile tanks by the sustainer turbopump. During mainstage, the vernier solo tanks are filled with propellants from the sustainer turbopump. The filled solo tanks provide the propellants used by the vernier engines during solo operation.

The missile helium supply furnishes the pneumatic power for actuating the vernier engine control valve and for pressurizing the vernier solo tanks and engine lubricating oil tanks during operation. Missile hydraulic power is used to operate the sustainer propellant-flow-regulating components.

C3.2.2 Propellant Utilization System

When a missile is launched, the propellant tanks contain fuel and liquid oxygen at a specified weight ratio. During missile acceleration, the heavier (more dense) liquid oxygen has a tendency to be supplied to the engines at a faster rate than the less dense RP-1 fuel. This tendency is equalized by the propellant utilization system, which regulates propellant flow in such a manner that both tanks empty evenly.

The propellant utilization system monitors the propellant level ratio and controls the propellant flow ratio to the sustainer engine during flight. (No such control is needed for booster and vernier engines.) To provide indications of propellant consumption, ultrasonic sensors are installed on still-wells at discrete propellant levels (stations) in both the fuel and oxidizer tanks. (See Fig. C3-4). These sensors relay changes in impedance as the propellant levels pass each sensor station. The stations are located so that similarly numbered sensors in both tanks are uncovered at the same time when the flow ratio is within limits.

Time differences in the signals received from a matching pair of sensors as they are uncovered indicate a discrepancy in the volume of flow. The propellant utilization computer assembly receives the impedance changes from the sensors and generates correcting signals, which are compared to servocontrol feedback signals from the propellant utilization valve transducer. A resulting control command signal is then sent to the propellant utilization servocontrol valve, which adjusts the fuel flow rate to produce the proper level in the tank at the next sensor station. To maintain a constant propellant flow, any change in the fuel flow rate causes an inverse change in the liquid oxygen flow rate through activation of the head suppression valve.

C3.2.3 Flight Control System

The flight control system (Fig. C3-5) consists of the autopilot subsystem and the inertial guidance subsystem. It has the function of steering and stabilizing the missile during the powered portion of flight so that when the re-entry vehicle separates from the tank section and enters free fall, it will continue in a ballistic trajectory to intersect the designated target area. The autopilot controls missile attitude during flight by issuing steering commands to the servovalves of the engine thrust chamber hydraulic actuators. The hydraulic actuators, in turn, regulate the angular displacement of the thrust chambers, thereby controlling the direction of forces acting upon the missile.

The combined actions of the flight control system control the four flight phases of the missile. The powered portion of flight consists of the booster, staging, sustainer, and vernier phases. During the first portion of this period, the inertial guidance subsystem supplies the roll voltage to the autopilot subsystem. The missile rotates about its roll axis to establish the yaw axis of the missile in a vertical plane through the desired cutoff trajectory. At the end of the roll maneuver, Vernier Engine 2 is oriented on the side facing the target. A missile pitchover, which lasts until staging, is then programmed to direct the missile toward the target.

The inertial guidance subsystem continuously corrects the missile flight attitude in order to select, incrementally, the optimum trajectory for successful accomplishment of the mission. Targets are preselected while the missile is on the ground by inserting specific target constants boards into the missile guidance computer.

The autopilot subsystem consists of a programmer package, a gyroscope package, and a filter-servoamplifier package. The programmer package contains electronic timing and switching circuits that generate signals to control the programmed functions of missile flight. Assisted in part by discrete commands from the inertial guidance subsystem, the preset switching circuits of the autopilot programmer control the sequencing of the following major inflight operations: missile roll maneuver and pitch program; booster engine cutoff and booster section jettison; sustainer and vernier engine cutoff; re-entry vehicle separation; firing retarding rockets; and destruction of the missile tank section.

There are a total of six displacement and rate gyroscopes in the gyroscope package. Displacement gyroscopes provide signals proportional to the amount of missile angular displacement from the yaw, roll, and pitch reference planes; rate gyroscopes provide signals proportional to the rate (angular velocity) of displacement from the yaw, pitch, and roll reference planes. Pitch, yaw, and roll displacement gyroscopes and the roll rate gyroscope are contained in a canister in the B-2 equipment pod. The pitch and yaw rate gryoscopes are housed in a canister mounted in the cable fairing, forward of the B-2 equipment pod, to compensate for bending modes in the pitch and yaw planes.

Missile attitude deviations from the programmed flight path are sensed by one or more of the rate and displacement gyroscopes. Each displacement gryoscope senses deviations in its particular reference plane (pitch, yaw, or roll). Displacement gyroscope reference planes are controlled by torque signal amplifiers, which are changed for steering purposes. Rate gryoscopes are in fixed reference planes; they sense the rate of missile attitude changes. Gyroscope output, in the form of error voltages, is fed to the filter-servoamplifier package.

The filter-servoamplifier package contains the electronic circuitry to convert gyroscope error signals into input data for the servoamplifiers. These servoamplifiers operate the servovalves of the engine actuators, enabling the hydraulic system to gimbal the engines in the proper direction to counteract the deviations sensed by the gryoscopes. Feedback transducers on the actuators send back signals to the filter-amplifier package that are proportional to engine displacement. Such a feedback signal opposes the gyroscope output signal, causing the engine to return to the null position during an attitude correction sequence.

The airborne inertial guidance subsystem shown in Figure C3-6 consists of a missile guidance control, an inertial guidance sensing platform, and a missile guidance computer. Aerospace ground equipment components, which include the Inertial Guidance Alignment-Countdown Set AN/GJQ-12, maintain the airborne equipment in a state of readiness so that its objectives can be achieved.

C3.2.4 Missile Hydraulic System

The missile hydraulic system furnishes power to the engine thrust chamber actuators to provide pitch, roll, and yaw control of the missile during flight. Hydraulic pressure from the system is also used to operate the sustainer propellant valves and gas generator valves during sustainer engine operation. There are three separate hydraulic subsystems operational during the various flight phases: the booster subsystem, the sustainer-vernier subsystem, and the vernier solo subsystem.

The booster hydraulic subsystem (Fig. C3-5) provides hydraulic pressure to gimbal the two booster engine thrust chambers for pitch, yaw, and roll control from missile launch until staging. During this period the sustainer-vernier hydraulic subsystem provides pressure to gimbal the vernier engines for roll control only. Following booster engine cutoff, the sustainer engine is gimbaled briefly for pitch and yaw control; then it is locked on center to permit jettisoning of the booster section. The sustainer-vernier hydraulic subsystem is again enabled after staging and until sustainer engine cutoff to gimbal the sustainer engine for pitch and yaw control and the verniers for roll control. After sustainer engine cutoff, the vernier solo subsystem supplies pressure to gimbal the vernier engines for the remaining seconds of powered flight.

Figure C3-5 shows the booster hydraulic subsystem only. Operation of the sustainer-vernier and vernier solo subsystems is similar. The booster and sustainer-vernier hydraulic pumps are operated by the booster and sustainer engine turbopumps respectively. Vernier solo hydraulic actuators are operated, after sustainer engine cutoff, by a hydraulic-pneumatic accumulator that has been pressurized with gaseous nitrogen. The hydraulic accumulator and hydraulic pumps supply hydraulic fluid to the servovalves and hydraulic pistons that gimbal the engine thrust chambers. Feedback transducers, physically connected to the hydraulic actuators, sense the amount of thrust chamber movement from the null position and feed back a cancellatior signal to the flight control system. The hydraulic tank serves as a fluid reservoir and is pressurized with gaseous nitrogen prior to flight and helium during flight to supply a positive head, preventing system cavitation. Accumulators, pressurized with gaseous nitrogen, are provided to prevent momentary line pressure drops if system demand exceeds pump capacity and to absorb pressure drops.

There are two hydraulic actuators mounted on each engine thrust chamber for gimbaling purposes. Each hudraulic actuator consists of a hydraulic controller and linear-motor feedback transducer. The booster and vernier engine controllers contain an electro-hydraulic servovalve, hydraulic actuator, and hydraulic flow limiter valve. Sustainer engine hydraulic controllers consists of an electro-hydraulic servovalve and a hydraulic actuator. The hydraulic actuators use hydraulic energy to produce mechanical motion. One end of an actuator is secured to the missile structure, and the other end is attached to a thrust chamber outrigger. The piston rods of booster and sustainer actuator assemblies are connected to lever-arm linkages that transfer piston movement to thrust chamber movement. Vernier actuators consist of two single-ended pistons connected by a rack. The rack turns a pinion gear on the shaft, which rotates the vernier engine thrust chambers.

The hydraulic actuator servovalves are two-stage, electro-hydraulic, four-way transfer valves that control a flow of hydraulic fluid proportional to the amount of net direct current received from the filter-servoamplifier canister of the flight control system. Direction of valve motion is controlled by the direction of armature deflection; direction of armature movement is controlled by the direction of net current flow in the valve coil. Flow limiter valves on the vernier and booster electro-hydraulic servovalves regulate the flow of hydraulic fluid to prevent the actuators from moving too rapidly.

Turning the COMMIT START key switch on the launch control console and then the COMMIT SWITCH on the ALCO COMM/CONTROL panel initiates evacuation of the airborne hydraulic fluid tanks. The reservoir shutoff valves close, and the evacuation chamber shutoff valves are opened. The application of 2000-psi hydraulic pressure forces the evacuation chamber piston to the nitrogen end of the chamber. This action draws approximately 65 cubic inches of fluid from each missile hydraulic fluid tank, providing space for fluid expansion or pressure surges. In normal countdown sequence, the turbopumps that drive the missile hydraulic fluid pumps start a few seconds after evacuation is completed.

C3.2.5 <u>Missile Pneumatic System</u>

The missile pneumatic system (Fig. C3-7) provides inflight pressurization for the fuel and liquid oxygen tanks from the time the pressurization is transferred to internal during the commit sequence of countdown through missile flight. Pressures in the propellant tanks are regulated and maintained at flight pressure by pressure regulators and the missile helium supply. The missile pneumatic system is protected from overpressurization by relief valves.

The missile pneumatic system contains two basic subsystems, the primary supply subsystem and the secondary supply subsystem. The primary supply subsystem consists of six shrouded helium bottles, control valves, and interconnecting tubing. It controls and distributes helium at a regulated pressure to the propellant tanks during the booster stage only and is jettisoned at staging with the booster section. The secondary supply subsystem, which consists of a single airborne ambient helium bottle, valves, and interconnecting tubing, supplies helium to activate the vernier control valves.

At launch, when the missile separates from ground connections, missile lines are sealed by spring-loaded valves in all but two of the missile halves of the connectors. The two liquid-nitrogen connectors have no valves; consequently, these lines remain open allowing liquid nitrogen in the shrouds of the helium bottles to drain. After rise-off, the airborne helium supply furnishes inflight pressurization for the propellant tanks, airborne hydraulic fluid tanks, and the engine lubrication tanks. The helium bottles are cooled by the liquid-nitrogen-filled shrouds prior to launch, enabling a greater volume of the gas to be contained in the bottles. An airborne heat exchanger, located in the No. 2 booster engine turbopump exhaust, heats and expands the chilled helium as it passes through the heat exchanger coil. The helium is then routed through regulators to the propellant tanks for pressurization.

The oxidizer tank pressure sensing line actuates a pressure regulator to furnish helium pressure, as needed, to maintain the oxidizer tank at approximately 26 psi. In similar fashion, fuel tank pressurization is maintained at approximately 62 psi. Helium pressure, also regulated to approximately 60 psi, is supplied to pressurize the hydraulic fluid tanks and engine lubrication tanks.

The shrouded helium bottles, heat exchanger, oxidizer tank pressure regulator, and the fuel tank pressure relief valve and pressure regulator are located in the booster section. When the booster section stages, these components are jettisoned. Jettisoning is accomplished by the release of separation latches, which are actuated by pressure from the ambient helium bottle. The four connectors in the propellant tank lines and the one connector in the ambient helium bottle line contain spring-loaded valves which close at staging, sealing the pressure in the sustainer section systems. Liquid oxygen tank pressure is maintained by boiloff vapor. The fuel tank has sufficient pressure for the remainder of flight. After staging, the sustainer engine lubrication tank and hydraulic tanks continue to receive pressure from the fuel tank pressure line. The vernier pneumatic manifold is pressurized throughout flight by the ambient helium bottle in the sustainer section.

C3.2.6 Missile Electrical System - The missile electrical system (Fig. C3-8) consists of the missile battery, 400-cycle missile inverter (1), power changeover switch, and interconnecting cabling and harnesses. It distributes ac and dc power to the various missile systems during standby, countdown, and flight. During standby and the first part of countdown, the power required by the missile is supplied from ground sources (except for power for the re-entry vehicle, which is supplied by a separate battery located in the nose section). The missile battery is activated, and the 400-cycle inverter is started during countdown. After the power changeover switch transfers power from ground to internal (airborne) during the commit sequence, the missile battery provides dc power, and the batter inverter combination provides ac power to missile systems. Both the battery voltage and the 400-cycle missile inverter output are checked before switching the missile to internal power.

The main missile battery consists of 20 silver-zinc-oxide cells, a battery activation mechanism, and a thermostatically controlled heater. Enclosed in a rectangular canister, the battery is located in the B-2 equipment pod along with the 400-cycle inverter and the power changeover switch. The battery is stored in a dry-charge state. It contains a squib activation circuit that permits remote energizing of the battery cells during countdown. Activation occurs when a signal from the launch control system is applied across the pins of the activation squib, resulting in the production of pressurized gas that ruptures two diaphragms, allowing the electrolyte to flow to the battery cells. The battery can supply 28 volts at 170 amperes for a maximum of 10 minutes.

The 400-cycle missile inverter is contained in a sealed canister and consists of a motor-driven inverter, a magnetic amplifier, a voltage regulator, and a frequency regulator. It converts 28 Vdc from the ground source or the main missile battery to 115-volt, 3-phase, 400-cycle, ac power.

The missile power changeover switch assembly is housed in a sealed canister and consists of a motor-driven switch and associated circuitry, seven receptacles, and a grounding plate. It distributes power to the airborne systems whether the source is from ground or missile generation equipment. The 28-Vdc motor in the unit drives the switching mechanism to close make-before-break dc contacts and break-before-make ac contacts on the switching mechanism, providing smooth transition from ground to missile electrical system power.

Ground power reaches the missile power changeover switch assembly through an umbilical connector. The umbilical plug contains a solenoid-triggered spring-loaded ejector mechanism that ejects the plug from the missile receptacle at rise-off. Should this electrical signal fail, a lanyard triggers the mechanism to eject the plug from the receptacle as the missile rises.

The frequency of the alternating current that operates missile loads must be held to a close tolerance. The two sensors that determine the frequency of the power applied to these loads is a part of the launch control equipment. One is connected to the missile ground power ac source. After the airborne inverter has been started on ground power, the output of the frequency sensor is automatically switched to the launch control indicator. When the inverter output is sensed to be within tolerance, the launch control equipment sends the changeover signal.

The motor-driven power changeover switch connects the missile battery to all dc loads on the missile, with the exception of the re-entry vehicle system, and then disconnects the missile ground power dc source from the loads. It next disconnects the missile ground power ac source and, a few milliseconds later, connects the loads to the inverter. It then sends a signal to the ground, indicating that these sequences have been completed. Main missile loads are powered by the missile battery and by the inverter during the period between changeover and the firing of retarding rockets, following separation of the re-entry vehicle.

C3.2.7 Missile Explosives

The missile is equipped with explosive and hypergolic devices to initiate various functions in a predetermined sequence during countdown, launch, and flight. These functions include activating the main missile battery, starting rocket engines, initiating booster separation, retarding the missile tank section after re-entry vehicle separation, and destroying the missile tank section after re-entry vehicle separation or (VAFB only) destroying the missile if range safety is threatened during a training launch. Figure C3-9 shows the location of missile explosive assemblies.

The main missile battery is energized at the start of countdown by a squib. Activation of the squib results in the production of gas, which upon expansion ruptures the plastic diaphragms containing the battery electrolyte, forcing it under gaseous pressure into the cells. After the electrolyte is forced into the cells, the gas escapes through a small orifice to a vent. The battery cannot be recharged. If the launch is aborted, the battery must be removed from the missile within ten hours following activation.

Four types of devices are used to start missile engines: solid propellant gas generators, solid propellant gas generator initiators, gas generator igniters, and hypergolic igniters. The solid propellant gas generators provide power to start the turbopumps supplying propellants to the booster and sustainer engines. Ignition of the gas generators takes place when an engine start command is sent from the launch control system, firing the solid propellant gas generator initiators that produce hot particles and gases to fire the solid propellant gas generators. The solid propellant gas generators, in turn, produce hot gas that drives the booster and sustainer turbopumps up to the speed and pressure required to feed liquid propellants to the engine thrust chambers and to the liquid propellant gas generators. The liquid propellant gas generators power the engine turbopumps after the solid propellant gas generators have burned out.

Gas generator igniters are pyrotechnic devices that start the burning of propellant mixtures in the combustion chambers of the liquid propellant gas generators. Two igniters are installed in each liquid propellant gas generator. They are fired electrically by signals from the launch control system. Sustainer gas generator igniters are fired simultaneously with the sustainer engine lock-in signal, and the booster gas generator igniters are fired simultaneously with the solid propellant generators that start the booster turbopumps. The pyrotechnic material from the igniter cartridges is already burring when the gas generator propellant valves admit the flow of liquid propellants, causing immediate ignition.

Hypergolic igniters start combustion in the booster, sustainer, and vernier engine thrust chambers. Each igniter consists of a casing containing a slug of hypergolic fuel held in place by burst diaphragms. The hypergolic fuel burns spontaneously upon contact with oxygen. When the turbopumps start, a portion of the fuel being pumped to the engines is sent through ignition fuel valves to the hypergolic igniters. There, fuel pressure ruptures the burst diaphragms, forcing a slug of hypergol into each trust chamber. The hypergol combines with the liquid propellants, which instantly ignite, starting the engines.

Booster section separation at staging is activated by the separation explosive valve assembly, consisting of two explosive valves mounted and wired in parallel. The firing of either valve activates the booster separation mechanism, which consists of four hook-type latches on the skin structure of the booster section that mate with four stirrups on the tank section. At staging, an electrical impulse from the flight control autopilot subsystem programmer fires the separation explosive valves, driving a plunger through a diaphragm, which releases pressurized helium gas for distribution through a manifold to the four latches. All latches open simultaneously, and the booster section separates from the missile body. The booster engines are shut off just before actuation of the separation explosive valves.

A single retarding rocket is located in a fairing forward of each vernier engine, and two retarding rockets are installed in the forward end of the B-2 equipment pod. The thrust chambers of these rockets point toward the front of the missile. The retarding thrust generated by the rocket slows the forward velocity of the missile body, enabling the re-entry vehicle to move away from it without interference. Two seconds after re-entry vehicle separation, the retarding rockets fire upon receipt of a signal from the flight programmer. The missile body is fragmented by a tank fragmentation explosive assembly; the re-entry vehicle continues on its planned trajectory to the target.

The tank fragmentation explosive assembly consists of a metal case, two squibs, two booster charges, the main explosive charge, and associated wiring. A destruct command signal received from the automatic pilot subsystem fires the explosive assembly at a preselected time after re-entry vehicle separation. Detonation of the unit ruptures the intermediate bulkhead between the fuel and oxidizer tank, fragmenting the missile.

C3.3 MISSION SCENARIO

C3.3.1 Launch Vehicle

The HGM16F missile can deliver a thermonuclear warhead or other re-entry vehicle payloads to a target area farther than 5500 miles away. Boosting and guiding of the re-entry vehicle into the desired ballistic trajectory are accomplished by the basic missile with five rocket engines, a flight control system, an inertial guidance subsystem, and associated subsystems.

The missile guidance control is located in the lower portion of the missile guidance pod. It functions as an interconnecting and switching point between the airborne elements of the subsystem and provides amplification and power for the operation of the inertial guidance sensing platform.

The coordinate system is established and maintained prior to missile launch by leveling the platform with the pendulums and aligning the X-axis accelerometer to the target azimuth. The y- and z-axis accelerometers are thus oriented to sense lateral and vertical accelerations respectively, while the x-axis accelerometer senses down-range accelerations.

Stabilization of the sensing platform assembly during standby is needed to prevent the accelerometers from sensing inflight accelerations that are not parallel to the axes of the reference coordinates. Two two-DOF gyroscopes maintain this orientation after missile launch. The gyroscopes are at their null position when the accelerometer axes correspond with the axes of the reference coordinates. Any tendency of the platform to deviate from its original launch point orientation is sensed by the gyroscopes, which dispatch signals to servoamplifiers, which power servo-motors to maintain the stable position.

The missile guidance computer is located in the upper portion of the missile guidance pod. The X, Y, and Z output signals from the accelerometers and the elapsed time furnish the data from which the computer calculates the re-entry vehicle release point. To determine missile position and velocity, the computer processes these output signals together with precomputed target constants and time. Computer output signals (discrete and steering) are sent to the autopilot subsystem to accomplish yaw steering, booster staging, sustainer engine cutoff, and vernier engine cutoff.

During powered flight, from the moment of launch until vernier engine cutoff, the computer continuously calculates missile position and velocity. Position is considered in the three dimensions (X, Y, and Z). From missile position and velocity calculations, the computer determines where the re-entry vehicle would strike the Earth's surface if all engines were shut off and the re-entry vehicle were allowed to enter free fall. The computed impact point is compared to the desired impact point. Continuous calculations of range error functions (REF) and cross-range error functions (CEF) are made during flight. Cross-range error function (CEF) is the error in azimuth which would result if the missile maintained its course. To bring the missile back on course when CEF varies from the programmed flight path, yaw steering signals are generated.

While the engines are operating, the velocity of the missile is constantly increasing, changing the calculated impact point. The calculated impact point thus comes closer to target point, and REF decreases as the missile traverses its trajectory.

Engine cutoff occurs in three stages. When the missile guidance computer determines that missile range velocity (Vx) has attained a specified value, it generates a signal to accomplish booster engine cutoff and staging. The sustainer and vernier engines continue to propel the missile. As the free fall point (re-entry vehicle separation) comes closer, and at a specified value of range error function, the computer generates a signal for sustainer engine cutoff. Missile velocity increases very slowly after sustainer engine cutoff because only the vernier engines are still operating. When the calculated range error function approaches a predetermined point, the computer generates a vernier engine cutoff signal. The re-entry vehicle then separates and enters a free-fall trajectory to the target.

C3.3.2 Re-Entry Vehicle - The Mark IV re-entry vehicle houses the missile payload, protects it during re-entry into the atmosphere, fuzes and arms the warhead, and transports the payload to the target area during the final portion of flight. It is of the ablative type, structurally consisting of a noze section, center section, and aft section. Functional systems include a separation system and an arming-fuzing system. Prearming of the warhead occurs during flight upon receipt of a signal from the flight control system when the missile has reached a predetermined position. When the ideal release point is reached, the re-entry vehicle separates from the missile tank section for its final plunge to the target.

After separation, the re-entry vehicle is rotated on its longitudinal axis at a slow spin rate for stability. Retarding rockets in the missile tank section are fired to decelerate the airframe to prevent the possibility of interference with the planned trajectory of the re-entry vehicle. The warhead (if installed) is detonated above the target (air burst) or on the target (ground burst) as determined by preselection. The ground burst capability also serves as a backup in the event of an airburst selection failure.

C3.3.3 Missile Flight Safety System (Vandenberg Air Force Base (VAFB) Only)

A missile flight safety system is provided in training launch missile for intentionally destroying the missile should the missile have an erratic flight which may endanger populated areas. Transducers aboard the missile monitor critical pressures, temperatures, velocities, and attitudes. The output signals of the transducers are adapted for transmission to ground control stations for evaluation. Beacon tracking signals and electronic (radar) surveillance data are used by the range safety officer in reaching a missile destruct decision. Should the range safety officer decide to destroy the missile, a destruct push-button on the range safety officer console is depressed, which causes the command destruct transmitter to send a coded signal to the missile, igniting primers of an explosive charge mounted near the fuel and oxidizer tank bulkhead. Detonation of this explosive charge destroys the missile.

C3.3.4 Countdown Exercises - Six different and separate launch exercises may be performed with the weapon system.

1. Tactical Launch Countdown. A tactical launch countdown shall be initiated at an alert status complex upon receipt of a valid and authenticated launch order. The complex is prepared for alert status using appropriate maintenance flow diagrams and is maintained on alert through daily inspection, servicing, and repair.

2. Training Launch Countdown. A training launch countdown may be initiated only after appropriate preparation. These exercises are performed to check functional integrity of the launch complex, missile, and combat crew personnel. Preparation for countdown and mission are different than that of a tactical launch; however, the countdown sequence is identical.

3. Propellant Loading Exercise Countdown (PLX). A propellant loading exercise countdown may be initiated only after appropriate preparation. These exercises are performed to check the (a) functional integrity of a launch complex and missile on EWO alert status, (b) evaluation of combat-ready crews in performance of countdown procedures, and (c) training of combat crew personnel. Countdown sequence to the point of launch is identical to that of a tactical countdown.

4. Maintenance Countdown. A maintenance countdown may be initiated only after appropriate preparation. These exercises are performed in support of maintenance functions.

5. Launch Signal Responder Countdown (LSR). A launch signal responder countdown may be initiated only after appropriate preparations. These exercises are performed in support of maintenance and evaluation/training of missile combat crew personnel. Countdown sequence as it appears in the launch control center is identical to a tactical or PLX countdown.

6. Simulated Countdown (Tank Through). A simulated countdown may be initiated at any time with no special preparation, as no systems are activated. These exercises are performed to train and maintain countdown proficiency of the missile combat crew members.

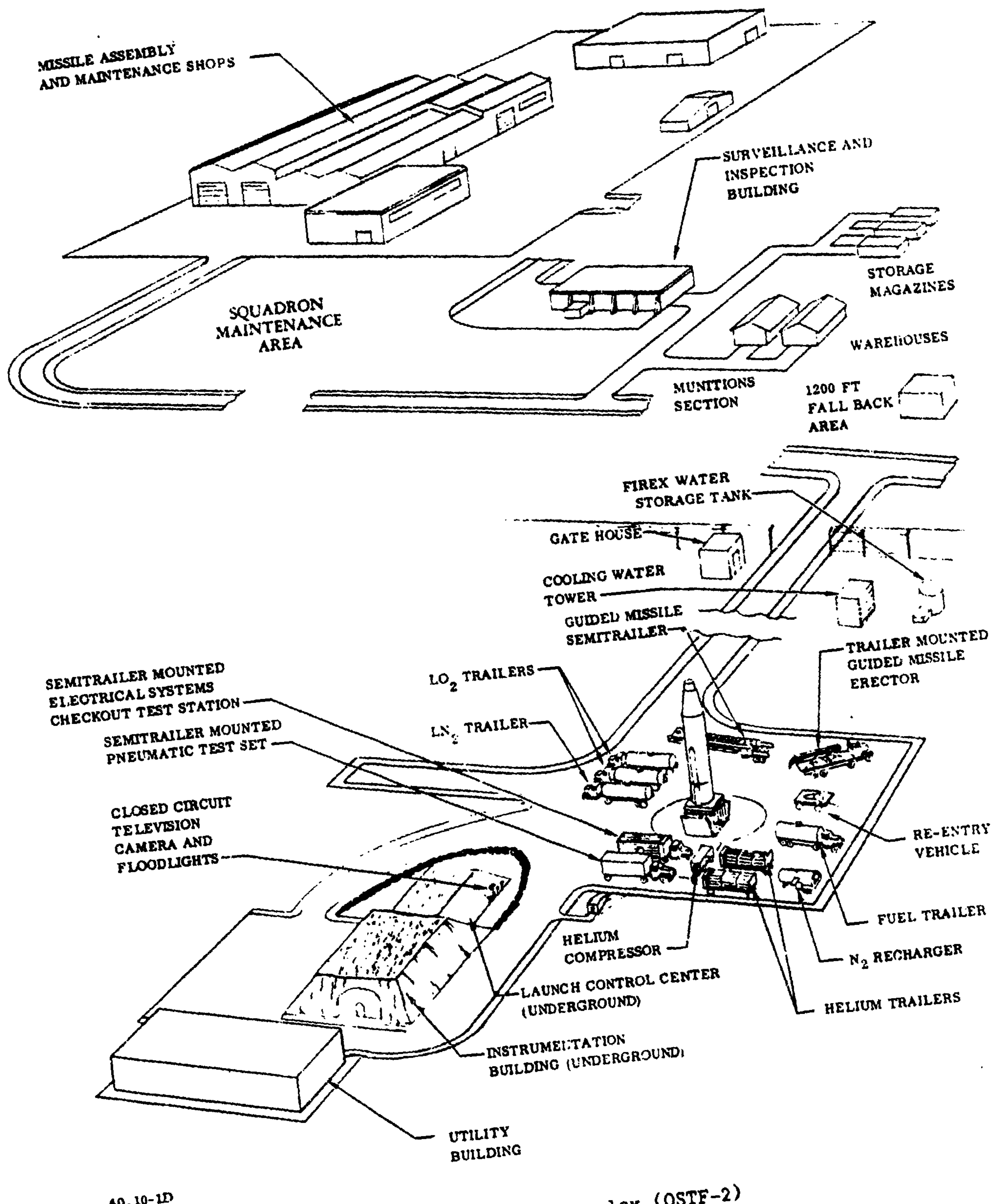
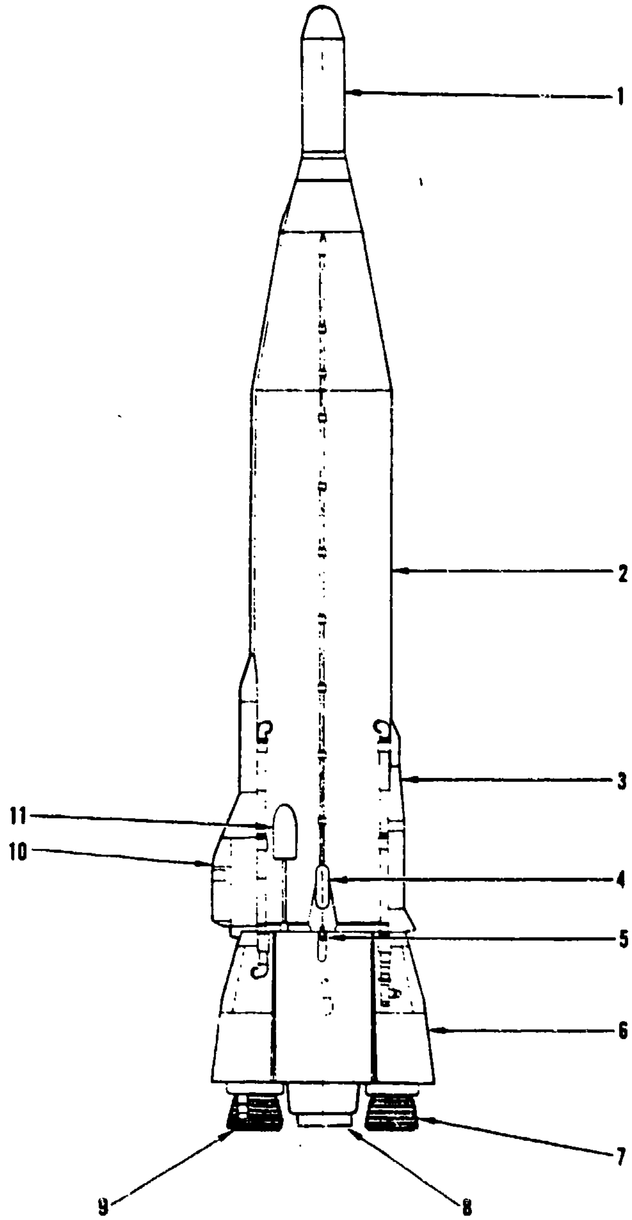


Figure C3-1 Launch Complex (OSTF-2)



1 RE-ENTRY VEHICLE
2 TANK SECTION
3 EQUIPMENT POD
4 VERNIER RETARDING ROCKET
5 VERNIER ENGINE
6 BOOSTER SECTION
7 BOOSTER ENGINE
8 SUSTAINER ENGINE
9 BOOSTER ENGINE
10 EQUIPMENT POD
11 DECOY POD

Figure C3-2 HGM16F Strategic Missile

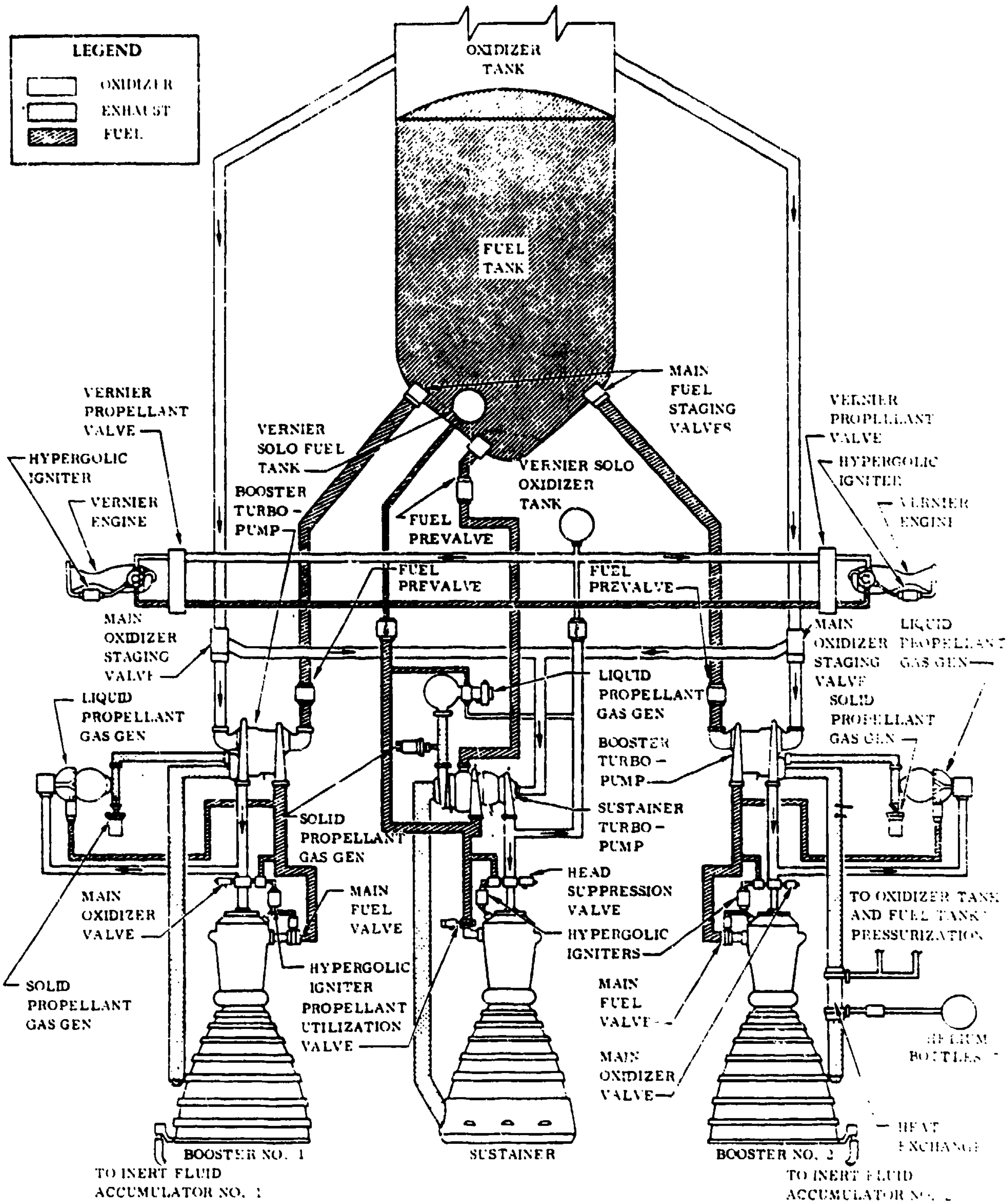
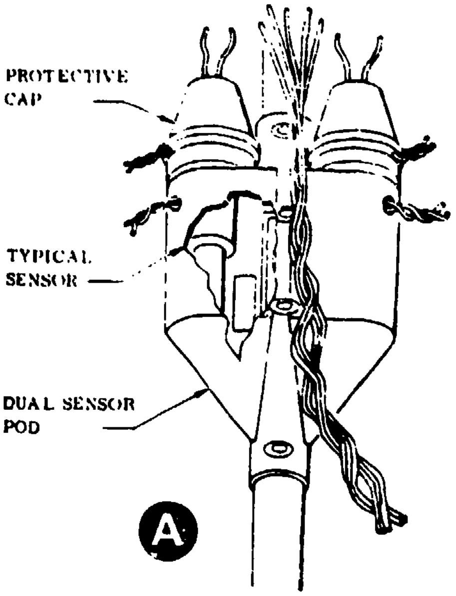
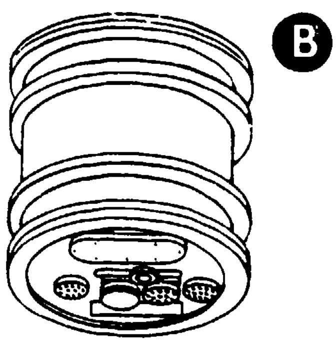


Figure C3-3 Missile Propulsion System



TYPICAL SENSOR POD



PROPELLANT UTILIZATION COMPUTER ASSEMBLY

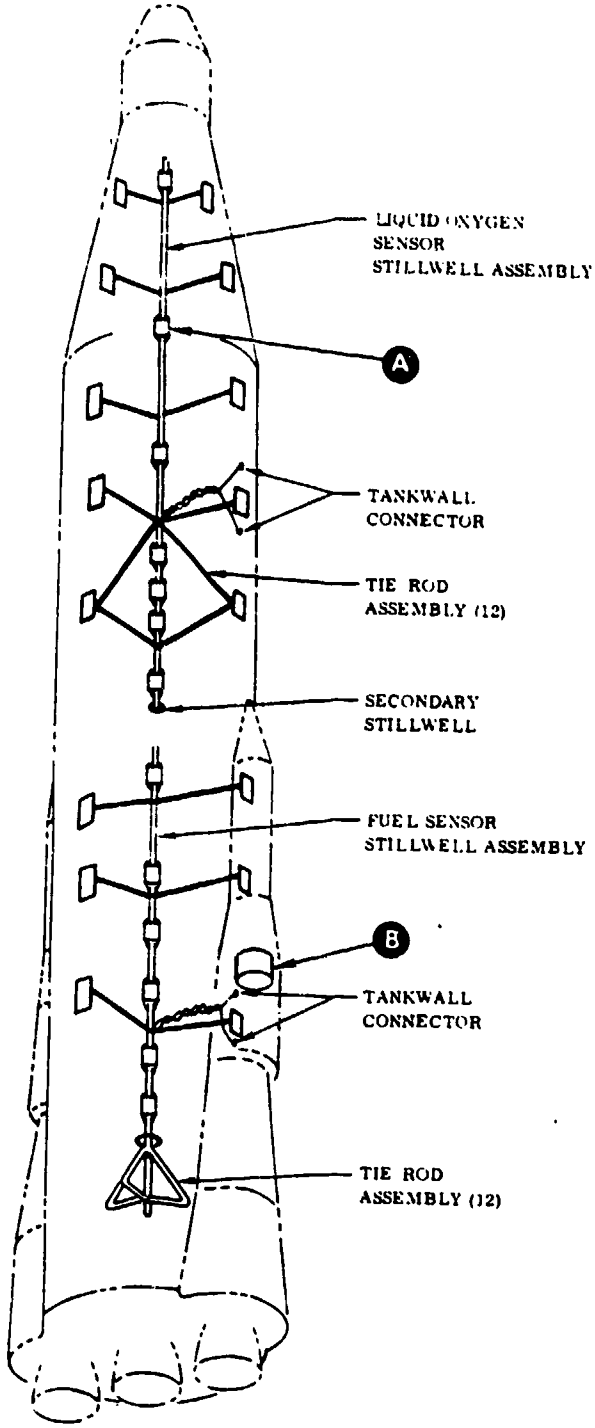


Figure C3-4 Propellant Utilization System

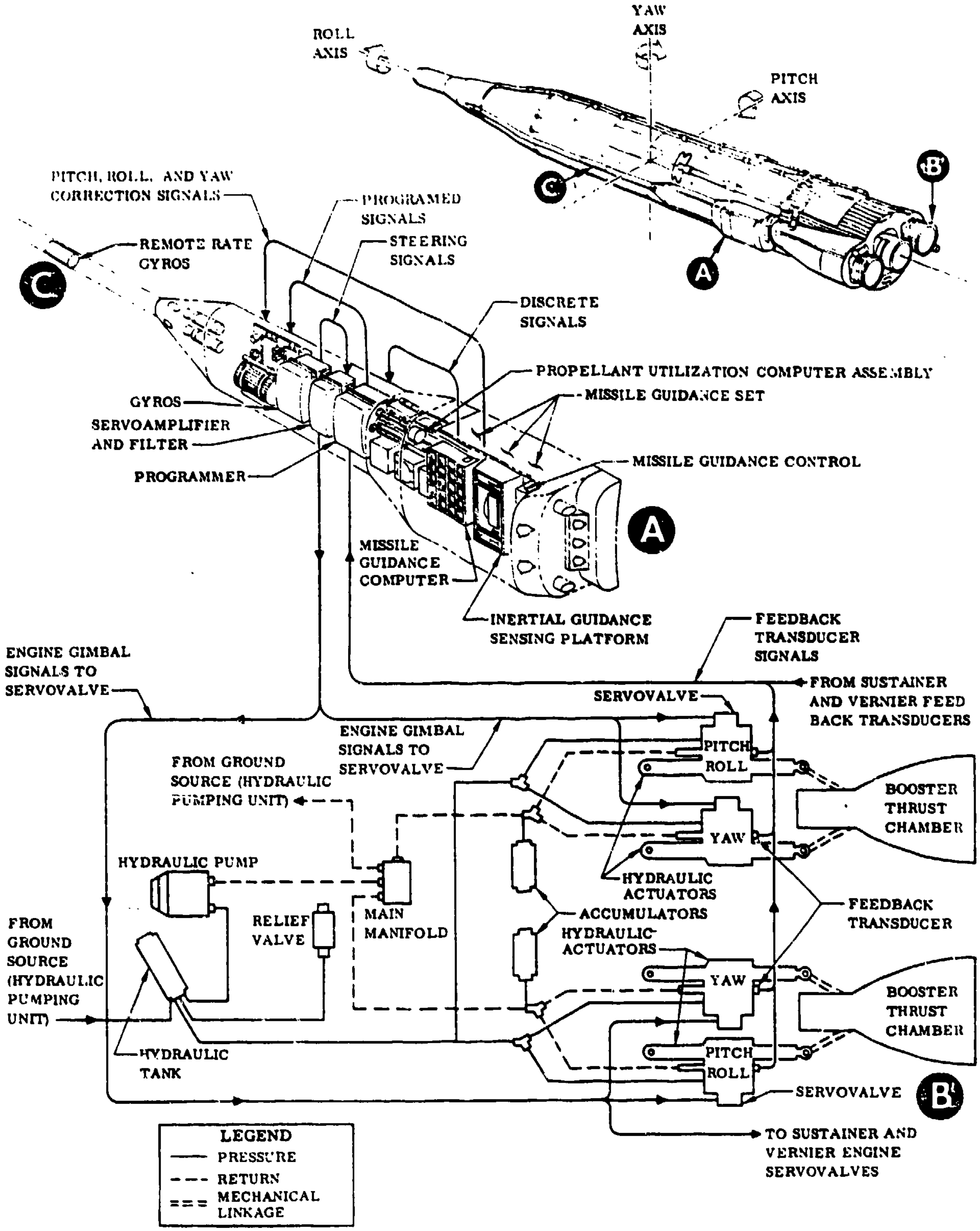


Figure C3-5 Missile Flight Control System, Guidance System, and Booster Hydraulic System

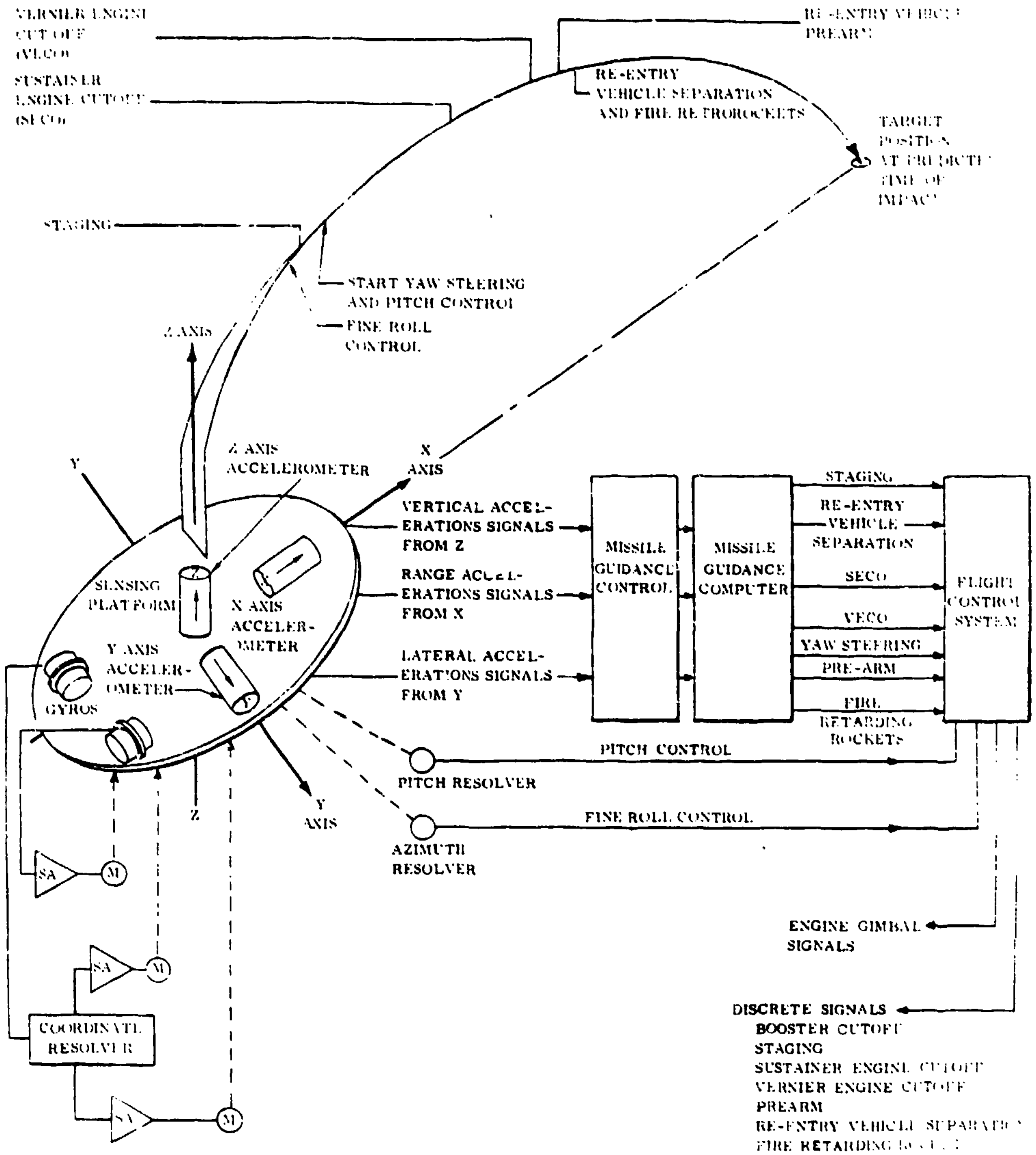


Figure C3-6 Inertial Guidance System

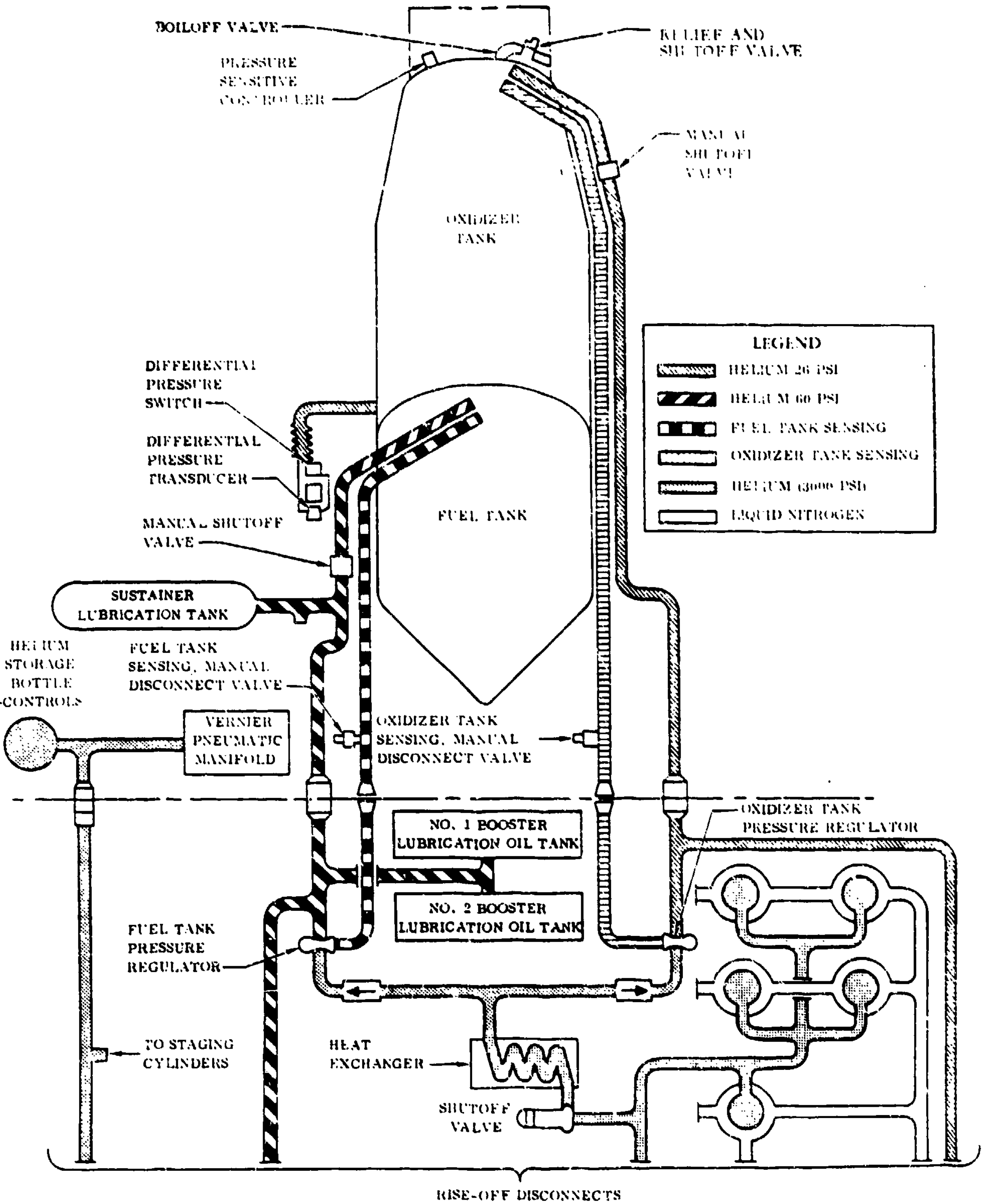


Figure C3-7 Missile Pneumatic System

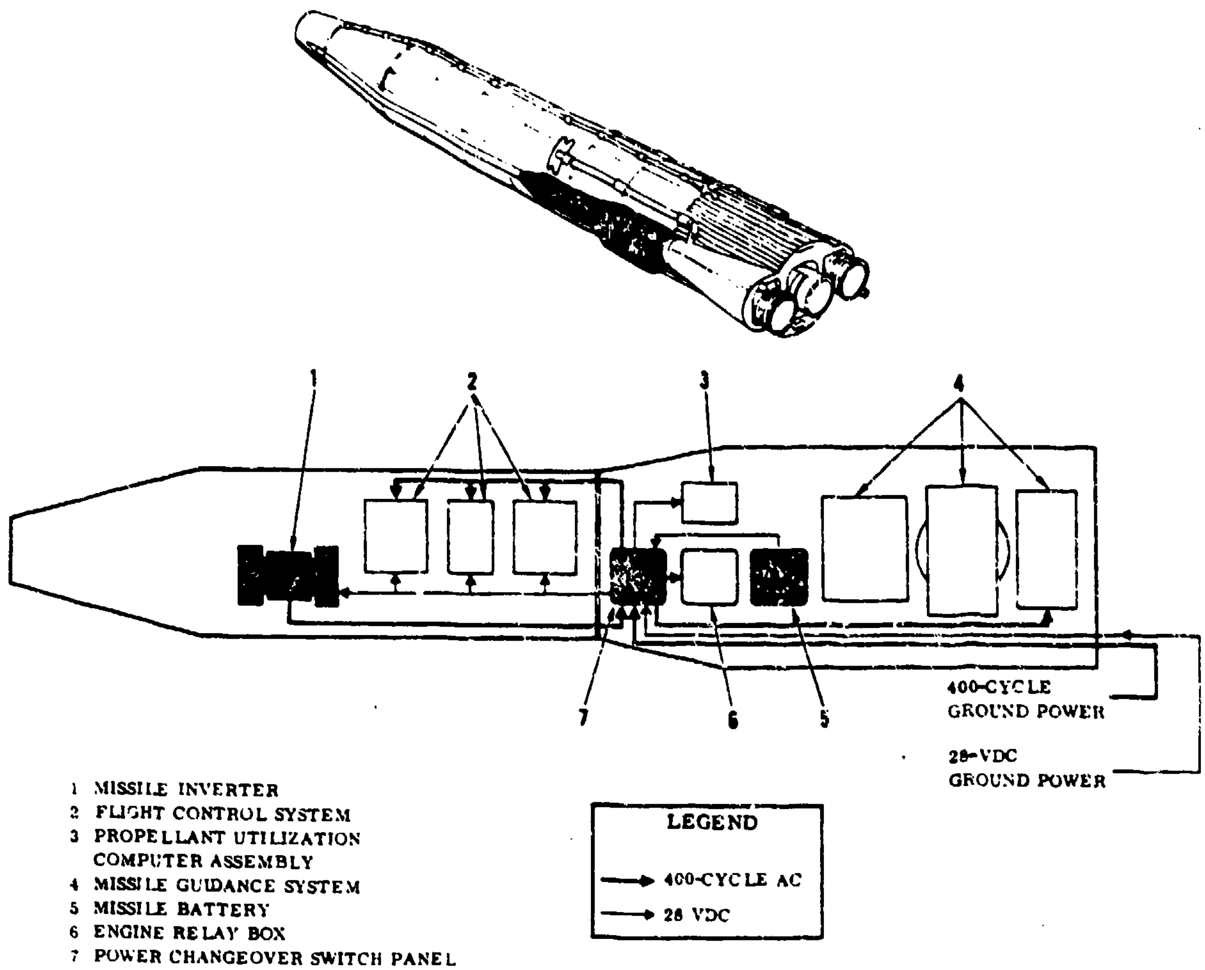


Figure C3-8 Missile Electrical System

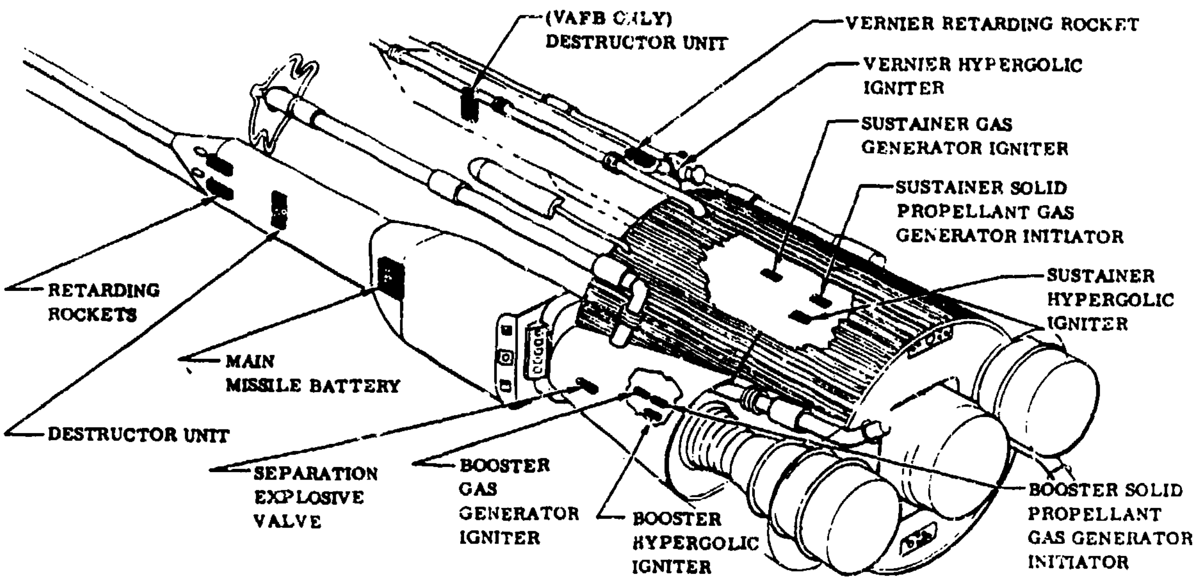


Figure C3-9 Missile Explosion Assemblies

Appendix C4
Atlas/Centaur

APPENDIX C4
ATLAS/CENTAUR

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APPENDIX C4
ATLAS/CENTAUR

C4.0 INTRODUCTION

The following data were extracted from the Countdown Summary, Launch/Hold Criteria, and Flight Events, ac-63, (Intelsat VA Mission F-10), Convair Division of General Dynamics, San Diego, CA, Feb 1985 and the Atlas/Centaur Configuration, Performance, and Weight Status Report, compiled by H.D. Nilsen, General Dynamics, Convair Division, Mar 1985.

C4.1 GENERAL DESCRIPTION

Atlas/Centaur (See Fig. C4-1) is 41.9 Meters (137.6 feet) tall and 3 meters (10 feet) in diameter. With payload, it weighs approximately 163,523 kilograms (360,500 pounds) at liftoff. Thrust of the Atlas is 1,950,074 newtons (438,416 pounds) at liftoff, and thrust of Centaur is 146,784 newtons (33,000 pounds) in a vacuum.

C4.1.1 Launch Vehicle

Atlas/Centaur vehicles are built by General Dynamics/Convair (GD/C) and launched by a combined NASA/GD/C team. This two-stage, liquid-fueled vehicle has been used to launch a variety of scientific and technological spacecraft. These have included Surveyors to the moon; Mariners to Venus, Mercury, and Mars; and Pioneers to Jupiter/Saturn. It has placed Applications Technology Satellites and COMSTAR, INTELSAT, and FLTSATCOM communications satellites into geosynchronous transfer orbits. Orbiting Astronomical Observatories weighing up to 2,268 kilograms (5,000 pounds) have been placed in orbits as high as 740 kilometers (460 statute miles) above the earth. The Atlas/Centaur is the most powerful unmanned vehicle now launched by NASA. In 1984 it was upgraded by lengthening the Atlas stage to provide larger propellant tanks. The Centaur stage has been improved by substituting attitude control thrusters powered by hydrazine (used as a monopropellant) for ones powered by hydrogen peroxide and by replacing the oxygen and hydrogen propellant pumps by pressure-fed systems.

C4.1.2 Atlas Stage

The 23.3-meter (76.3-foot) first stage is an uprated version of the flight-proven Atlas vehicle used in the national space program since 1959. The Rockwell International/Rocketdyne MA-5 engine system burns RP-1, a highly refined kerosene, and liquid oxygen. The MA-5 uses two main engines, a 1,679,120-newtons (377,500-pound) thrust booster engine with two thrust chambers and a smaller sustainer with a single thrust chamber that produces 266,900 newtons (60,000 pounds) of thrust. The sustainer nozzle is located between the two larger ones of the booster engine. Two small vernier engines that help control the vehicle in flight are also burning at liftoff for a total thrust of 1,950,074 newtons (438,416 pounds). Total weight at liftoff is about 163,523 kilograms (360,500 pounds).

An unusual feature of the Atlas vehicle is its "stage-and-a-half" construction. All five thrust chambers are burning at liftoff. After more than 2.5 minutes of flight, the booster engine cuts off and its supporting structures are jettisoned, deleting a large portion of the structural weight of this stage. The sustainer and vernier engines continue to burn until the propellants are gone, at about 4.75 minutes. This means that an Atlas retains most of the weight reduction advantage gained by jettisoning a used-up stage but does not have to ignite its engines in flight as a separate stage must.

The only radio frequency system on the Atlas is a range safety command system consisting of two receivers, a power control unit, and a destruct unit. The Atlas can be destroyed in flight by ground control if necessary, but otherwise receives all its control directions from the Centaur stage.

C4.1.3 Centaur Stage

The Centaur stage sits above the Atlas on a barrel-shaped interstage adapter. The Atlas and Centaur separate two seconds after the Atlas burns out. Eight small retrorockets near the bottom of the Atlas fuel tank then back this stage away from the Centaur.

The Centaur stage is 9.1 meters (30 feet) in length without the fairing on top. Exclusive of payload, it weighs about 17,700 kilograms (39,000 pounds) when loaded with propellants. The main propulsion system consists of two Pratt and Whitney engines burning liquid oxygen and liquid hydrogen, producing 146,784 newtons (33,000 pounds) thrust in the vacuum of space in which they are designed to operate. These engines can be stopped and restarted, allowing the Centaur to coast to the best point from which to achieve its final trajectory before igniting for another burn. While coasting, the stage is controlled by 12 small thruster engines, powered by hydrazine. These hold the stage steady and provide a small constant thrust to keep the propellants settled in the bottoms of their tanks, as needed for a second or third burn.

A cylindrical nose fairing with a conical top sits on the Centaur and protects the spacecraft. Total vehicle height is 41.9 meters (137.6 feet). Both stages are three meters (10 feet) in diameter.

An adapter on top of this module connects to the payload adapter on the bottom of the spacecraft. These electronic packages provide an integrated flight control system that performs the navigation, guidance, autopilot, attitude control, sequence of events, and telemetry and data management functions for both the Atlas and Centaur stages. The heart of this system is a Digital Computer Unit (DCU) built by Teledyne. The DCU sends commands to control most planned actions, including all but items one, two, and five in Table C4-1. The DCU receives guidance information from a combination of sensors called the Inertial Measurement Group, built by Honeywell, and sends steering commands to all Atlas and Centaur engines. The Centaur also has a ground-controlled destruct system, similar to that on the Atlas, in case the vehicle must be destroyed in flight.

The Centaur uses the most powerful propellant combination available, has a lightweight structure, and has an engine burn time of up to 7-1/2 minutes, the longest of any upper stage now in service. This gives it the most energy for its size of any stage yet built. (See Figs. C4-2 through C4-5 for configurations) (See Table C4-1 for a performance summary for the FLTSATCOM mission.)

C4.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C4.2.1 Atlas Stage

1. Stage Identification - EID69-0050

2. Vehicle Structures
   Atlas G Configuration (81 inches longer than Atlas D)
   Additional Thermal Protection (B1 and B2 pod noses with cork insulation and chemseal on aft nacelle in area of kick strut fitting)

3. Propulsion System
   Rocketdyne MA-5 system (two booster engines, two vernier engines, one sustainer engine)

   a. Atlas Booster Engines
      - 370K Thrust Biased:

Thrust, lb	377,500
Isp, sec	258.96
Mixture Ratio	2.25

   b. Atlas Sustainer Engine

      - 60K Thrust Biased

Thrust, lb	60,500
Isp, sec	220.008
Mixture Ratio	2.22

   c. Atlas Vernier Engines

- Thrust, lb	922.1
- Isp, sec	178,538
- Mixture Ratio	1.5324

   (See Figs. C4-6 through C4-9 schematics)

Sustainer LO_2 reference pressure regulator modified to sense constant sea-level reference pressure during flight

Minimum residual propellant sensors

Engine relay box (mounted in equipment pod)

Table C4-1 Performance Summary for FLTSATCOM Mission, ac-67

FLTSATCOM MISSION	ac-67 CURRENT	PREVIOUS	CHANGE
Nominal Centaur Burnout Weight (BOW)	9,472	9,471	+1
Centaur Mission Peculiar Jettison Weight	4,240	4,224	+16
Basic Jettison Weight	4,142	4,124	+18
Spacecraft Adapter	86	86	0
Centaur-LVMP (Launch Vehicle Mission Peculiar)	12	14	- 2
Payload Systems Weight Commitment (PSWCMT)	5,112	5,114	- 2
Payload System Weight Capability (PSWC)	5,112	5,114	-2
Payload System Weight (PSW)	5,112	5,114	- 2
Separated Spacecraft	5,014	5,014	0
Spacecraft Adapter	86	86	0
Centaur LVMP	12	14	- 2
Nose Fairing LVMP	55	55	0
Net Payload Capability Margin (PCM)	0	0	0
Centaur Propellant Ecess (PE)	218	233	-15
Flight Performance Reserve (FPR)	212	220	- 8
Launch Vehicle Contingency (LVC)	6	13	- 7
Centaur Propellant Margin (PM)	0	0	0

4. Hydraulic System

Booster hydraulic pump accumulator, hydraulic fluid tank, and four engine actuators.

Sustainer/vernier hydraulic pump, accumulator, hydraulic fluid tank, and six engine actuators (vernier yaw actuators locked and non-operating).

Fuel/hydraulic interface valve

(See Fig. C4-10 schematic)

5. Pneumatic System

Tank pressurization - refrigerated helium storage
Eight bottles, 34.1 ft^3

Engine controls - one ambient temperature helium bottle, 2.69 ft^3

LO_2 tank venting - single boiloff valve

Programmed LO_2 tank pressurization

Propellant Densities (Tanking)

- Atlas RP-1, 49.81 lb/ft^3 at 77.7°F (mean annual)
- Atlas RP-1, 49.69 lb/ft^3 at 82.57°F (June)
- Atlas RP-1, 50.05 lb/ft^3 at 67.58°F (March)
- Atlas LO_2, 69.40 lb/ft^3 at 25.00 psia

Atlas Tanking Pressures (Minimum and Maximum)

- RP-1 tanking, Phase I
  - LO_2 tank, 6.55 - 7.45 psig
  - RP-1 tank, 14.65 - 16.10 psig

- LO_2 tanking, Phase II
  - LO_2 tank, 9.50 - 11.00 psig
  - RP-1 tank, 67.45 - 69.20 psig

- Phase III
  - LO_2 tank, 28.50 - 30.20 psig
  - RP-1 tank, 67.45 - 69.20 psig

- Flight/Internal
  - LO_2 tank, 26.0 29.0 psig t = 0 to 20 sec
    - 33.0 - 35.0 psig t = 20 to 60 sec
    - 32.0 - 35.0 psig t = 60 sec up
  - RP-1 tank, 64.0 - 67.0 psig

(See Fig. C4-11 schematic)

6. Propellant Utilization

LO_2 and fuel mercury manometer matched set

Propellant Utilization Unit (PUU)

Tank Volume, Nominal (Including Lines)

- Atlas G
  RP-1, 1911.69 ft^3
  LO_2, 3078.72 ft^3

Ullage Volume (Nominal at Tanking0

- Atlas RP-1, 19.61 ft^3

- Atlas LO_2, 28.36 ft^3

(See Figs. C4-12 and C4-13 schematics)

7. Electrical System

Single 28-Vdc main vehicle battery

Single phase inverter (combined with servo in SIU)

Power changeover switch (dc only)

Pyrotechnic control unit (mounted in B-1 equipment pod)
Programmed pressurization - one box
Booster section separation - one box

Retrorocket pyro control unit (mounted in B-2 equipment pod)

8. Range Safety Command

Two RSC receivers
Two 28-Vdc RSC batteries
One destruct unit

Power control unit

Hybrid junction

Two antennas (one in each equipment pod)

9. Instrumentation and Telemetry System

PCM signal conditioner

Remote multiplexer

9. Separation Systems

Booster thrust section - ten pneumatically operated separation fittings

Atlas/Centaur - eight retrorockets at Sta 1133

(See Fig. C4-14 for Atlas/Centaur Separation Instrumentation and Antenna Locations.)

10. Flight Control

2-R ate gyro package mounted at Sta 454

Servoamplifier combined with inverter (SIU)

Derived roll rate signal from Centaur

Position reference from Centaur

Isolation transformer

(See Figure C4-15 for Atlas flight control system block diagram.)

C4.2.2 Centaur Stage

1. Vehicle Structures

Centaur D-1 configuration without booster pumps.

Equipment module for electronic packages and payload support

LH_2 tank baffle not installed

2. Stub Adapter

Approximately 25-inch-high aluminum skin and stringer cylindrical structure (approximately 120-inch diameter) with cork and hinge beef-up.

3. Equipment Module

Approximately 119 to 68-inch-diameter aluminum skin and stringer conical structure for mounting electronic packages and payload adapter support.

4. Payload Support

Convair adapters - aluminum skin and stringer construction:
INTELSAT, 18.75-inch-high conical
FLTSATCOM, 23.6 inch-high cylinder

5. Insulation

Forward bulkhead - multilayer insulation blankets

Aft bulkhead - thermal radiation shielding

Cylindrical section - four jettisonable fiberglass sandwich panels

6. Payload Fairing

Approximately 29-foot-high conical/cylindrical fiberglass nose fairing attached to a 56-inch aluminum skin and stringer split barrel.

7. Propulsion System

Centaur Main Engines RL10A-3-3A performance
(Two engines)

Nominal at 5:1 Mixture Ratio with silver throat.	
Thrust, lb	32,823
Isp, sec	448.37
Centaur Engine prestart impulse	
First Burn, lb-sec	2,670
Second Burn, lb-sec	2,540
Centaur Engine start impulse (MES to MRS +3.0 sec)	
First Burn, lb-sec	34,940
Second Burn, lb-sec	34,940
Centaur Engine Shutdown Impulse (MECO to MECO + 0.5 sec)	
First burn, lb-sec	3,407
Second Burn, lb-sec	3,127
Centaur Engine Leakage (Parking Orbit Coast)	
LH_2 W lb/min	0.0026
LO_2 W, lb/min	0.0078

(See Fig. C4-16, Centaur engine propellant flow)

9. Reaction Control System (RCS)
   - Two clusters - four 6-lbf thrusters per cluster
   - Two clusters - two 6-lbf thrusters per cluster
   - One hydrazine bottle with 170-lbf storage capacity
   - Associated valves
   - Heated feed lines
   - Hydrazine control panel with heaters

(See Fig. C4-17, Centaur Hydrazine RCS)

10. Hydraulic System

Hydraulic power units (one on each engine)

Four engine actuator assemblies

(See Fig. C4-18, for Centaur hydraulics system schematic:)

11. Pneumatic System

a. Helium bottle supply (tank + engine controls)

(1) Titanium (19,380 in^3 total)
one 4650 in^3 bottle
two 7365 in^3 bottles

(2) Composite (16,362 in^3 total)
two 8181 in^3 (minimum) bottles

b. Inflight purge system - single helium bottle supply 4650 in^3 in ISA

c. Zero-g purge on LO_2 standpipe

d. Tank venting

(1) Dual LH_2 tank vent valves, both with locking capability

(2) Single LO_2 vent valve

(3) Dual LH_2 vent ducts

e. Pre-MES tank pressurization (helium)

(1) Pressurization of vehicle tanks controlled by DCU/SCU using three analog pressure sensors in each tank

(2) LH_2 tank pressurization - two valves in parallel

(3) LO_2 tank pressurization - two valves in parallel

(4) Control valve upstream of the primary solenoid valves

f. Tank pressurization during engine firings

(1) Pressurization of vehicle tanks controlled by DCU/SCU using three analog pressure sensors in each tank

(2) LH_2 tank pressurization
GH_2 bleed pressurization

(a) Two parallel legs with two valves in series

(b) One leg with constant flow orifice

(3) LO_2 Tank Pressurization
Helium pressurization (same system as pre-MES pressurization)

Centaur Tanking Pressures

Regulating Range (minimum and maximum)
LO_2 vent valve range, 29.0 - 32.0 psia
LH_2 primary vent valve range, 19.0 - 21.5 psia
LH_2 secondary vent valve range, 24.8 - 26.8 psia

Propellant Densities (Tanking)

Centaur LH_2, 4.20 lb/ft^3 at 21.80 psia
Centaur LO_2, 68.6 lb/ft^3 at 30.30 psia

(See Figs. C4-19 and C4-20 for Centaur Pneumatic System.)

12. Propellant Level Indicating System

Probe mounted sensors internal to LO_2 and LH_2 tanks

13. Propellant Utilization

PU electronics incorporated within SIU

LO_2 and LH_2 tank probes

Two servopositioners

Tank Volume, Nominal

Centaur
LH_2, 1268.22 ft^3
LO_2, 376.94 ft^3

Ullage Volume

Centaur LH_2, 11.20 ft^3

Centaur LO_2, 6.60 ft^3

(See Figs. C4-21 and C4-22 for Centaur Propellant Utilization System.)

14. Guidance and Flight Control

Digital computer unit (DCU) (contains PCM central control unit, random access core memory)

Inertial measurement group (IMG) (consists of inertial reference unit and systems electronics unit)

Servo inverter unit (SIU)

Sequence control unit (SCU)

15. Range Safety Command

    Two RSC receivers
    Two 28-Vdc RSC batteries
    Power control unit

    Hybrid junction

    Two tank-mounted antennas

    Safe/arm initiator mounted on Centaur

    High-explosive block charge

    MDF (from H.E. charge to S/A initiator)

    (See Fig. C4-23 for Centaur Range Safety System.)

16. Tracking System

    C-Band transponder

    Two tank mounted antennas

17. Electrical System

    Single 28-Vdc main vehicle battery (100 A-Hr)
        with Lightweight Case

    Static inverter
        Combined with SCU

    Power changeover switch
        Combined with SCU

    Umbilicals
        Three Centaur
        Payload umbilical (one)

18. Instrumentation and Telemetry System

    Ring Coupler

    Two S-Band antennas mounted on stub adapter

    PCB Telemetry
        S-Band transmitter
        Remote multiplexer (1)
        Signal conditioner
    Add instrumentation for insulation panels separation.

    R&D Instrumentation

19. Interstage Adapter

Aluminum skin and stringer structure

20. Pyrotechnic Systems

Centaur to booster separation

One pyrotechnic control unit in ISA
Shaped charge and detonator installation
Fired by Atlas vehicle battery

Insulation panel separation
One pyrotechnic control unit in nose fairing
One pyrotechnic control unit in ISA
Shaped charge and detonator installation
Fired by Atlas vehicle battery/pyrotechnic batteries

Insulation panel vent/equipment module vent actuation
One pyrotechnic control unit in nose fairing

Nose fairing separation
Four pyrotechnic control units in nose fairing

Two pyrotechnic batteries

Explosive bolts

Payload pyrotechnic functions
Pyrotechnic control unit

Pyrotechnic shutoff valves
Pyrotechnic control unit mounted in interstage adapter (fired by Atlas battery)

Pyrotechnic hydrazine isolation valves
Pyrotechnic control unit mounted in ISA (fired by Atlas battery)

(See Fig. C4-24 for Atlas/Centaur Separation Pyrotechnic Diagram.)

C4.3 MISSION SCENARIO

Figure C4-25 gives a mission summary for the INTELSAT VA (F10), which places a communication satellite into geosynchronous orbit.

C4.3.1 Launch Countdown Automatic Sequence Functions

1. Automatic Sequence Prerequisites

The major operational events required to complete the launch control relay network for entering the final "automatic sequence" portion of the launch countdown (at T-31 seconds) are listed in Table C4-2. These prerequisites begin on F-1 Day and end with automatic sequence start enable at T-40 seconds.

Table C4-2 Automatic Sequence Prerequisites

Event	Time (Nominal)
A. TCC Prestart Ladder Preparation	
1. Launch/Test Switch to Launch	T-900 Min (F-1 Day)
All Engines Null	T-120 Min
Holdown and Release Ready	T-100 Min
Eject Ready	T-90 Min & Holding
CCLS Ready	T-6 Min
2nd Stage Engines Ready	T-5 Min
Boom System Ready	T-4 Min
1st Stage Power Ready	T-4 Min
Water System Ready	T-4 Min
2nd Stage Pressure Ready	T-3 Min
1st Stage Tanking Ready	T-2 Min 35 Sec
2nd Stage Power Ready	T-2 Min
1st Stage Range Safety Ready	T-2 Min
2nd Stage Range Safety Ready	T-2 Min
Range Ready	T-1 Min 30 Sec
2nd Stage Tanking Ready	T-75 Sec
1st Stage Pressure Ready	T-70 Sec
Hydraulics Ready	T-55 Sec
2. Prestart Ladder Complete	T-40 Sec
B. Atlas Engines Preparation	
1. Atlas Eng Gas Gen Fuel & LO_2 Purge On	T-5 Min
2. Engine Control Arm Switch to Arm	T-2 Min 35 Sec
3. Atlas Engines Prep Complete	T-2 Min 35 Sec
C. Vehicle Preparation Complete	
Contingent Upon:	
Atlas Engines Prep Complete	
TCC Prestart Ladder Complete	

2. Automatic Sequence Events

The automatic sequence events that can cause a launch abort/recycle from T-30 sec to T-0.60 sec are listed in Table C4-3

Table C4-3 Automatic Sequence Events

Event	Time (Nominal)
DCU Count Started	T-29.9 Sec
Arm-Safe SW-1 Armed Monitor	T-29.8 Sec
Arm-Safe SW-2 Armed Monitor	T-29.8 Sec
LH_2 Vent Valve Closed	T-29 Sec
A/B Instillation Panel Purge OK	T-10.5 Sec
Flight Mode Accept	T-9.2 Sec
LO_2 Start Tank Pressurized	T-9.1 Sec
Fuel Start Tank Pressurized	T-8.9 Sec
Insulation Panel Vent and Equipment Module Doors Open	T-8.0 Sec
Cent Upper Umbilicals and Aft Plate Eject Command	T-4.4 Sec
Cent Upper Umbilicals and Aft Plate Ejection Complete	T-3.8 Sec
T-0 Umbilicals Mated	T-3.8 Sec
Flight Mode Accept Removed	T-3.10 Sec
Gas Generators Ignition	T-3.08 Sec
Atlas Main Stage	T-2.10 Sec
Main Engines Complete	T-0.77 Sec
Release Signal	T-0.60 Sec

3. Atlas/Centaur INTELSAT Flight Mark Events

The major flight events (called Mark Events) to be reported in real time during the launch by ETR and/or KSC personnel on TOPS Channel 2 are presented in Table C4-4. Figure C4-26 presents a Flight Events Profile.

Table C4-4 ac-63/INTELSAT VA (F-10) Mark Events

Mark Event No.	Description	Time from Liftoff (Sec)	(Min:Sec)	Altitude (n.mi.)	Range (n.mi.)	Velocity (ft/sec)*
0	Liftoff	0	0:00.0	0	0	0
1	BECO	153.6	2:33.6	31.5	46.8	7,976
2	Booster Jettison	156.7	2:36.7	33.0	50.7	8,222
3	Panels Jettison	178.6	2:58.6	43.0	79.1	8,596
4	Nose Fair Jett.	226.9	3:46.9	61.5	150.2	10,056
5	SECO	284.2	4:44.2	78.7	253.0	12,563
6	A/C Separation	286.2	4:46.2	79.2	257.1	12,579
7	MES1	296.8	4:56.8	81.8	278.1	12,542
8	MECO1	615.2	10:15.2	83.1	1185.4	24,207
9	MES2	1379.8	22:59.8	162.4	4241.9	24,657
10	MECO2	1469.9	24:29.9	187.1	4632.3	31,703
11	Start Cent. Spin	1588.9	26:28.9	255.2	5212.2	31,345
12	S/C Separation	1604.9	26:44.9	266.7	5282.5	31,282
13	End Cent. Spin	1614.9	26:54.9	275.0	5329.3	31,259
14	Start Centaur Retro Turn	1619.9	26:59.9	279.2	5352.7	31,215

*Velocity is relative to a rotating earth.

Data Source: GDC Report No. GDC-SP-85-002, "Firing Tables for the INTELSAT VA Mission, Atlas/Centaur ac-63," dated January 1985.

NOTE: The final velocity of 34,783 kilometers (21,613 miles) per hour places the spacecraft in a transfer orbit with an apogee of 35,818 kilometers (22,256 miles) and a perigee of 311 kilometers (193 miles). INTELSAT then assumes control of the spacecraft. At an apogee chosen by INTELSAT controllers, the onboard apogee kick-motor will be fired to circularize the orbit at geosynchronous altitude, 35,789 kilometers (22,238 miles) above the equator. It will then be "drifted" to its assigned place in the INTELSAT global network. The spacecraft will have a final velocity of about 11,071 kilometers (6,879 miles) per hour. It will complete one orbit every 24 hours and so remain above the same spot of the equator. Figure C4-27 presents a Mission Trajectory Profile.

C4.3.2 Nonstandard Sequencing

Nonstandard sequencing is provided by the DCU sequencer software if an abort situation presents itself, if the Centaur engines do not ignite, or if premature MECO occurs.

1. Abort Sequence

If a pre-liftoff test fails prior to GO-INERTIAL Plus 6.00 seconds (T-4.00 sec), the first program will immediately enter the abort sequence. If liftoff is not detected by GO-INERTIAL plus 59.98 seconds (T+49.98 sec), the flight program will initiate the abort sequence.

2. Restart Sequence

Six seconds after a MES command, acceleration level is checked to see if the engines started. If acceleration is below 6 ft/sec^2, it is assumed that the engines did not start. The sequencer then goes through a restart sequence resulting in a second MES command 10 seconds later for MES1 and 80 seconds later for MES2.

Only one restart attempt is made per MES event. If the restart is unsuccessful, the failure is treated as a premature MECO. The spacecraft is not released if MES1 restart is not successful. Failure of the MES2 restart attempt will result in a sequence advance to MECO2, which will be followed by normal sequencing, including spacecraft separation.

3. Premature MECO

Seven seconds after an MES command, assuming a successful MES, the sequencer began looking for a premature thrust decay (a_T 6ft/sec^2). If this occurs, the sequencer will issue all remaining discretes in the phase at 60-millisecond intervals and continue normal sequencing from there.

4. Tumble Recovery

Tumble recovery is enabled during parking orbit coast, transfer orbit coast, and MES1 restart sequences. During these times, the sequencer tests attitude error against a preset limit (20 degrees in all phases except transfer orbit and post-tumble recovery maneuver, which test against a 30-degree limit). When this limit has been exceeded for three consecutive 20-millisecond intervals, the vehicle is considered to be tumbling. Null steering vectors are then issued until acceptable vehicle rates are attained. Once the vehicle has recovered (rate 1 deg/sec), normal vehicle control returns the vehicle to the desired attitude.

5. Insulation Panel Jettison Backup Sequence

If the insulation panels have not been jettisoned prior to SECO -11 seconds, the sequencer will unconditionally set Switch 33 to jettison the panels at this time. Two seconds later Switch 33 will be reset. This sequence ensures that the panels will be jettisoned prior to SECO.

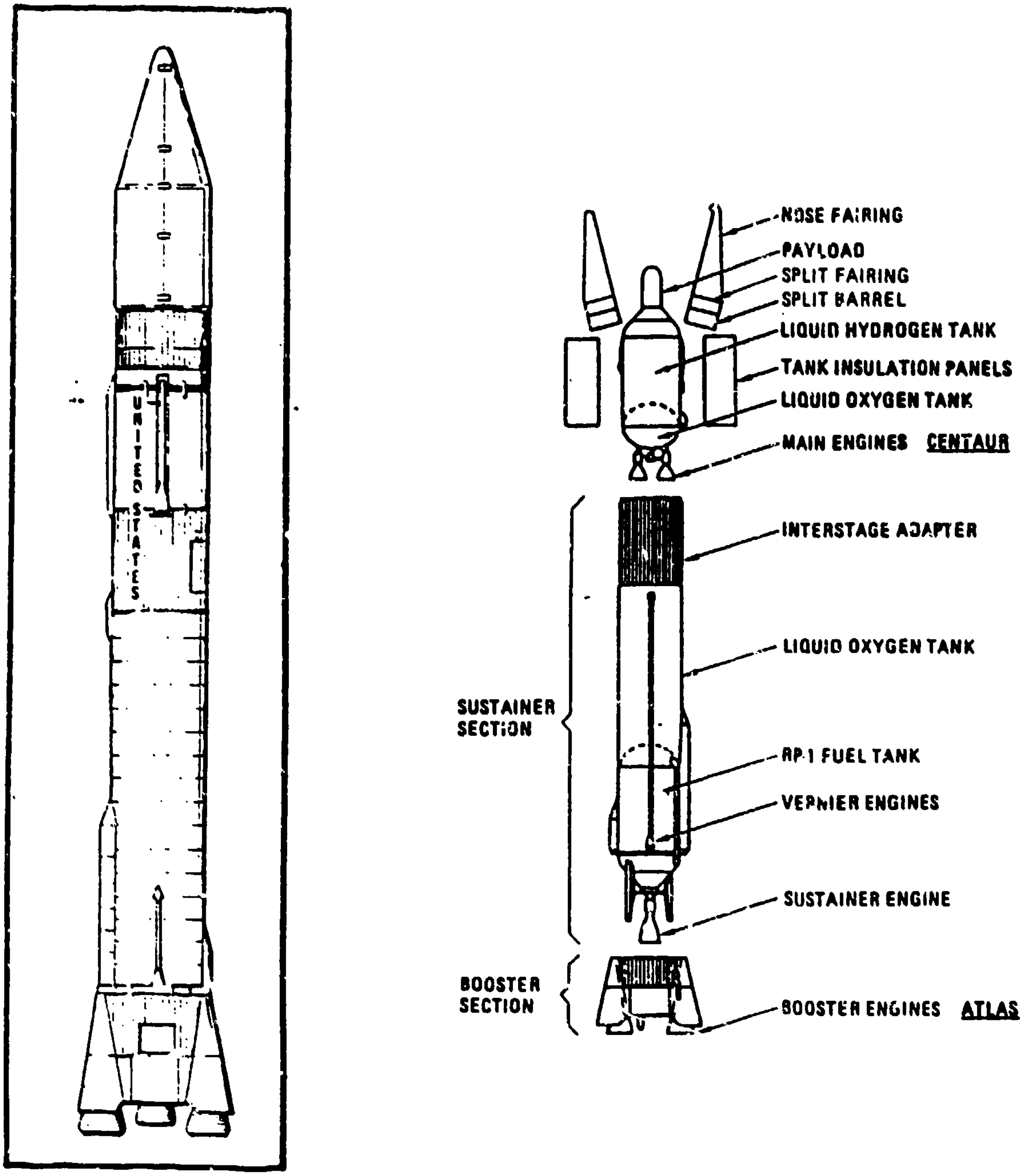


Figure C4-1 Atlas/Centaur - Launch Vehicle

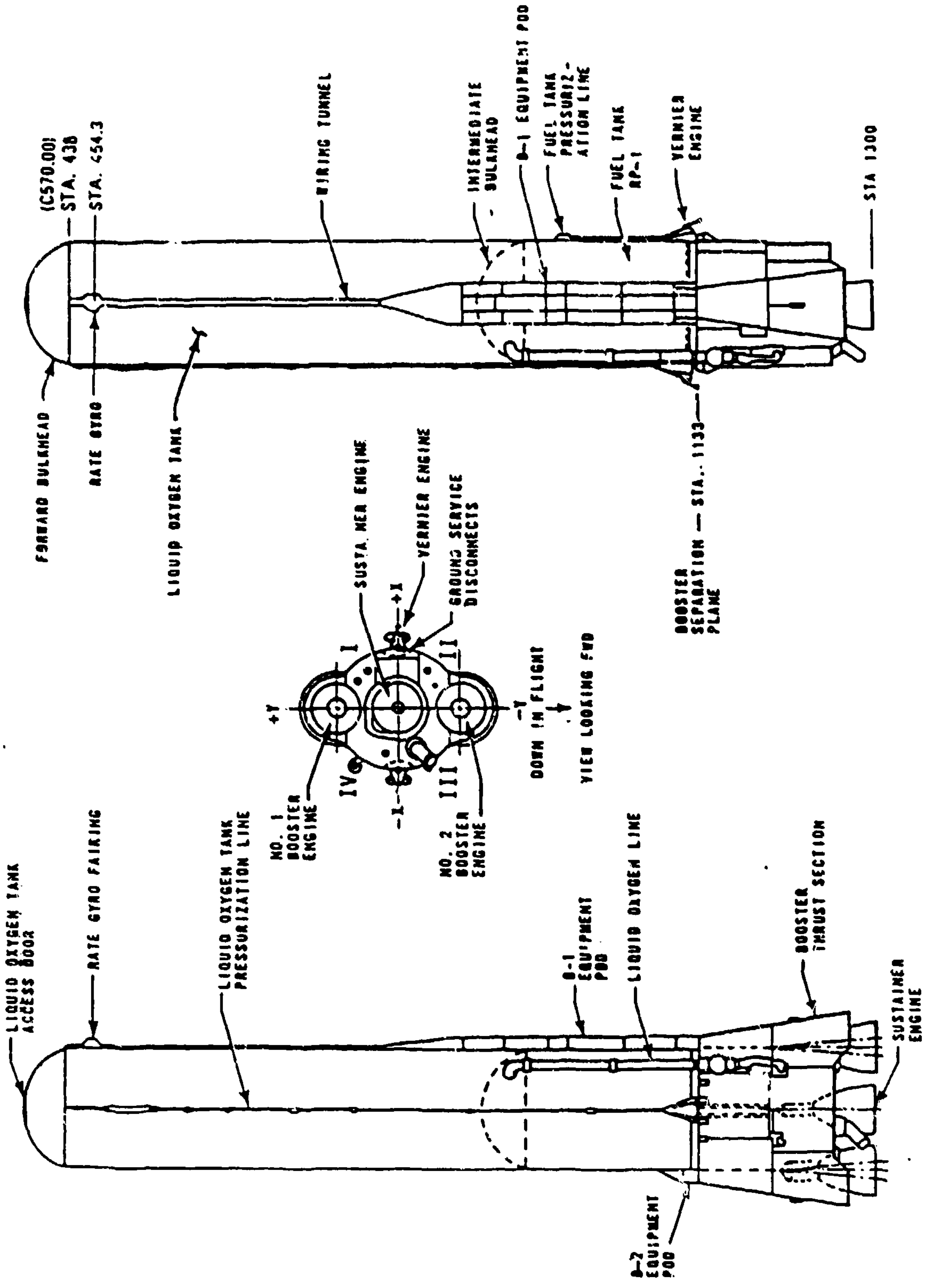


Figure C4-2. Atlas G Assembly

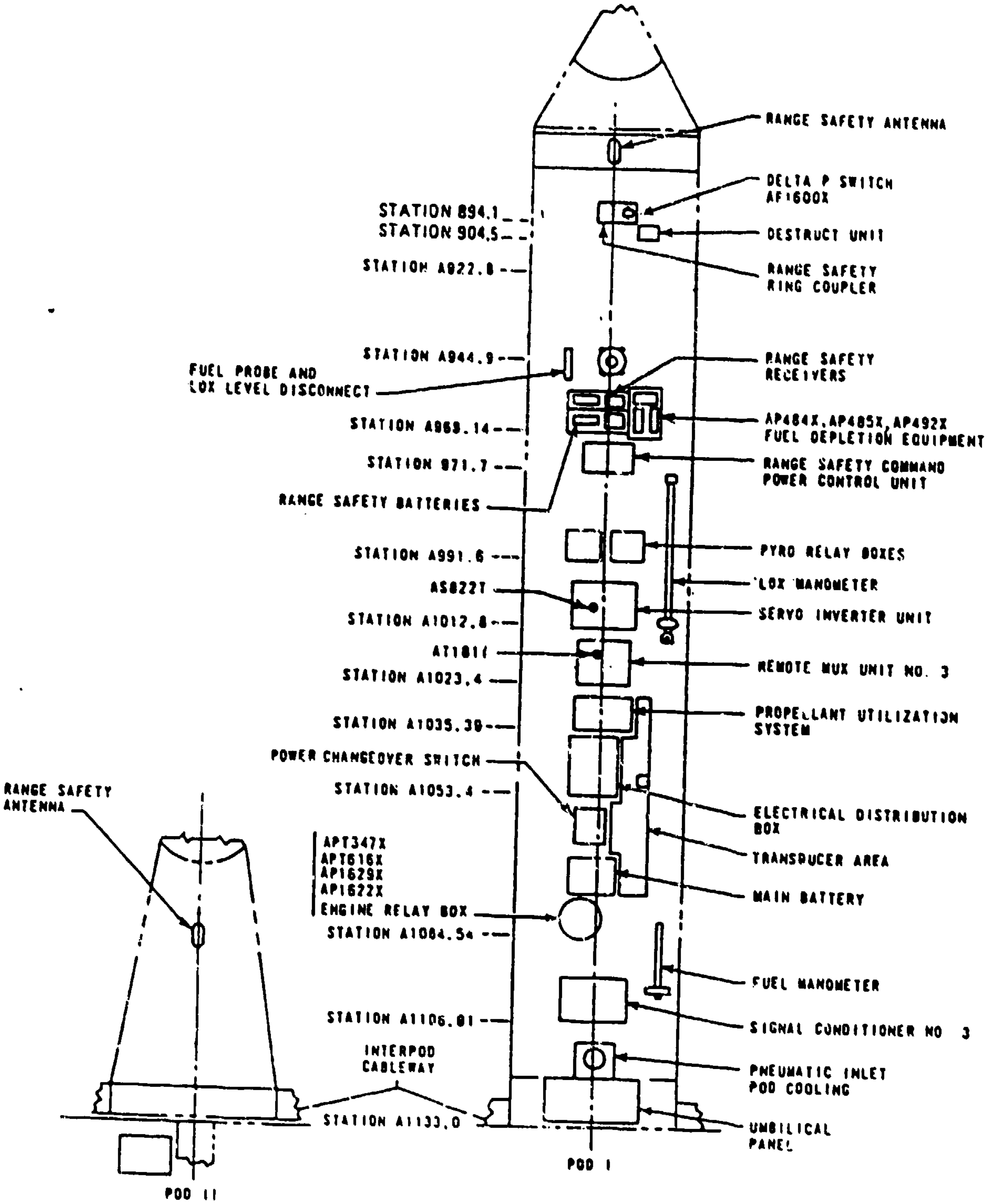


Figure C4-3 Atlas G Pod Configuration

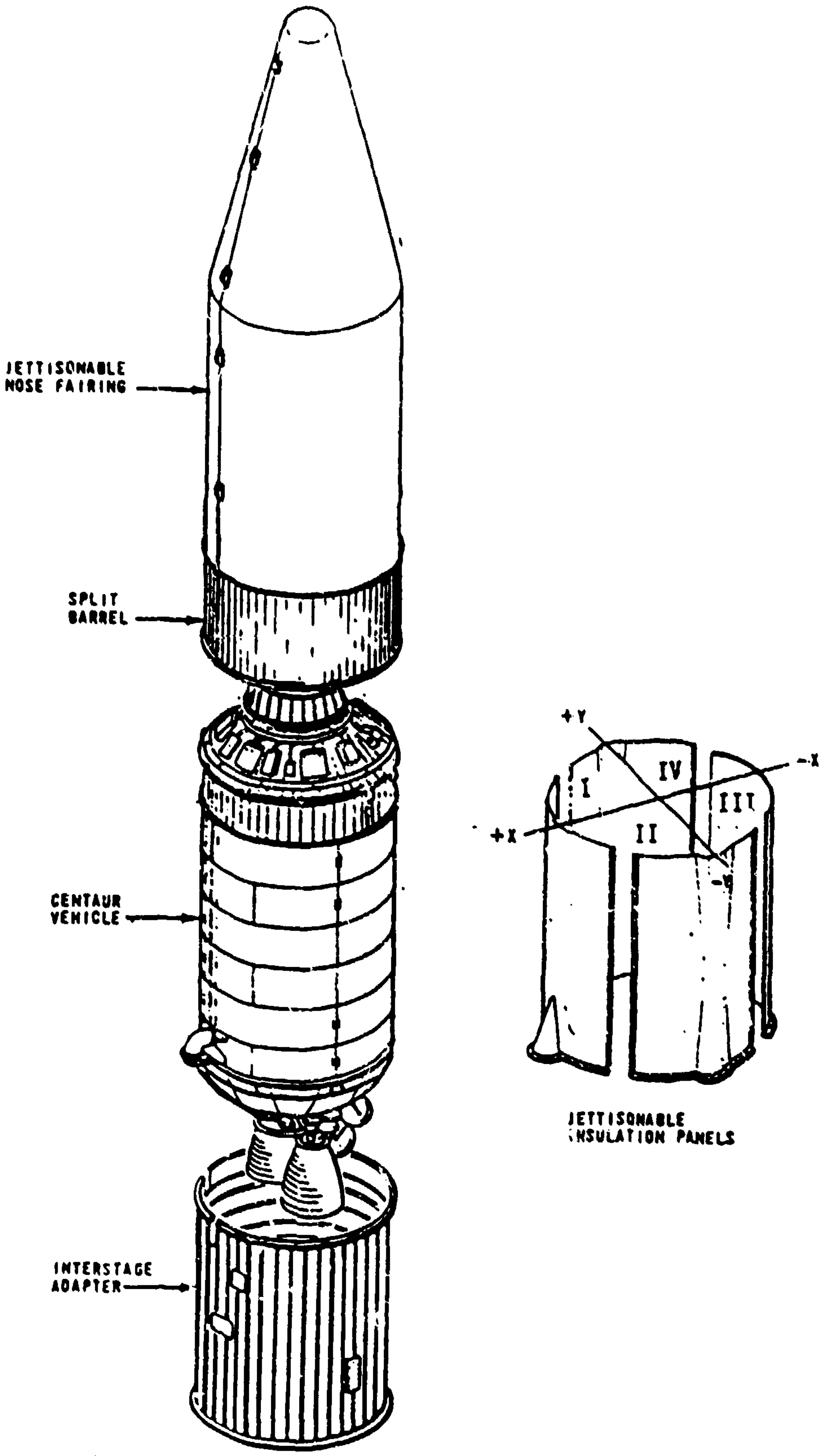


Figure C4-4 Centaur D-1AR Assembly

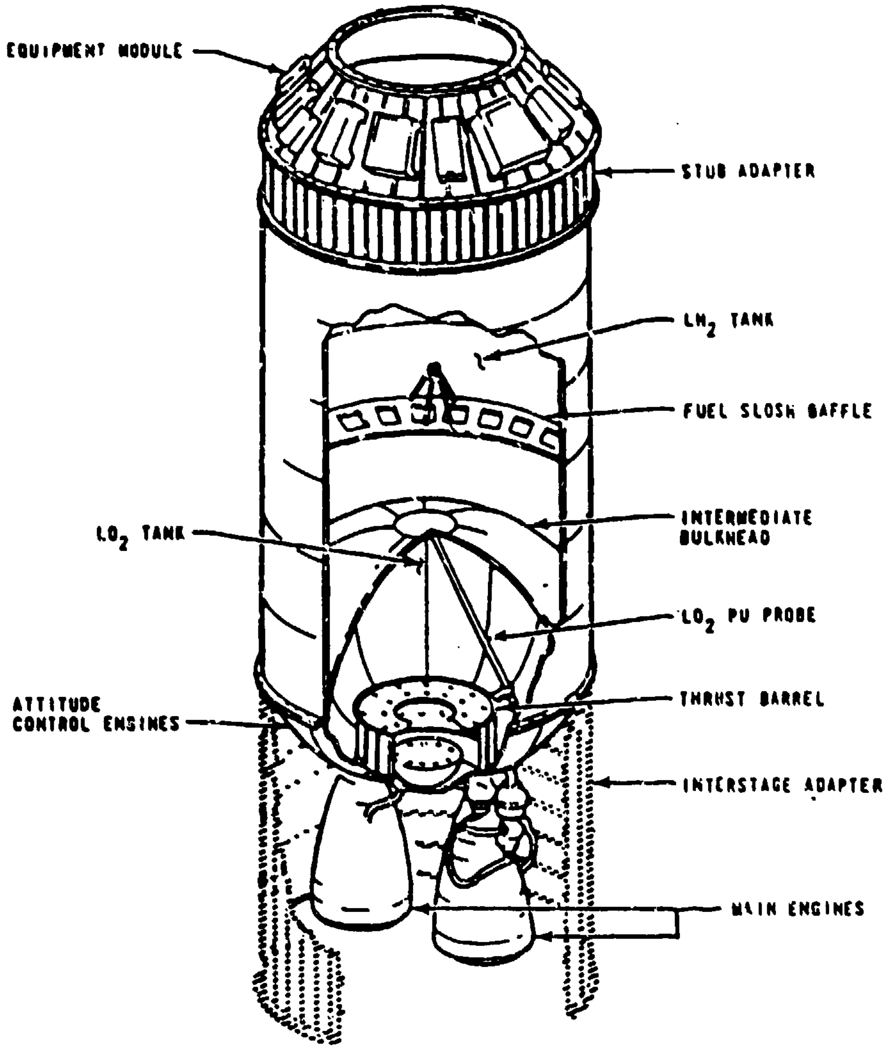


Figure C4-5 Centaur D-1AR Configuration

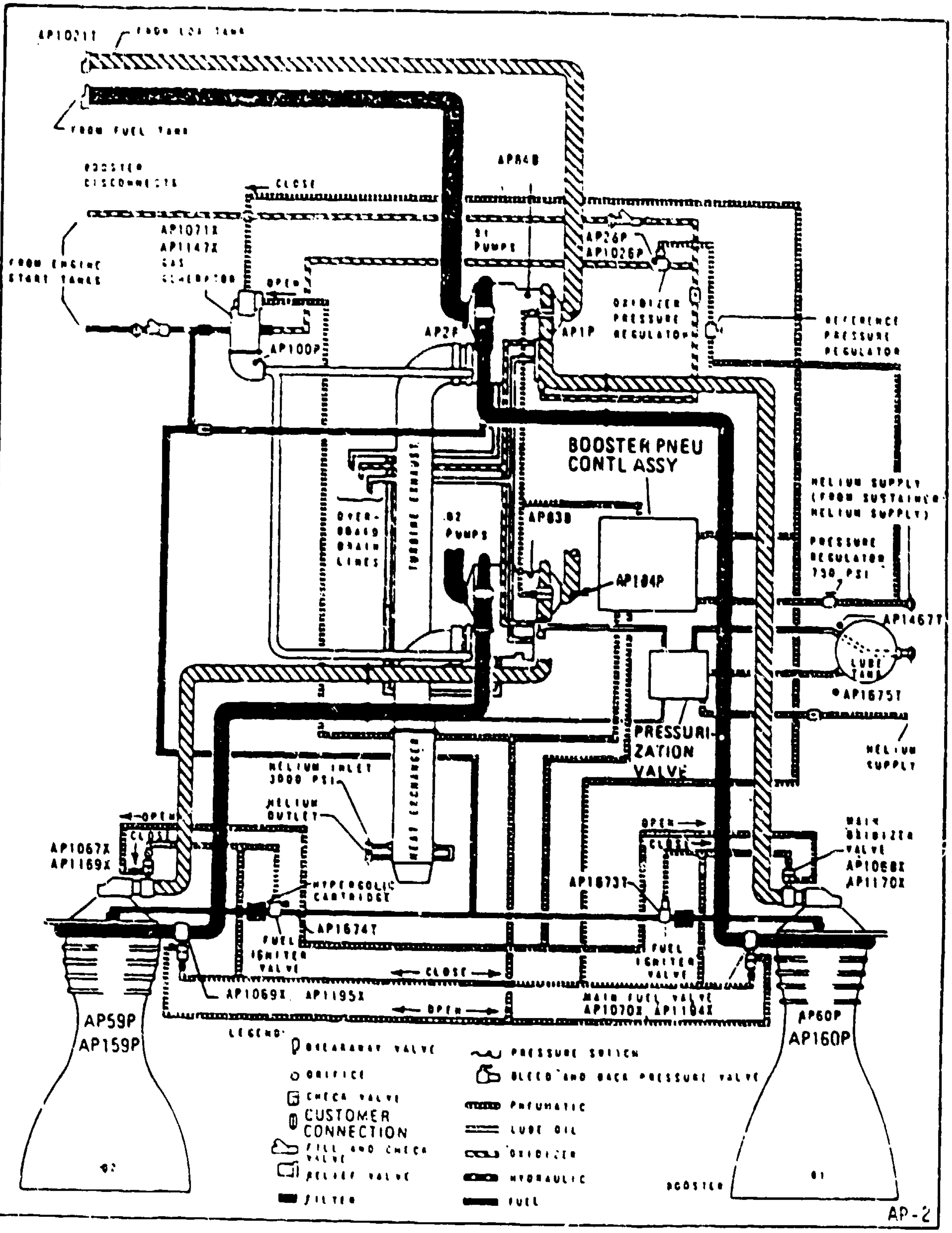


Figure C4-6 Atlas B1 and B2 Engine Schematic

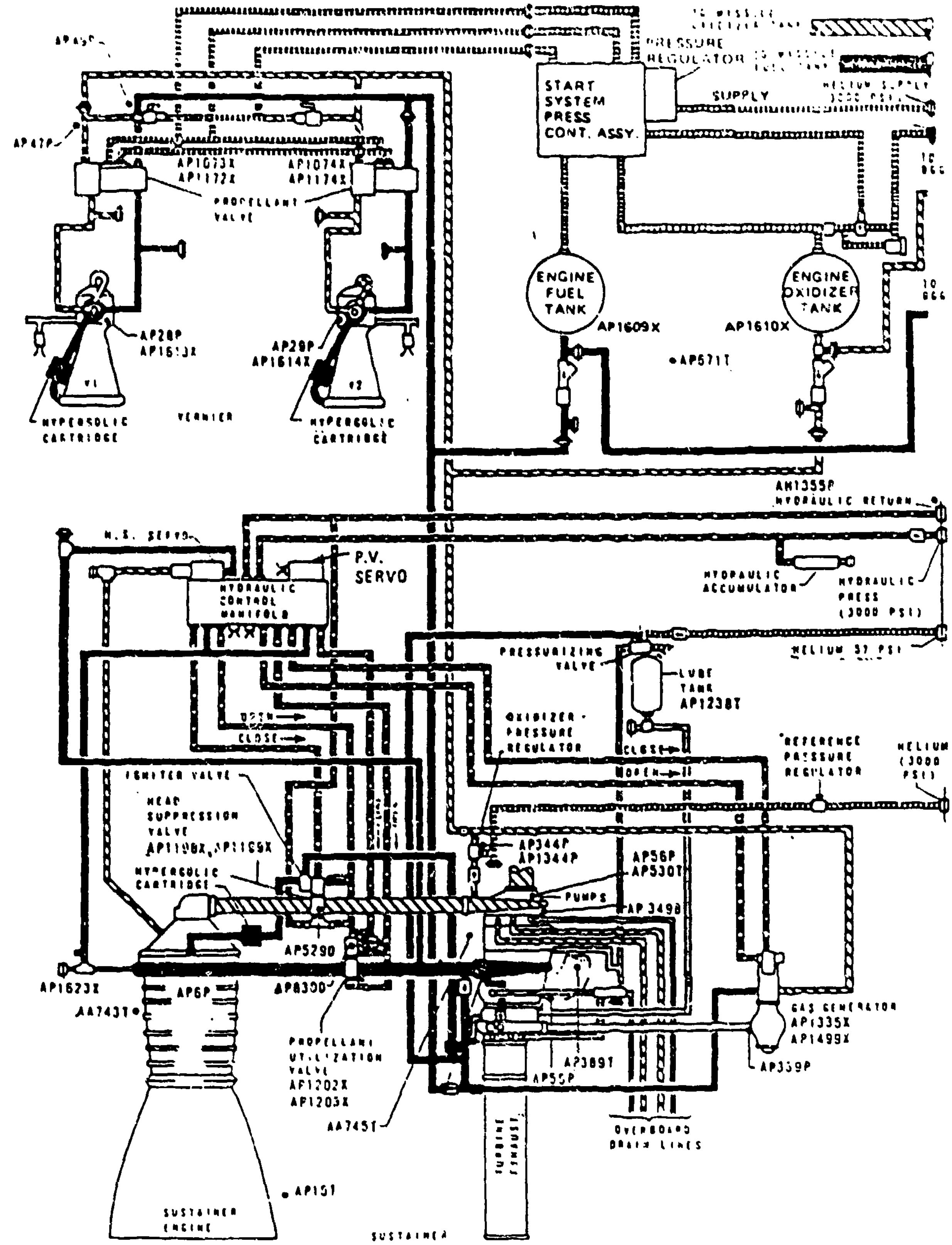


Figure C4-7 Atlas Sustainer and Vernier Engines Schematic

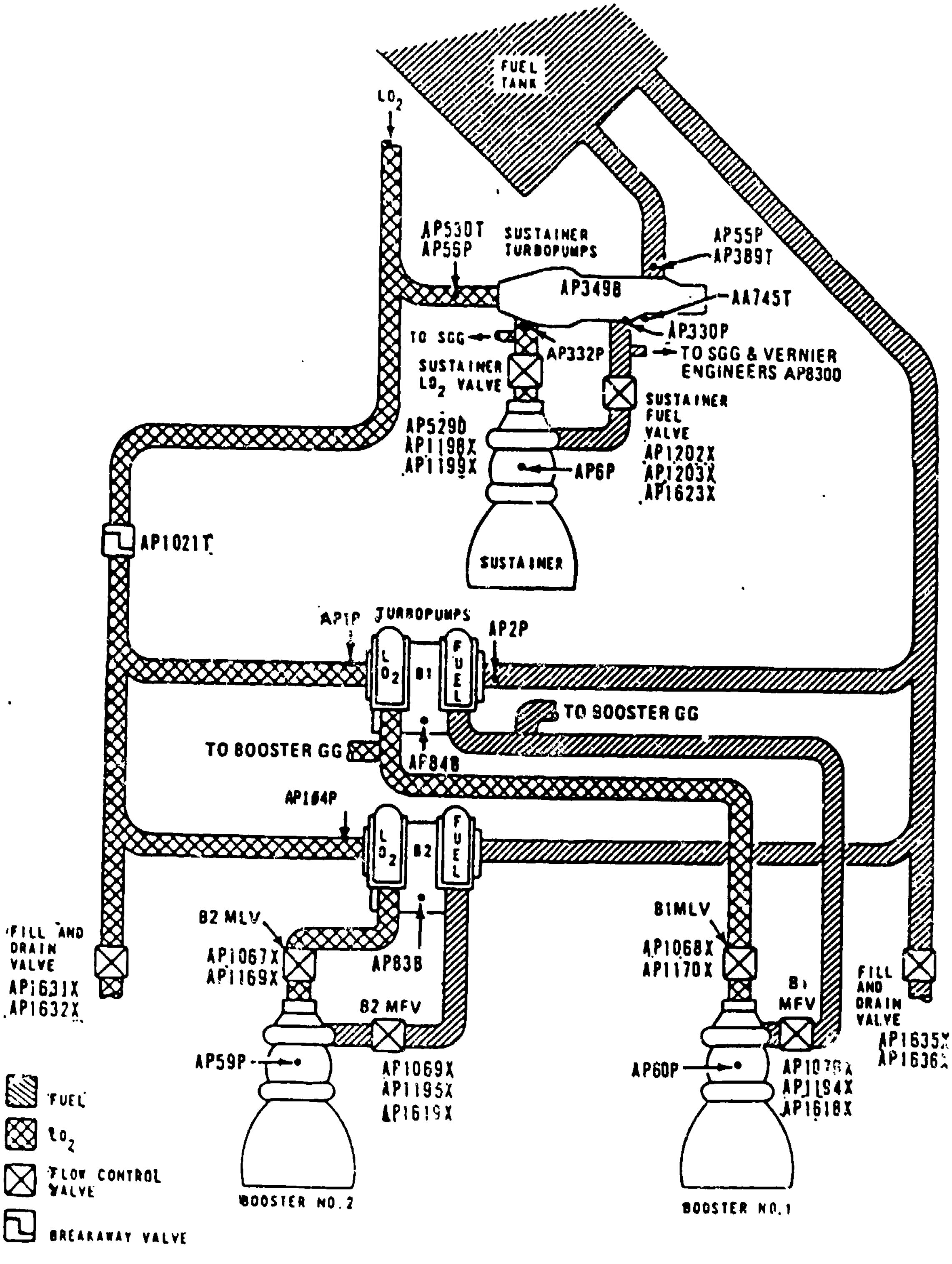


Figure C4-8 Atlas B1, B2, and Sustainer Engine Propellant Flow Schematic

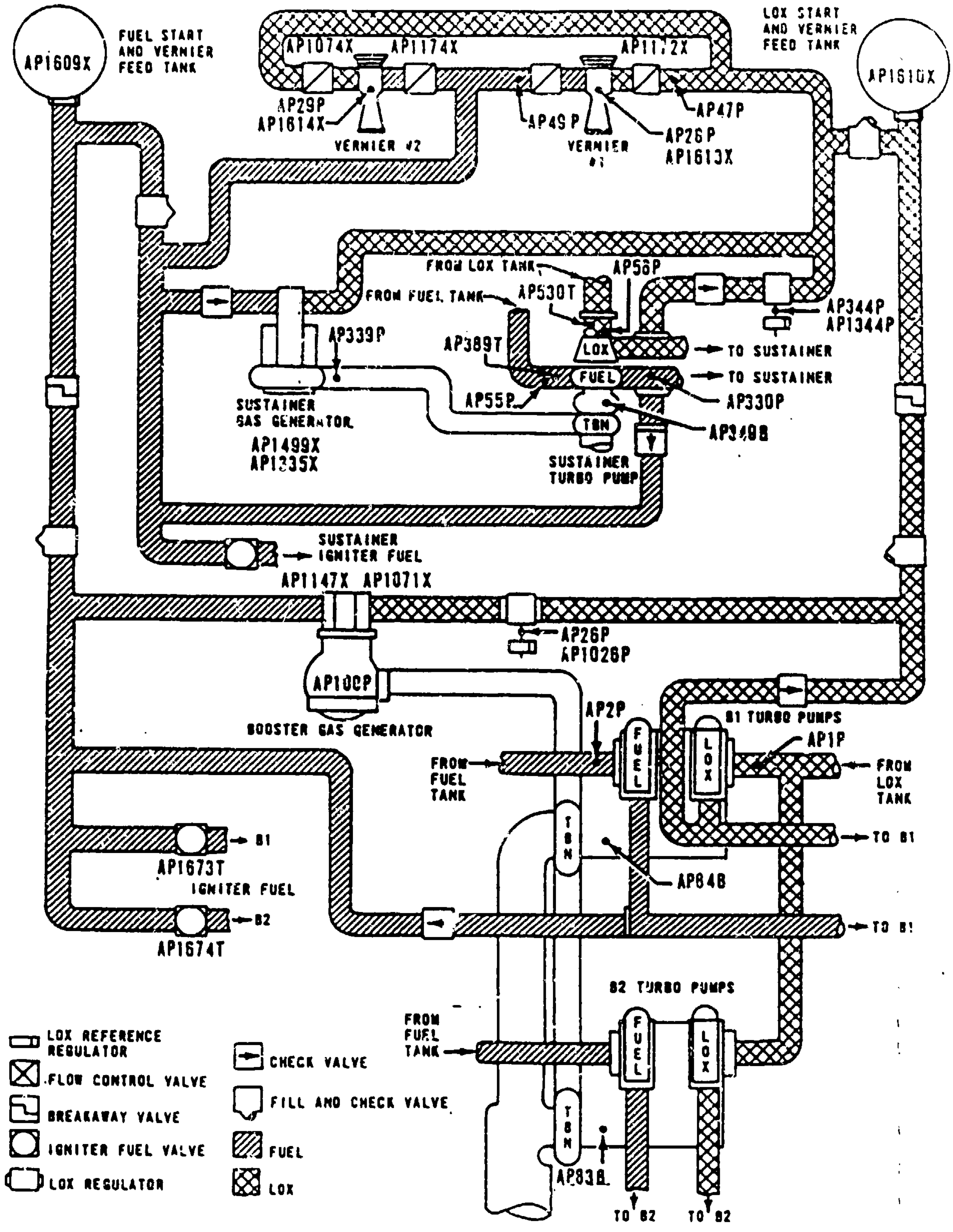


Figure C4-9 Atlas Propellant Feed Schematic

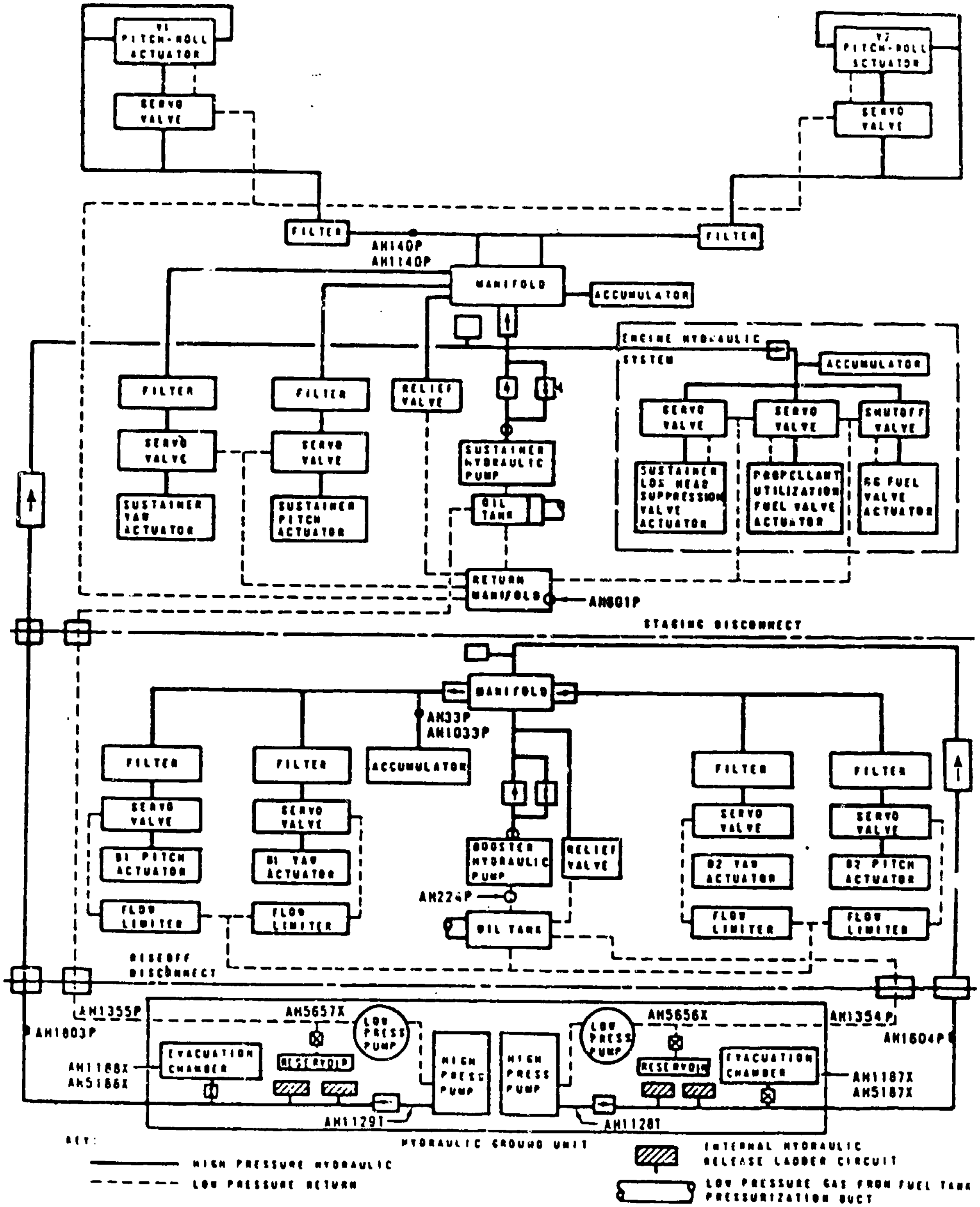


Figure C4-10 Atlas Hydraulic System Schematic

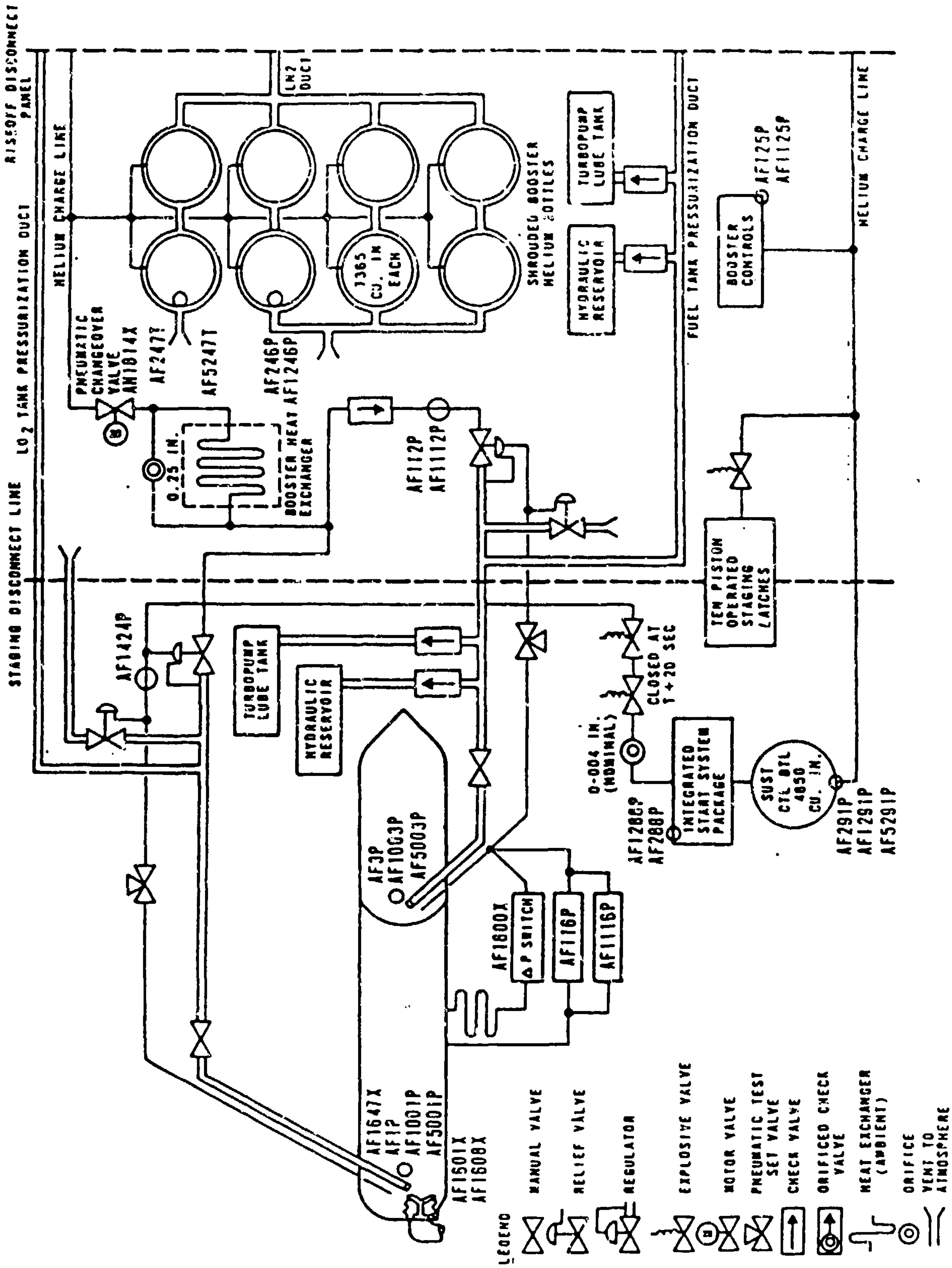


Figure C4-11 Atlas Pneumatic System

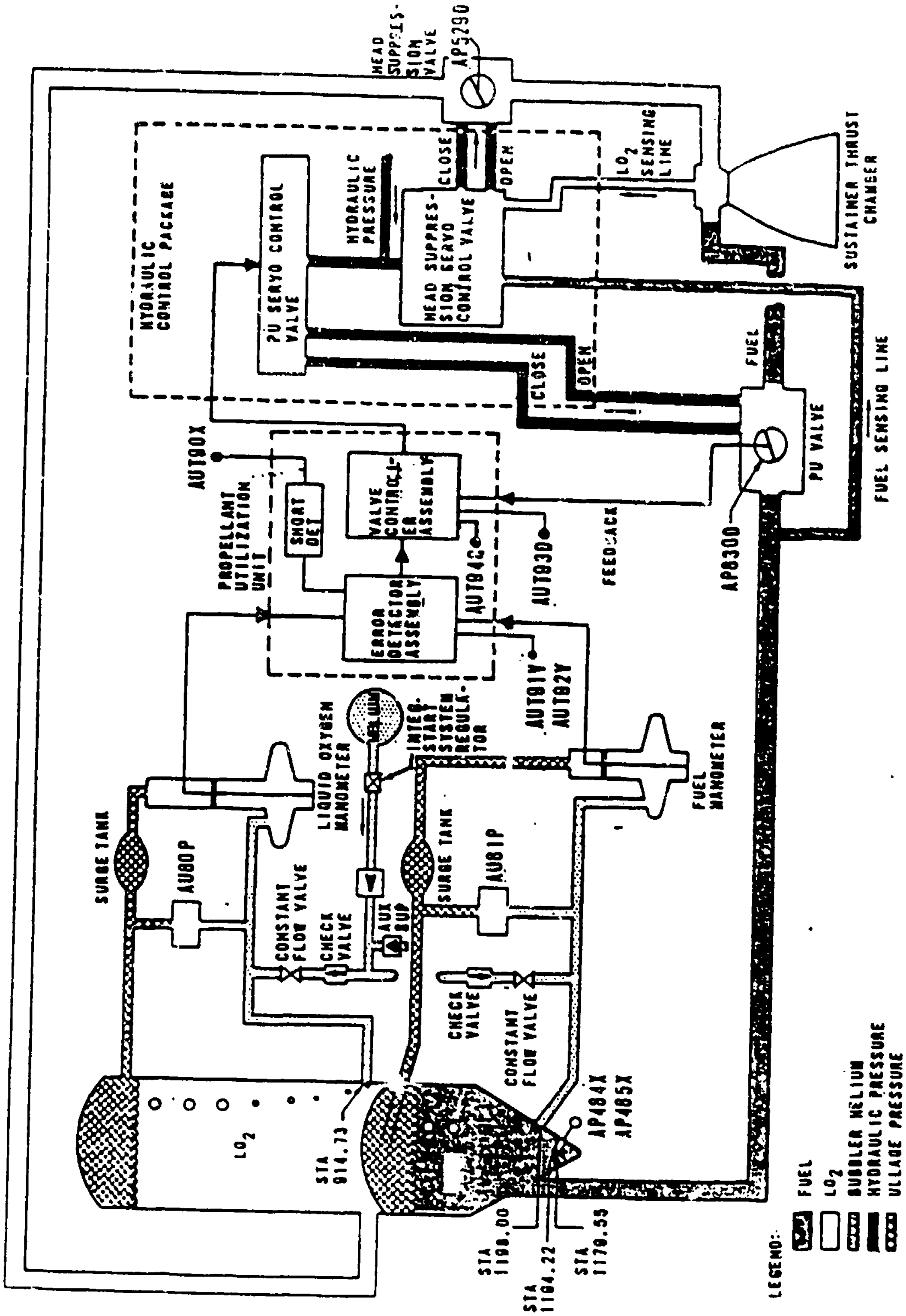


Figure C4-12 Atlas Propellant Utilization System

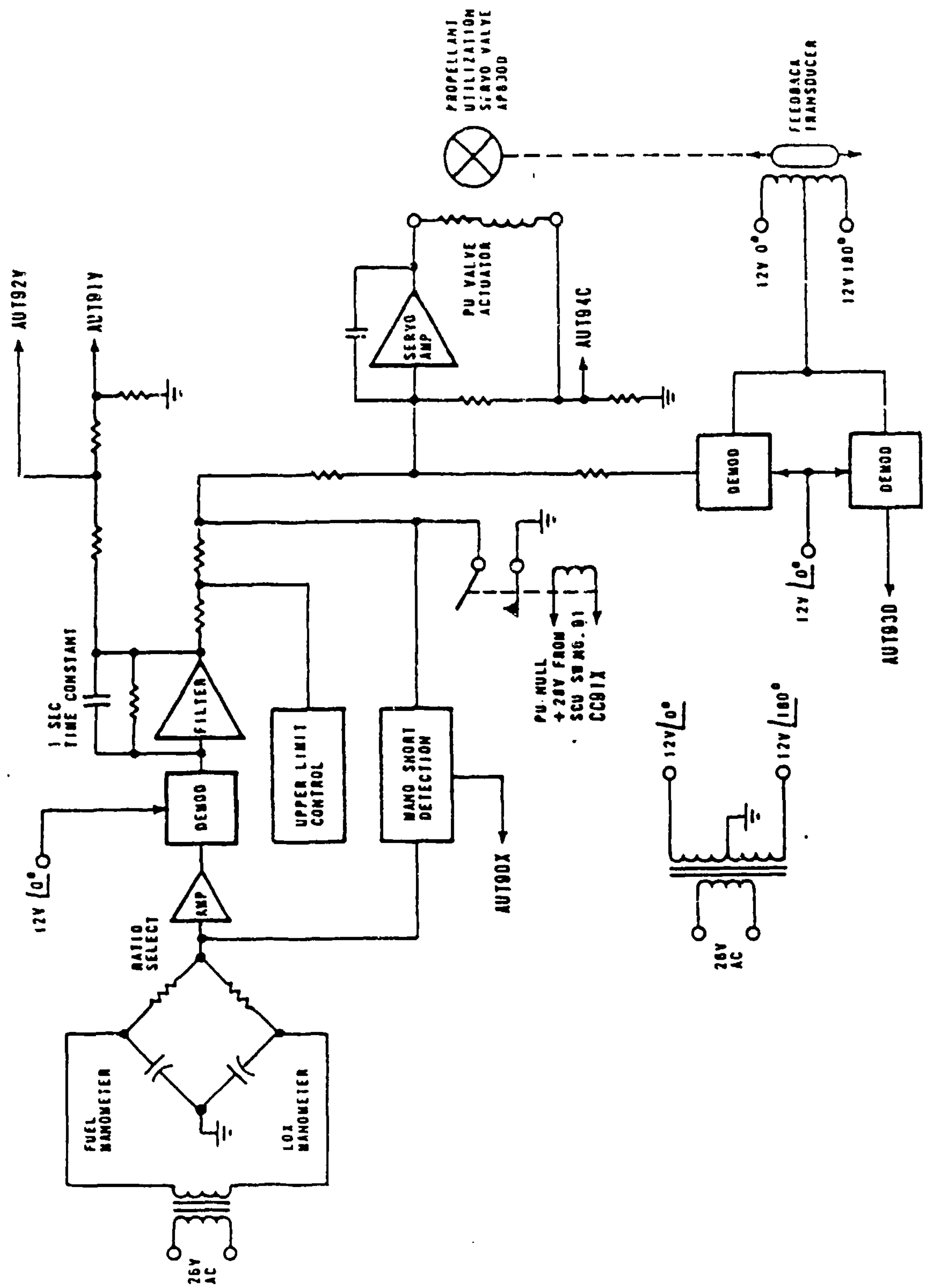


Figure C4-13 Atlas Propellant Utilization System Schematic (Simplified)

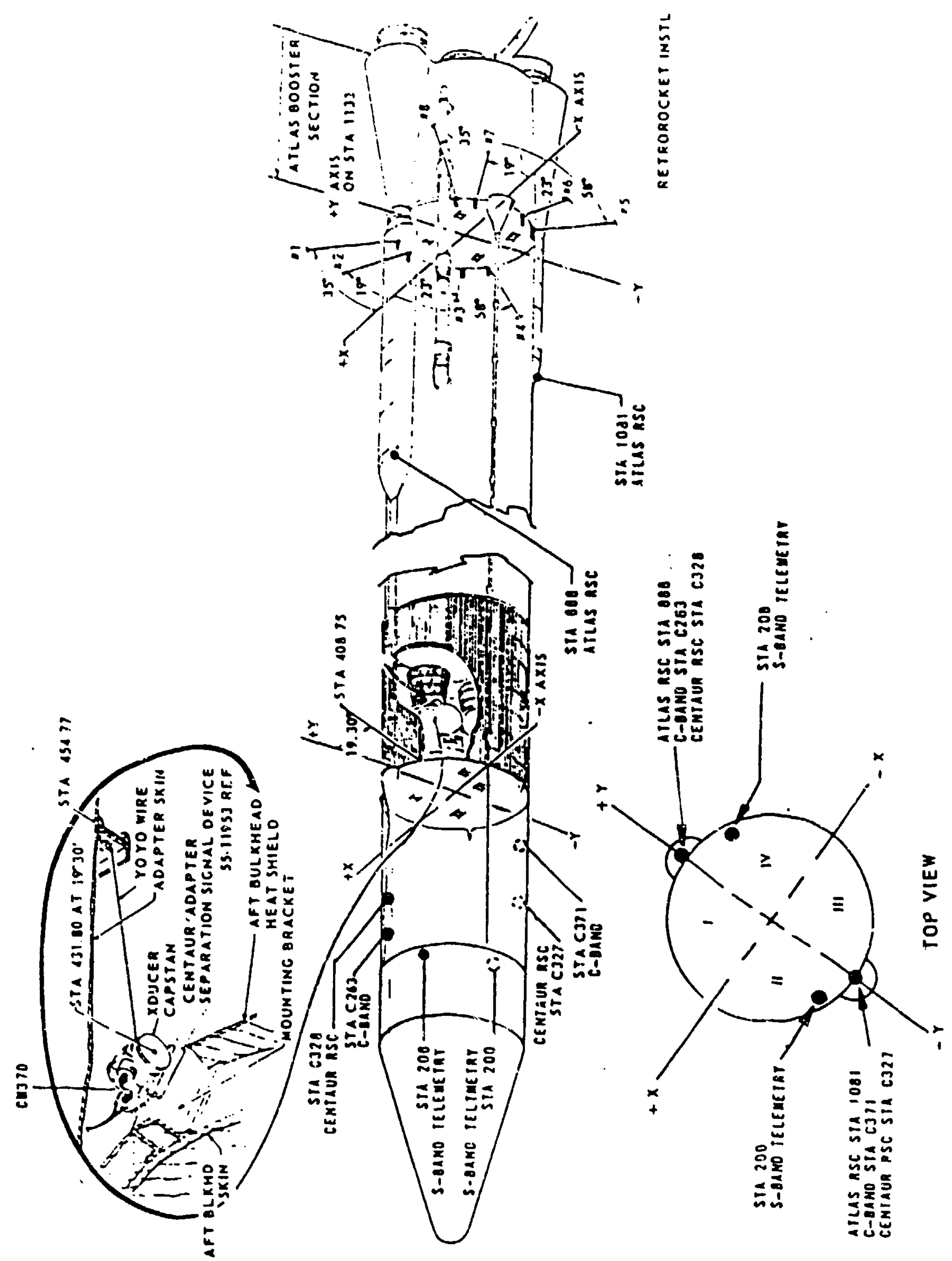


Figure C4-14 Atlas/Centaur Separation Instrumentation and Antenna Locations

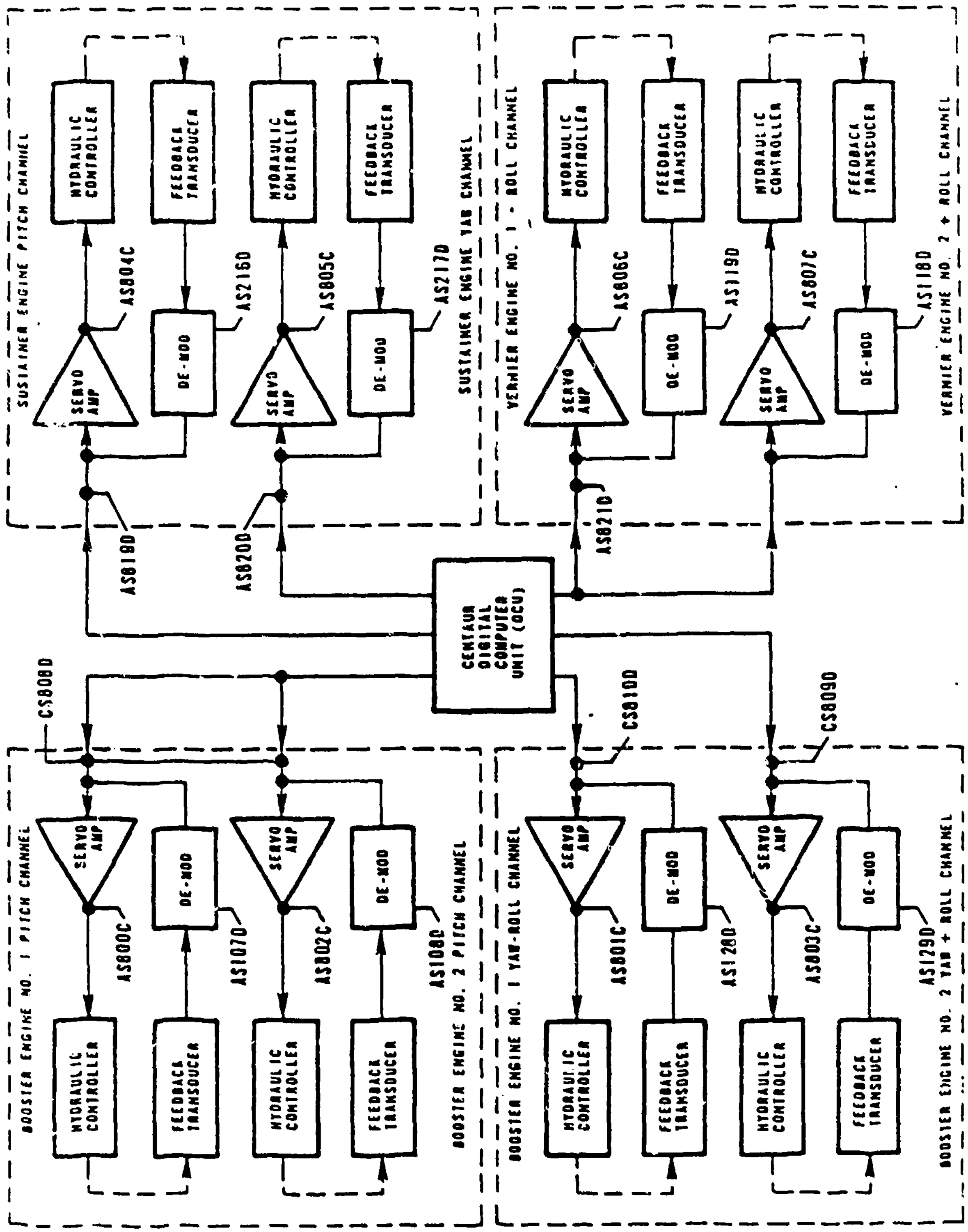


Figure C4-15 Atlas Flight Control System Block Diagram

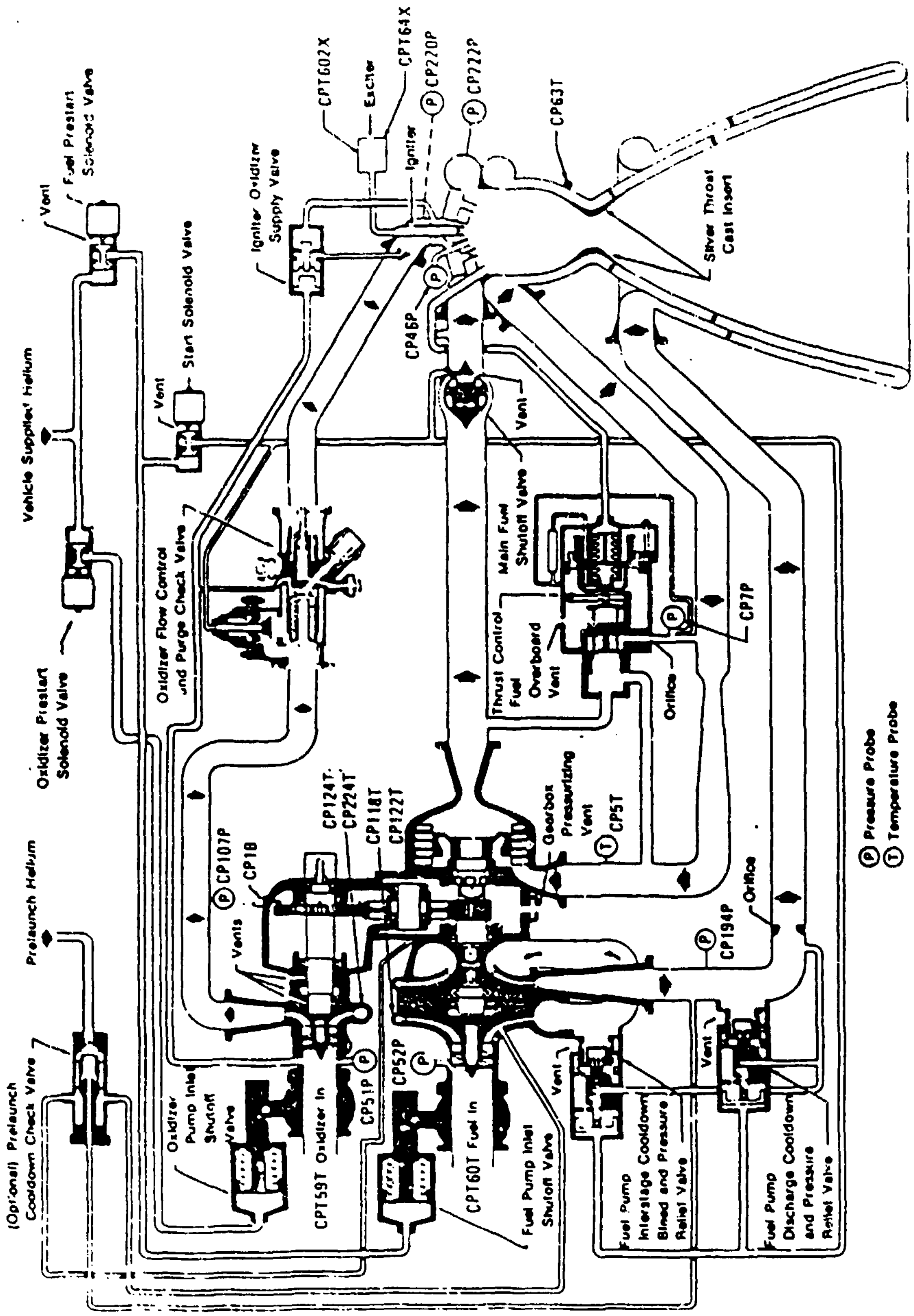


Figure C4-16 Centaur C-1 Engine Propellant Flcw

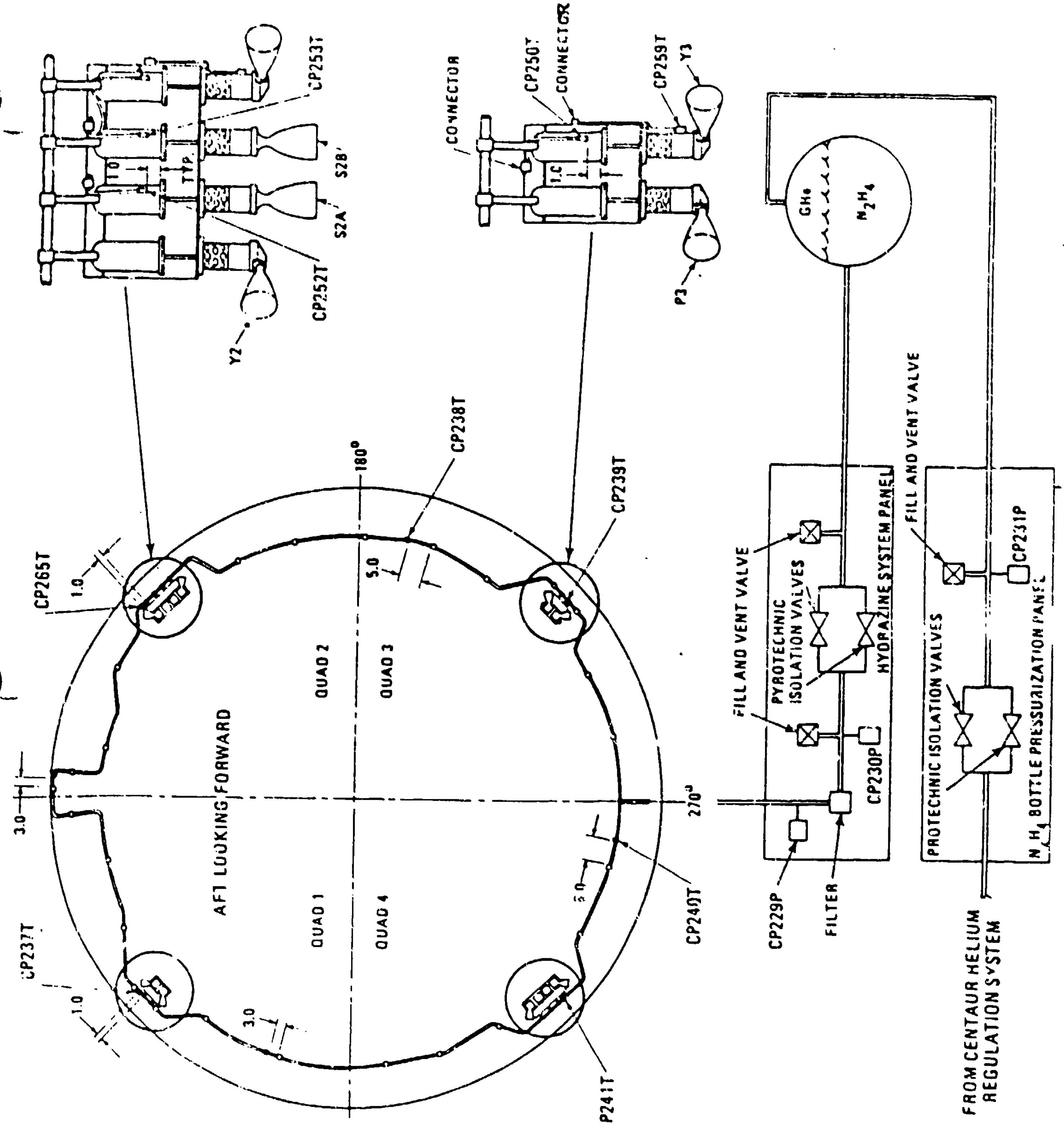


Figure C4-17 Centaur Hydrazine RCS

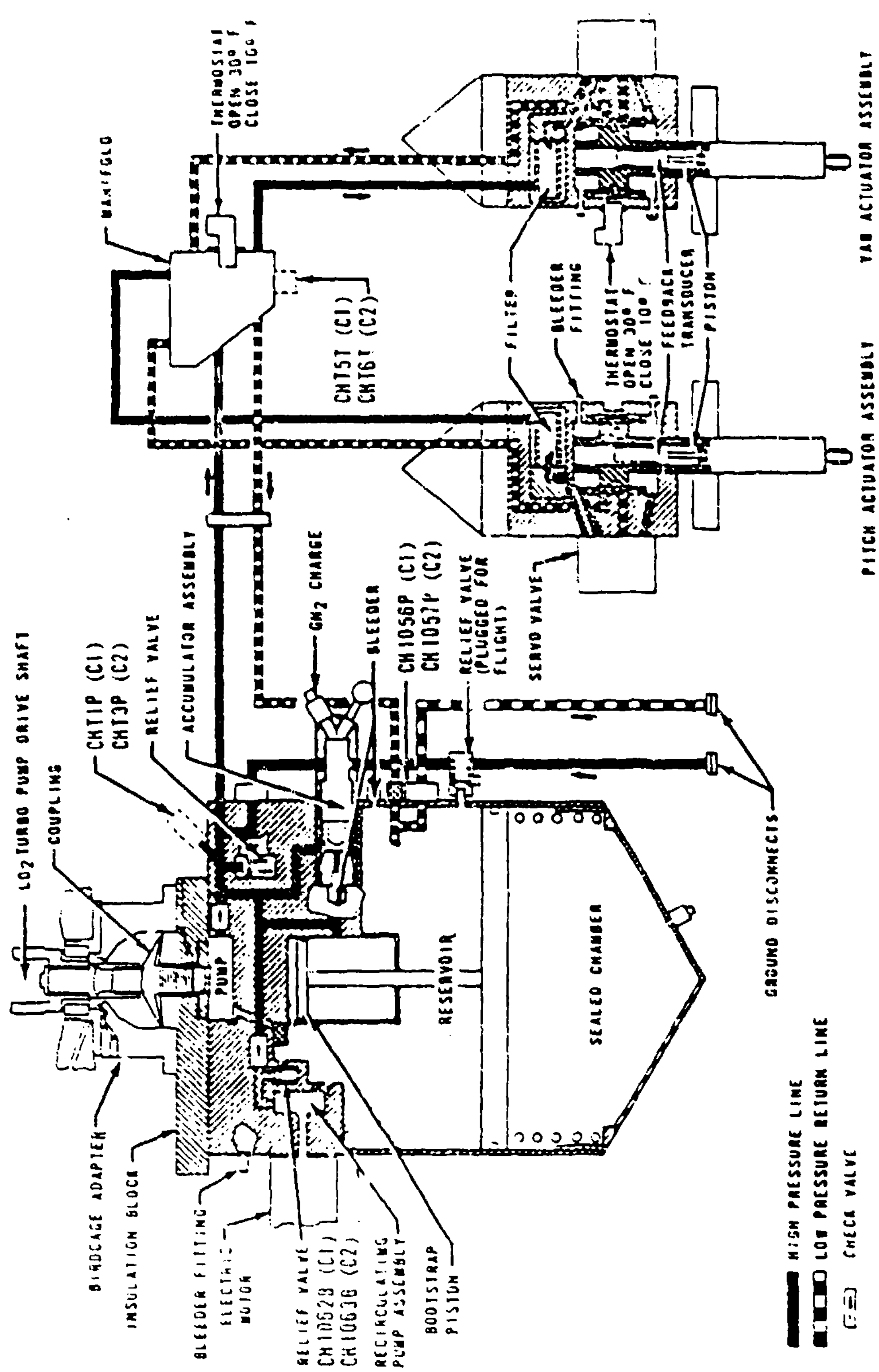


Figure C4-18 Centaur Hydraulics System Schematic

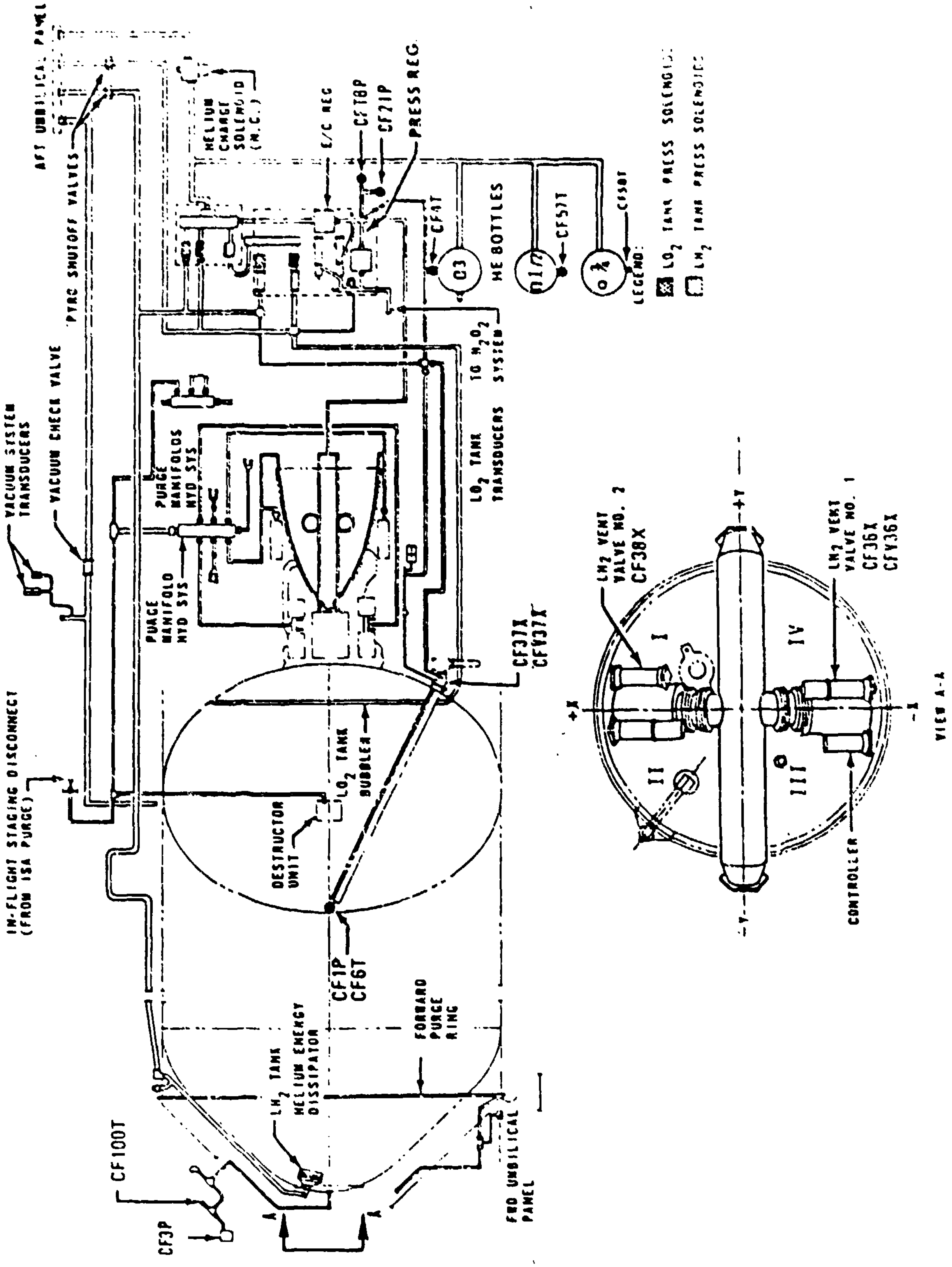


Figure C4-19 Centaur Pneumatic System

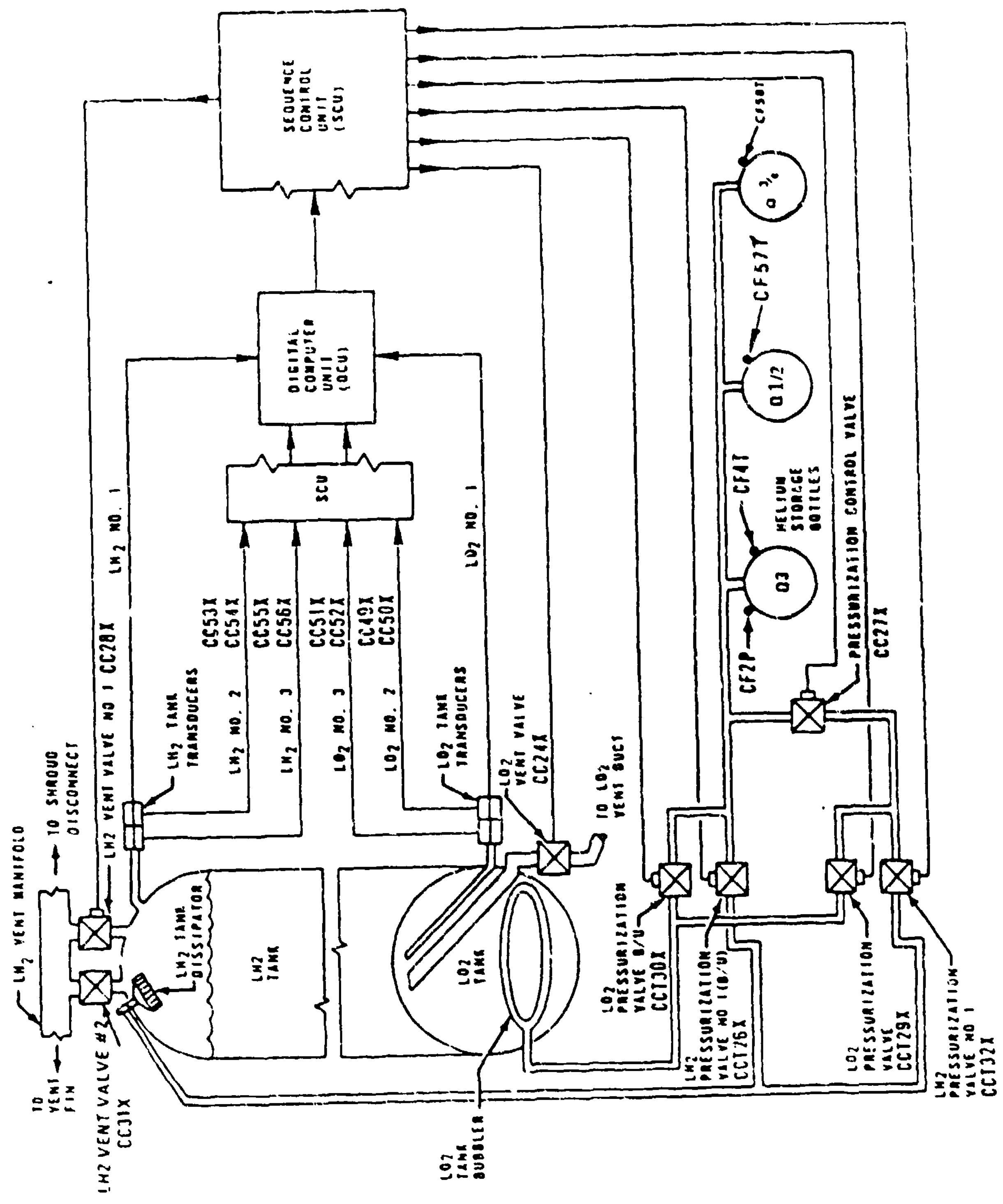


Figure C4-20 Centaur Computer-Control Venting and Pressurization Schematic

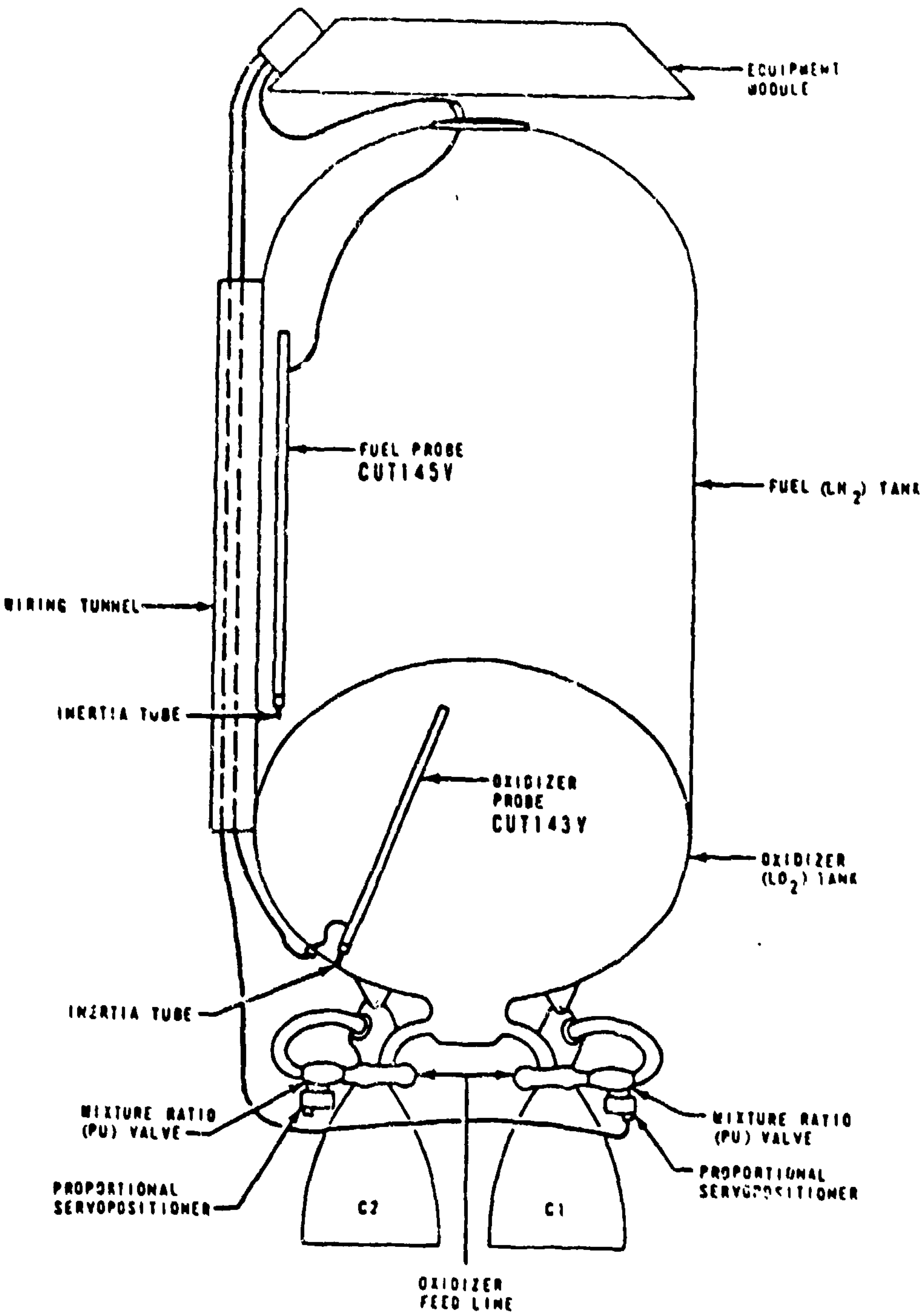


Figure C4-21 Centaur Propellant Utilization System

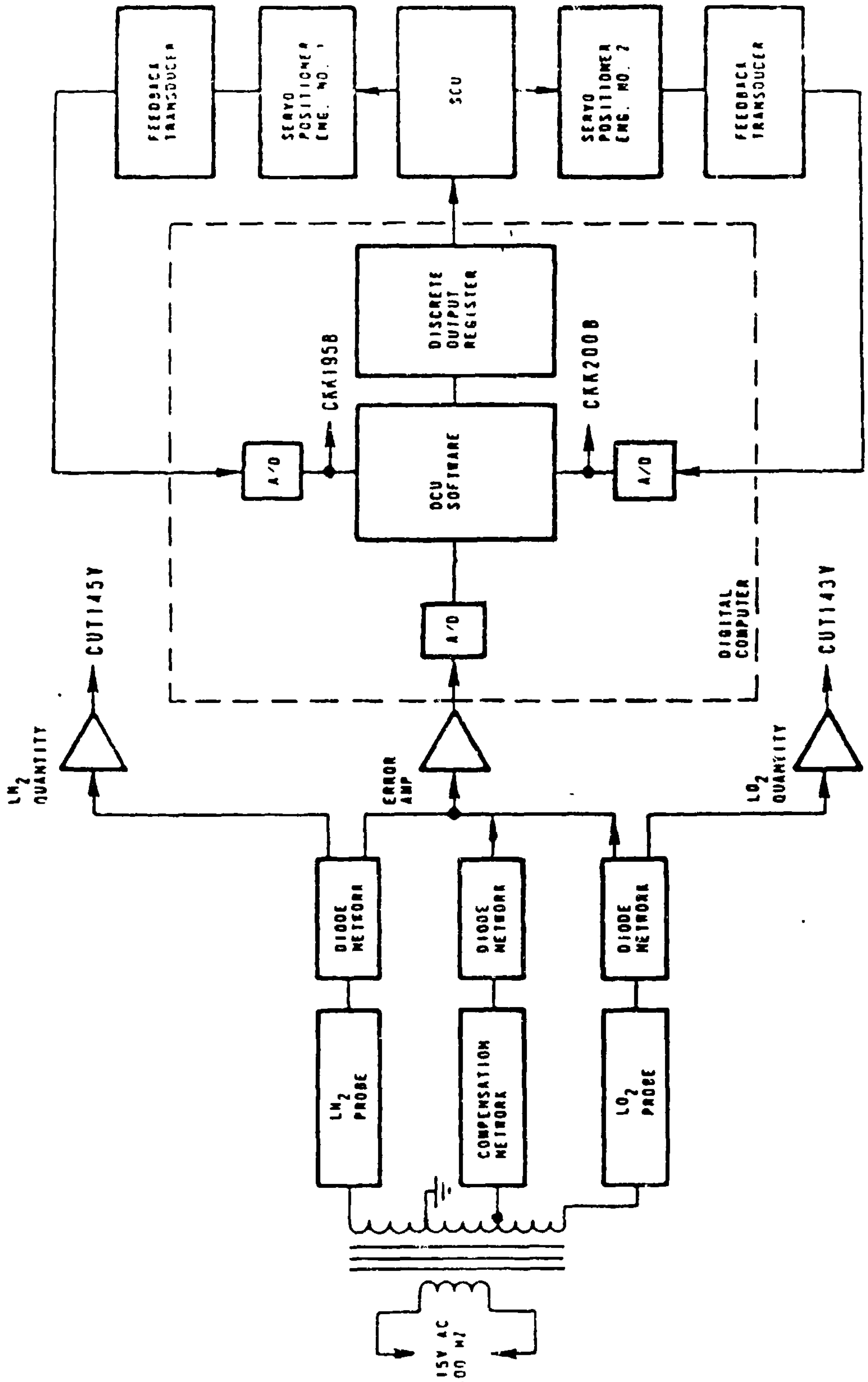


Figure C4-22 Centaur PU System Block Diagram

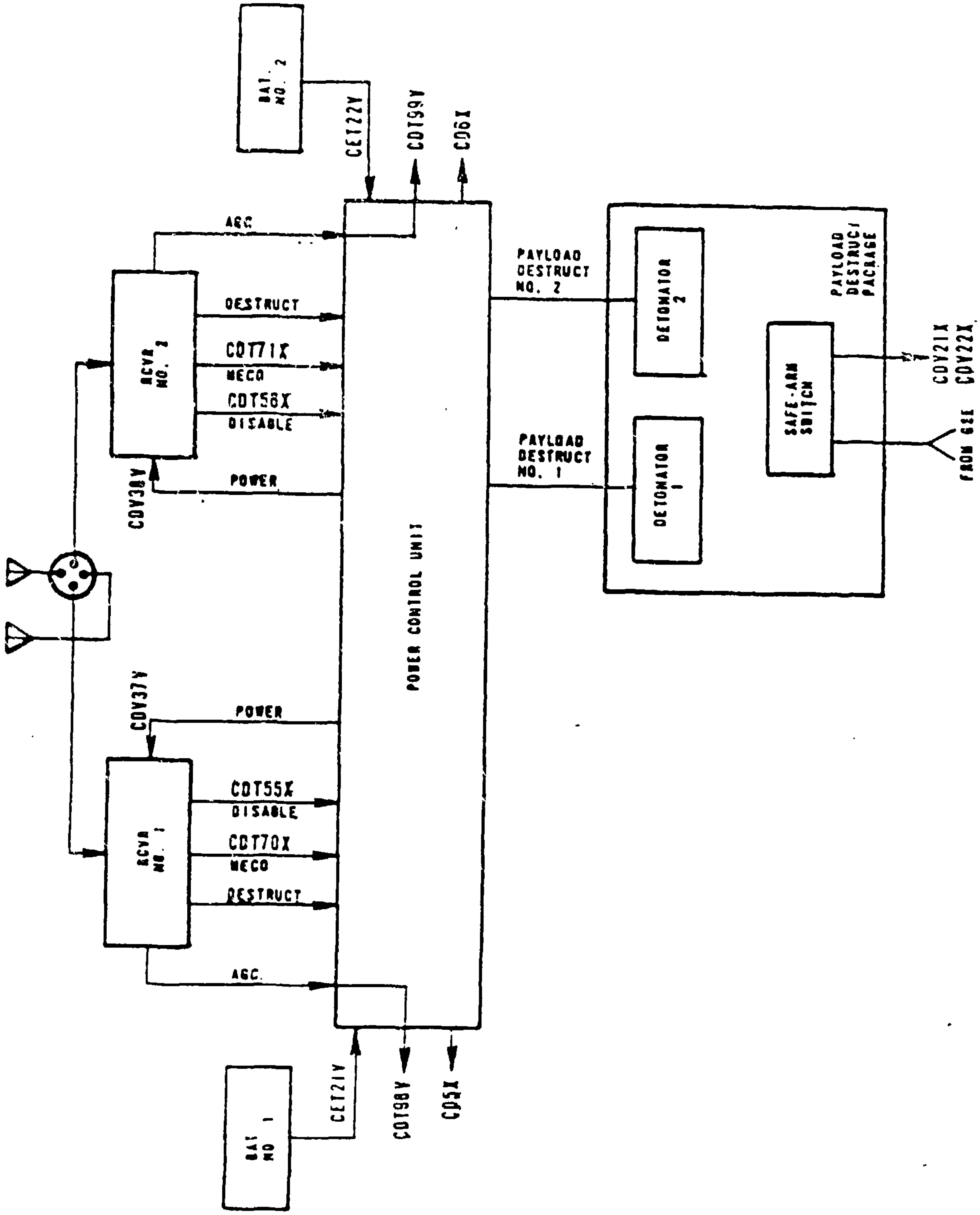


Figure C4-23 Centaur Range Safety System Block Diagram

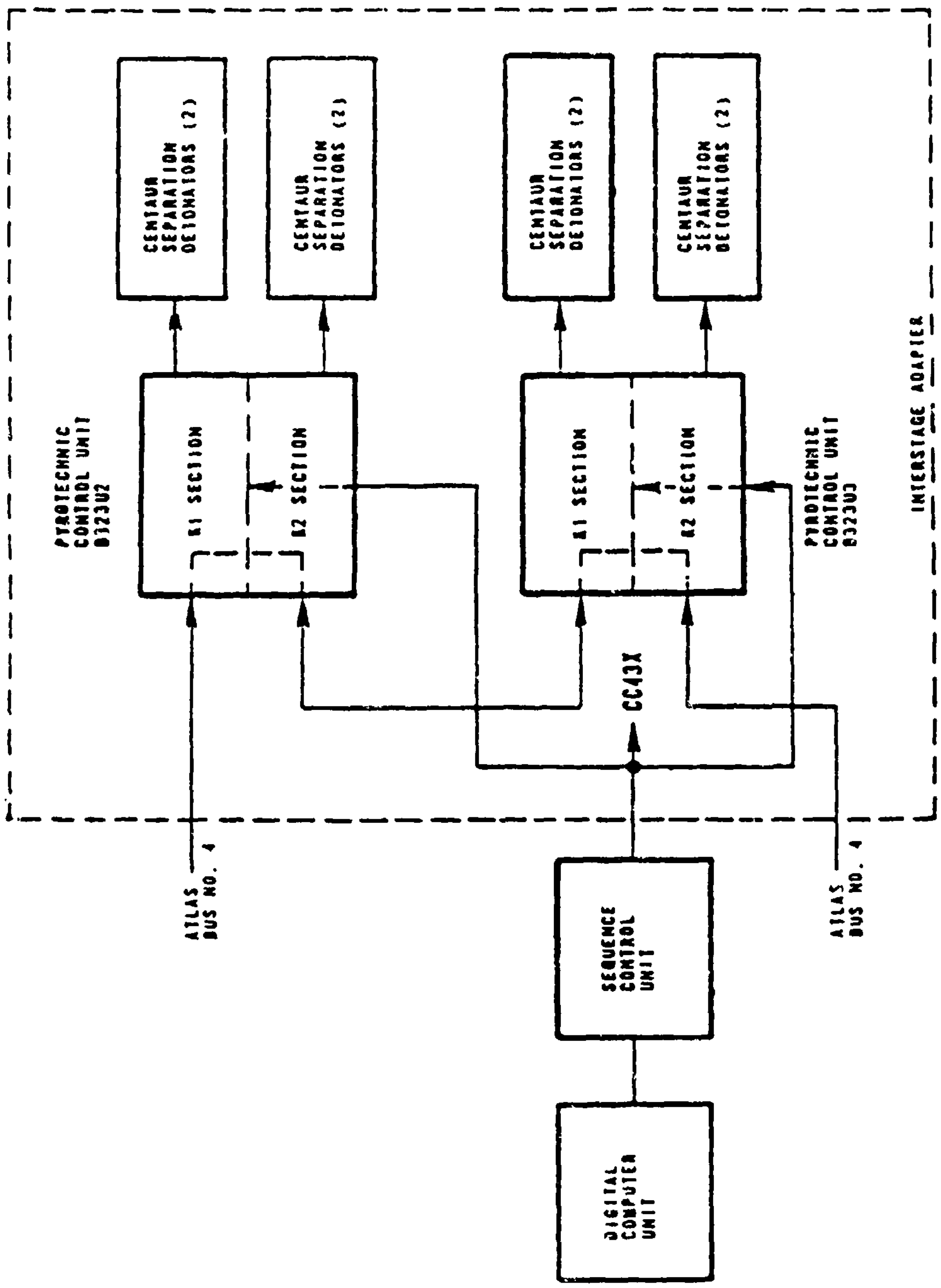
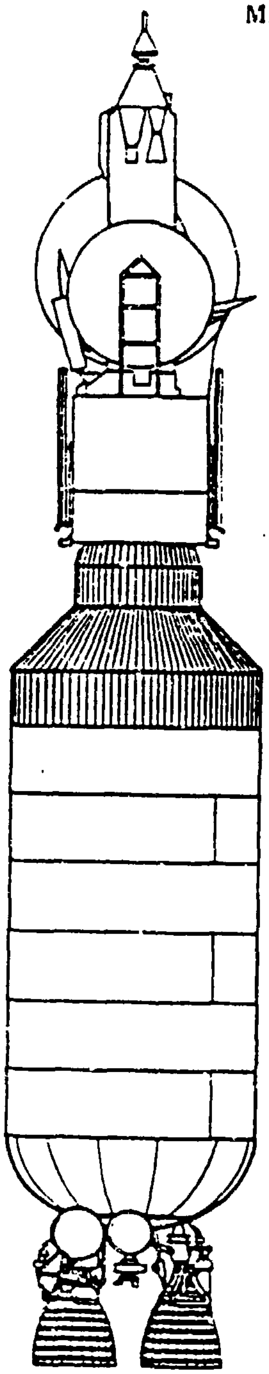


Figure C4-24 Atlas/Centaur Separation Pyrotechnic Block Diagram

MISSION SUMMARY



MISSION PARAMETERS	
MISSION DESIGNATION	INTELSAT VA (F10)
MISSION OBJECTIVE	PLACEMENT OF COMMERCIAL COMMUNICATION SATELLITES INTO GEOSTATIONARY ORBIT.
MISSION FINAL ORBIT	STATIONARY IN EARTH EQUATORIAL PLANE.
FINAL STATIONARY POSITION	TO BE SELECTED BY INTELSAT (DESIRED POSITIONING ACHIEVED BY COMBINED USE OF MULTIPLE REVOLUTIONS IN LAUNCH TRANSFER ORBIT PLUS POST-APOGEE-BURN DRIFT ORBIT).
LAUNCH PHASE PARAMETERS	
LAUNCH COMPLEX	ESMC COMPLEX 36B
LAUNCH MODE	PARKING ORBIT ASCENT (TWO BURN)
LAUNCH AZIMUTH	103.0 DEGREES
CENTAUR PARKING ORBIT	
PERIGEE/APOGEE ALTITUDE	80/657 NAUTICAL MILES
ORBIT INCLINATION	28.5 DEGREES
COAST TIME IN ORBIT	12.7 MINUTES
CENTAUR SECOND BURN	
LOCATION OF BURN	FIRST EQUATORIAL CROSSING
BURNOUT ALTITUDE (MECO2)	187 NAUTICAL MILES
SPACECRAFT TRANSFER ORBIT	
AT SPACECRAFT SEPARATION	
PERIGEE ALTITUDE	168 NAUTICAL MILES
ORBIT INCLINATION	23.36 DEGREES
ARGUMENT OF PERIGEE	178.40 DEGREES
AT FIRST APOGEE	
APOGEE ALTITUDE	19,324 NAUTICAL MILES
ORBIT PERIOD	633 MINUTES
SPACECRAFT APOGEE BURN	
LOCATION OF BURN AND BURNOUT LONGITUDE	INTELSAT WILL COMMAND S/C APOGEE BURN VIA RF LINK AT ONE OF THE TRANSFER ORBIT APOGEE OCCURRENCES (YET TO BE SELECTED).
SPACECRAFT FINAL ORBIT	
PERIGEE/APOGEE ALTITUDE	19,324/19,324 NAUTICAL MILES
ORBIT INCLINATION	0 DEGREES
ORBIT PERIOD	23.935 HOURS

* NOMINAL PARAMETERS FOR EARTH STATIONARY ORBIT. ACTUAL INTELSAT V/VA SPACECRAFT FINAL ORBITS MAY HAVE SLIGHT VARIATION IN ALTITUDE AND/OR INCLINATION ANGLES.

Figure C4-25 Mission Summary

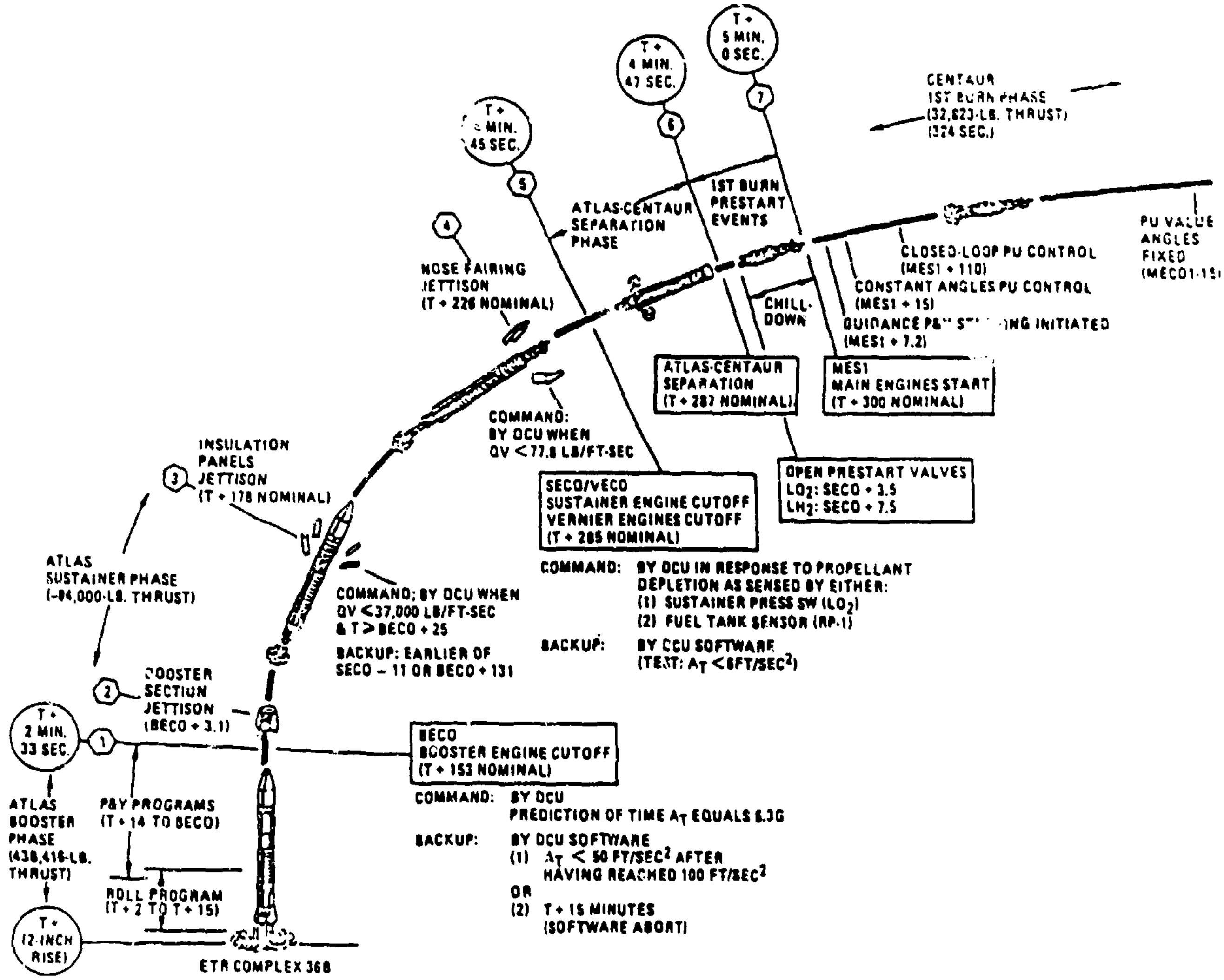


Figure C4-26 Atlas/Centaur-62/63 INTELSAT-V Flight Events Profile (Sheet 1 of 4)

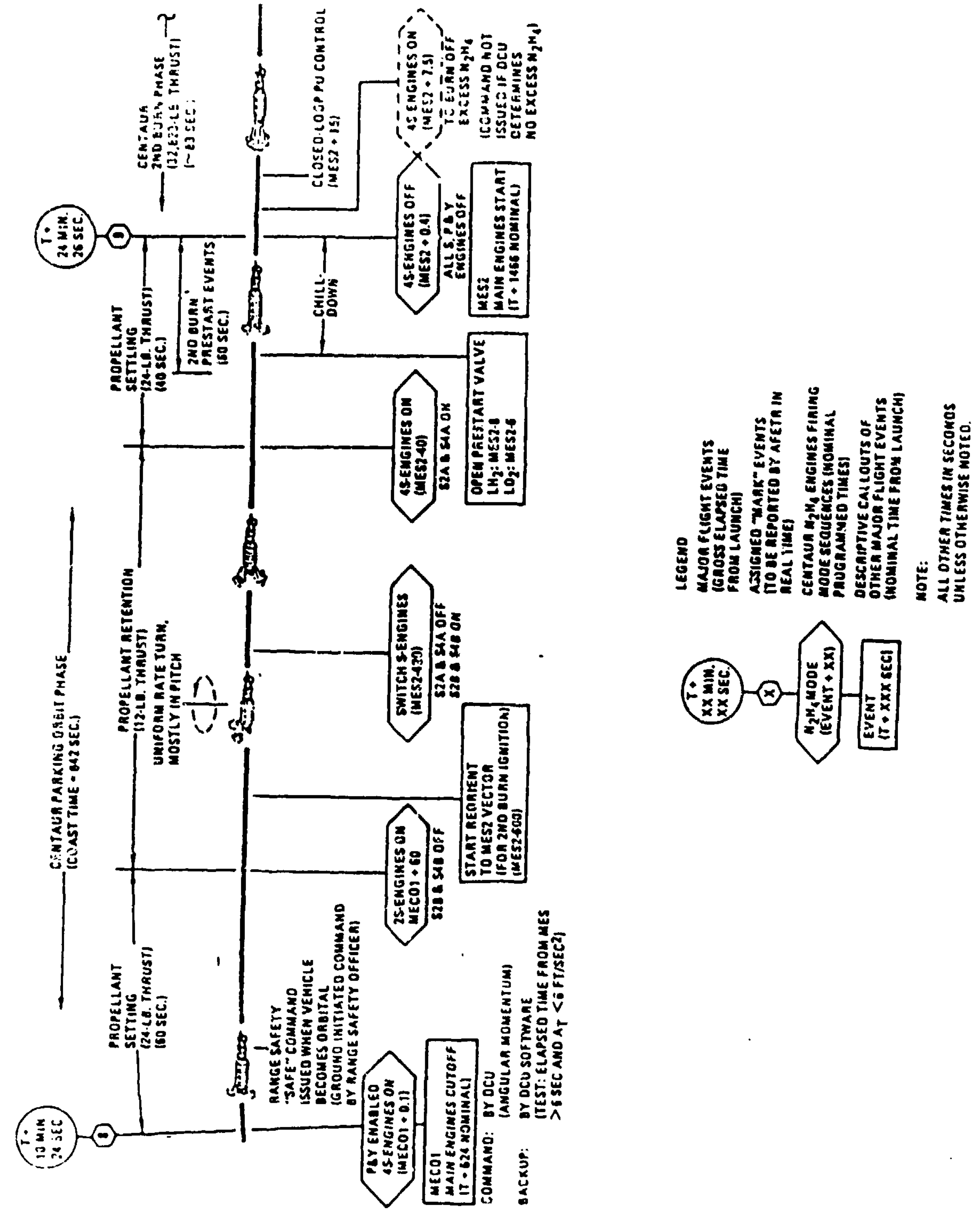


Figure C4-26 Flight Events Profile (Sheet 2 of 4)

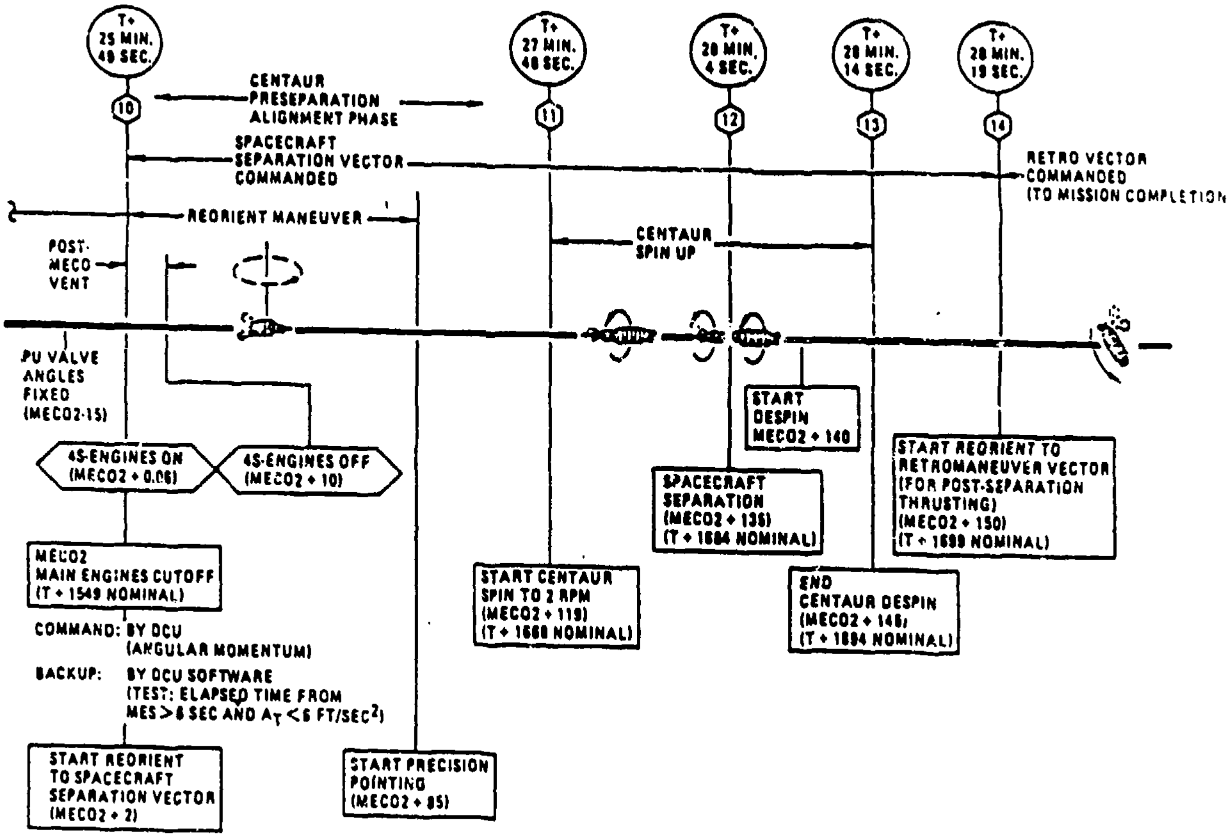


Figure C4-26 Flight Events Profile (Sheet 3 of 4)

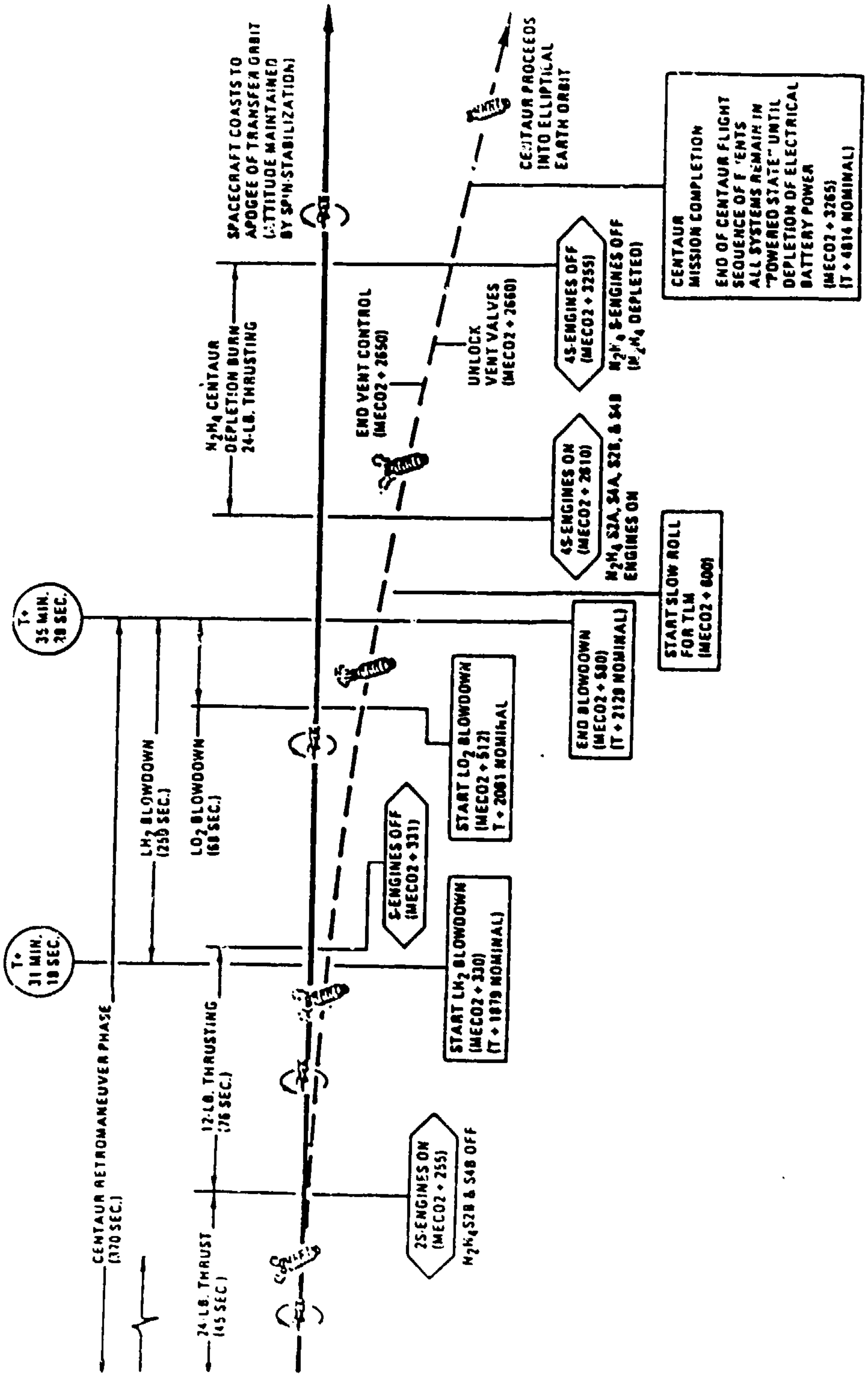


Figure C4-26 Flight Events Profile (Sheet 4 of 4)

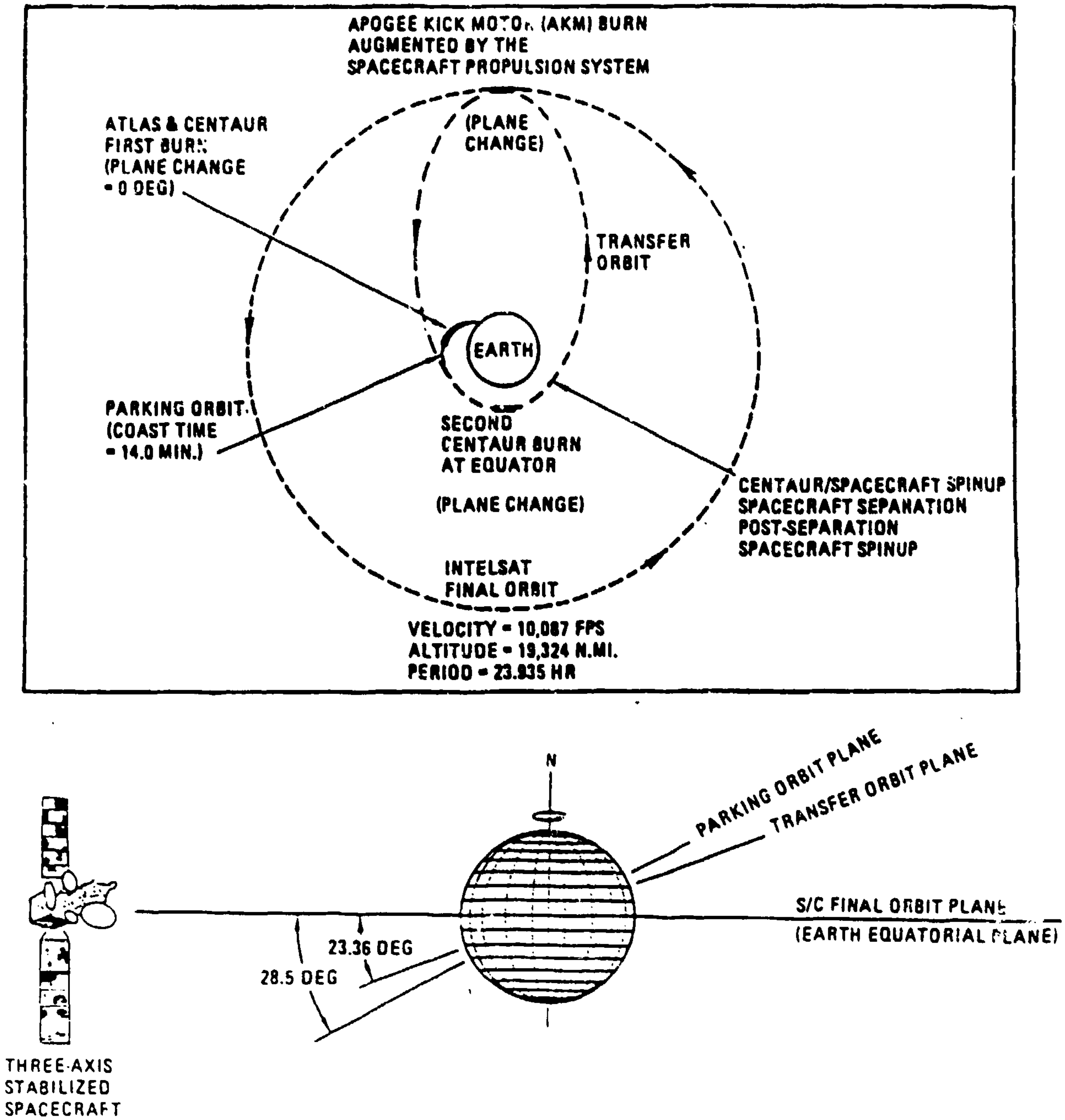


Figure C4-27 INTELSAT VA Mission Trajectory Profile

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Delta

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APPENDIX C5
DELTA

C5.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref 241.

C5.1 GENERAL DESCRIPTION

The Delta Launch Vehicle was developed for the National Aeronautics and Space Administration (NASA) by the McDonnell Douglas Astronautics Company (MDAC) and has been actively utilized since 1960 for launching spacecraft into Earth, lunar, and solar orbits. Operations are conducted from the Eastern Space and Missile Center (ESMC) and the Western Space and Missile Center (WSMC). The Delta Program is managed for NASA by the Delta Project Office located at the NASA Goddard Space Flight Center (GSFC). The Delta Project Office has overall responsibility for design, procurement, and launch of the Delta Launch Vehicle. The NASA Kennedy Space Center (KSC) is responsible for launchsite operations. MDAC is the prime contractor for launch vehicle design, manufacturing, and checkout at their plant in Huntington Beach, California, and for launch services at the launchsites.

By continuing modification or replacement of the respective stages, the Delta vehicle performance capability has been periodically upgraded. The growth history is depicted in Figure C5-1 for the evolution of vehicle configurations through the 3920/PAM configuration for geosynchronous transfer orbit useful load (which includes the payload attach fitting).

The Delta Launch Vehicle DSV-3P-11E/F is a vertically launched space vehicle. The first stage consists of a liquid propellant booster with nine Thiokol Corporation Castor IV solid propellant rocket motors attached for thrust augmentation. The second stage consists of a liquid propellant propulsion system. The third stage consists of a Thiokol Corporation TEM-M-364-3, -4, or -14 solid propellant rocket motor. Figure C5-2 depicts this configuration, showing a PAM third-stage and attach fitting and the Aerojet AJ10-118K second-stage engine.

C5.2 DESCRIPTION OF SAFETY CRITICAL SUBSYSTEMS, HAZARDOUS MATERIALS, SCHEMATICS

C5.2.1 Liquid Propellant Systems (First Stage)

The first-stage propulsion system consists of a liquid bipropellant (LOX and kerosene fuel) main engine, two liquid bipropellant (LOX and kerosene fuel) vernier engines, nine solid propellant thrust augmentation motors, associated tanks, interconnecting lines, and controls. The main engine provides prime vehicle thrust and control during the first-stage powered flight with thrust of about 205,000 pounds at liftoff. Six of the nine solid propellant thrust augmentation motors are ignited at liftoff, and the other three are ignited after the six ground-start motors have burned out. Total thrust is approximately 625,000 pounds with six solid-propellant motors firing at liftoff and 493,500 pounds with three solid-propellant motors firing.

The main engine consists of the following subsystems and major assemblies: turbopump lubrication subsystem; pneumatic control subsystem; gas generator and exhaust subsystem; electrical subsystem; main propellant subsystem; and combustion chamber.

a. The two vernier engines, each contributing 1200 pounds of thrust during pump-fed operation and 925 pounds during tank-fed operation, provide roll control during first-stage powered flight.

b. The fuel tank subsystem is comprised of a fuel tank, valves, fittings, and connections that control the filling and pressurization of the main fuel tank (see Fig. C5-3).

c. The liquid oxygen subsystem is composed of a liquid oxygen tank, valves, fittings, and connections that control the filling and the pressurization of the main liquid oxygen tank.

d. The electrical subsystem is an electrical control and utility system that provides the necessary circuitry and electrical units to control main engine components and operate heating elements. The heaters protect critical components in or near the liquid oxygen from freezing. Heaters are provided on the main liquid oxygen valve, pneumatic regulator, turbopump gearcase, and fuel additive blender unit (FABU).

e. The main engine and vernier engine ignition subsystem consists of a container holding pyrophoric chemicals that ignite spontaneously when combined with oxygen. These hypergolic igniters are installed in the fuel flow lines to the combustion chambers and are isolated by burst diaphragms.

C5.2.2 Liquid Propellant Systems (Second Stage)

The second-stage propulsion system for the 391X or 392X vehicles consists of a gimbal-mounted thrust chamber assembly, propellant systems, gaseous helium propellant pressurization system and retro thrust system (if required), interconnecting plumbing, gimbal actuation system, gaseous-nitrogen-powered flight roll control and coast flight attitude control system and restart propellant settling system, forward tank skirt, and necessary airframe structures. (See Figure C5-4).

The thrust chamber assembly used on the 391X Vehicle is the TRW TR-201 Engine. The engine used on the 392X Vehicle is the Aerojet Improved Transtage Injector Program (ITIP) Engine. Both thrust chamber assemblies are ablatively cooled. A summary of the significant propulsion system parameters is shown below.

Table C5-1 Propulsion System Parameters

Parameter	391X	392X
Propellant Tank Pressure (Nominal), psig	255	231
Propellant Weight, lb	10,000	13,200
Delivered Specific Impulse (sec)	301.4	319.2
Thrust, lbs	9,425	9,443
Mixture Ratio	1.6	1.9
Area Ratio, AE/AT	46:1	65:1
Chamber Pressure, psia	105	125

The propellant system consists of an oxidizer tank and a fuel tank separated by a common bulkhead that has the dome on the oxidizer side of the tank. The propellants of this system are nitrogen tetroxide (N_2O_4) as the oxidizer and Aerozine-50 (A-50) as the fuel. The propellants are loaded into their respective tanks through a fill port that consists of a fill valve and interconnecting plumbing to the tanks. The propellant is pressure-fed to the engine through flexible hose assemblies containing turbine-type flowmeters and using helium gas as described in the helium system section. The remainder of the system consists of a fuel vent valve (FVV) and an oxidizer vent valve (OVV) and two drain disconnect ports.

The propellant system schematic and some of the major component schematics of this system are shown in Figure C5-4.

The helium system consists of a helium fill quick disconnect (QD) assembly and check valve, which is used to fill the gaseous helium spherical bottles, and a pressure regulator, which provides a common pressurization supply to both propellant tanks, and one each oxidizer and fuel tank pressurization shutoff valve. This system also includes a helium vent valve (HVV), which is used to vent the helium pressurization system through the vent QD at the umbilical upon receiving a ground command; a squib-fired retro valve, used for retro function on vehicles where it is required after the third stage separation; and the oxidizer and fuel tank ground service system, used during propellant loading to provide monitor capability for propellant overflow. All the plumbing on the helium system is interconnected with brazed joints except for a few connections that require disconnection to support checkout.

The pressurization system uses gaseous helium stored at 4350 $\pm$ 50 psig. The helium system schematic and some of the major component schematics of this system are shown in Figure C5-4. The nitrogen system schematic and some of the major component schematics of this system are shown in Figure C5-5.

The nitrogen system consists of a gaseous nitrogen fill QD, fill check valve, and one 1.27 ft^3 spherical bottle that stores gaseous nitrogen at a pressure of 4150 psig for 391X vehicles and 4365 psig for 392X vehicles. The Attitude Control System (ACS) consists of a regulator to provide 255 ± 20 psig nitrogen gas pressure, a relief valve set at 380 to 400 psig, two control modules (each of which contains six solenoid-actuated controls and nozzles) and a propellant-settling system that contains two integrally mounted valves and two settling nozzles.

The ACS provides thrust to control the stage in roll during engine operation and in pitch, yaw, and roll planes during coast periods. In addition, the ACS provides positive thrust for settling the propellants in their respective tanks and for removing entrapped gas from the propellants prior to each engine start. The ACS is controlled by the Delta Inertial Guidance System (DIGS). The control valves within the ACS are redundant to the extent that if a malfunction is detected in the primary control bank of valves, the pneumatic and electronics will be switched to the backup bank of solenoid valves. The pneumatic switchover is caused by the firing of an explosive squib on the control module, which moves the spool to route the pneumatics from the primary bank of solenoid valves to the backup bank of solenoids. The attitude control modules are located on the Quad I and Quad III positions of the second-stage propulsion unit (SSPU).

C5.2.3 Solid-Propellant Systems (First Stage)

The first stage solid-propellant propulsion system consists of nine Thiokol Castor IV solid motors used to augment the thrust of the main booster engine. Each motor consists of the basic solid rocket motor, a pyrogen igniter assembly, and two thru-bulkhead initiators (TBIs). Six of the nine Castor IV motors (TX526-2) are ignited on the ground; the remaining three (TX526-3) are ignited in the air. The TX526-2 motors have an 11-degree canted nozzle; the TX426-3 motors have a 7-degree canted nozzle. The ignition system consists of redundant electric detonators and explosive transfer assemblies.

NAME:	SOLID PROPELLANT MOTOR
PART NUMBER:	Thiokol Corporation Castor IV (TX526-2 and TX526-3)
MILITARY HAZARD CLASSIFICATION:	1.3
MILITARY STORAGE COMPATIBILITY:	C
DOT CLASSIFICATION:	B
EXPLOSIVE COMPOSITION AND WEIGHTS:	
EXPLOSIVE	WEIGHT %
A. Ammonium Perchlorate	70
B. HA Polymer and Binder	16
C. Aluminum	14
Total Propellant Wt.	20,500 lb per motor
QUANTITY PER VEHICLE:	9
MANUFACTURER:	Thiokol Corporation, Huntsville Div.

The solid-propellant motor (Model No. TX526-2 or -3) has a 53.9-second nominal web burning time; 84,900 pounds average thrust; a total impulse of 4,[illegible]22,900 pounds-seconds; and a propellant specific impulse of 229.0 pound-seconds/pound at 80°F and sea-level conditions. The motor case is fabricated of 4130 steel. The nozzle is a convergent-divergent type with either an 11-degree or a 7-degree cant angle and is fabricated from steel and composite plastic materials. Nominal motor assembly dimensions (nose cone included) are 439.4 inches overall length and 40 inches case diameter. The motor is attached to the booster by means of two ball and socket and spring thruster arrangements. A sketch of the motor attachment features is shown in Figure C5-6. The TX526-2 and 3 motors use an ammonium perchlorate-hydrocarbon propellant cast in the motor case. The internal surface of the motor case is lined with a polyisoprene insulating material to protect the case from overheating during propellant burning.

C5.2.4 Pyrogen Igniter Assembly

The pyrogen unit is a small rocket motor located at the forward end of each of the TX526-2 and -3 solid propellant motors and serves to ignite the motor. It is installed in the motor by the motor manufacturer.

The pyrogen unit, when initiated by either of the two TBIs, projects high-temperature flame along the surface of the solid motor propellant and ignites the solid motor propellant grain.

NAME:	PYROGEN IGNITER ASSEMBLY
PART NUMBER:	R54334
MILITARY HAZARD CLASSIFICATION:	1.3
MILITARY STORAGE CLASSIFICATION:	C
DOT CLASSIFICATION:	B
DOT SHIPPING NOMENCLATURE:	Jet Thrust Unit, Class B Explosive
EXPLOSIVE COMPOSITION AND WEIGHTS:	
EXPLOSIVE	WEIGHTS
A. PBAA Propellant	2700 grams
B. $BKNO_3$ Pellets	35 grams
QUANTITY PER VEHICLE:	9
MANUFACTURER:	Thiokol Corporation

C5.2.5 Solid-Propellant Systems (Second Stage)

The second-stage solid-propellant systems consist of as many as eight solid-propellant spin rockets. These spin rockets rotate the third stage prior to separation to a predetermined rotational velocity to stabilize the third stage for orbital entry.

C5.2.6 Solid-Propellant Systems (Third Stage)

The third-stage solid-propellant propulsion system consists of a TE-M-364-3, -4, or -14 solid-propellant motor and an E25393 motor igniter. The third-stage motor provides the thrust to position the spacecraft into the proper transfer orbit.

NAME:	THIRD STAGE SOLID PROPELLANT MOTOR (TE-M-364-3, TE-M-364-14)
PART NUMBER:	Thiokol Corp. E18518-20(-3), E31586-01(-14)
MILITARY HAZARD CLASSIFICATION:	1.3
MILITARY STORAGE COMPATIBILITY:	C
DOT CLASSIFICATION:	B
EXPLOSIVE COMPOSITION AND WEIGHTS:	
EXPLOSIVE	WEIGHT
PBAA Composite Propellant (TP-H-3062)	1440 lb (-3), 1230 lb (-14)
QUANTITY PER VEHICLE:	1
MANUFACTURER:	Thiokol Chemical Corp, Elkton, MD

The TE-M-364-3, as shown in Figure C5-7, is a solid-propellant rocket motor with 43.6 seconds nominal burn time and 9480 pounds average thrust, provides a total impulse of approximately 418,000 pound-seconds and a propellant specific impulse of approximately 290.4 pound-seconds/pound (vacuum conditions). The motor uses a composite-base propellant grain. It is mounted in the spin table at the forward end of the second stage and is held in place by a V-clamp. The motor has 18-inch- and 37-inch-diameter forward support rings to interface with spacecraft attachment fittings.

The TE-M-364-14 motor is identical to the TE-M-364-3 motor with the exception of the propellant load. The motor has a 39-second burn time, a 9160-pound average thrust, a 209.4-second propellant specific impulse, and a 357,200-pound-second total impulse.

C5.2.7 Third-Stage Solid Propellant Motor Igniter

The third-stage motor igniter, as shown in Figure C5-8, is located in the forward dome of each of the TE-M-364-3, -4, and -14 solid propellant motors. Dual initiators provide redundancy for ignition. The flame from the igniter is dispersed along the surface of the propellant of the TE-M-364 motor for rapid and uniform ignition of the motor propellant.

NAME:	THIRD STAGE SOLID PROPELLANT MOTOR IGNITER (TE-M-364-3, -4, AND -14)
PART NUMBER:	Thiokol Corporation E25393
MILITARY HAZARD CLASSIFICATION:	1.3
MILITARY STORAGE COMPATIBILITY:	C
DOT CLASSIFICATION:	B
EXPLOSIVE COMPOSITION AND WEIGHTS:	
EXPLOSIVE	WEIGHT
A. PHAA Composite (TP-H-3062)	277 grams
B. Boron-Potassium Nitrate	19.0 grams
QUANTITY PER VEHICLE:	1
MANUFACTURER:	Thiokol Corp, Elkton, MD

C5.2.8 Electrical Systems (First Stage)

Guidance and control for the booster is provided by the second-stage Delta Inertial Guidance System (DIGS), which maintains vehicle stability and guides the vehicle through the desired trajectory. The DSV-3P-15A Guidance Computer and DSV-3P-14 Delta Redundant Inertial Measurement System (DRIMS) form the DIGS for the first stage. The DIGS, with an applicable flight program, provides inertial sensing and issues the necessary steering signals to the DSV-3P-12 Electronic Package installed in the booster center section. The Electronics Package, in turn, controls the main and vernier engines by the gimbal actuation system. The DIGS also provides discrete commands to the Booster Ordnance Sequence (BOS) Box and the Electronics Package for booster sequencing.

The BOS Box initiates the sequencing for the solid motor ground ignition and flight ignition, solid motor separation, MECO enable, VECO and I/II stage separation. For the DSV-3P-11E and DSV-3F-11F vehicles, six Castor IV solid-propellant motors are ignited at liftoff. The three remaining solid motors are ignited at predetermined time after liftoff. Shortly after flight ignition, the six expended solid motors are separated in groups of three by means of pyro-delay (Solid Motor Separation 1). The remaining three solid motors are separated after burnout at Solid Motor Separation 2.

The instrumentation system and range safety system installed in the first-stage center section consist of two multicoders, transmitter, command destruct receiver (CDR), associated antennas, and telemetry battery. The instrumentation system converts signals from transducers and monitor systems aboard the first stage to RF signals for transmission to ground stations. The range safety system provides first-stage engine cutoff and destruct of the first stage in event of mission failure.

C5.2.9 Electrical Systems (Second Stage)

The guidance compartment of the second stage houses the major components of the power distribution system, sequencing system, instrumentation system, range safety system, and the DIGS.

The DSV-3P-15A Guidance Computer, DSV-3P-14 Delta Redundant Inertial Measurement System, and DSV-3P-13 Electronics Package interface together to form the DIGS. The DIGS serves the same function for the second stage as for the first stage to achieve the required mission trajectory. During powered flight, pitch and yaw control are achieved by gimbaling the engine; roll control is provided by gas jets initiated by DIGS commands. During the coast phase, the vehicle is attitude-controlled by pitch, yaw, and roll jets actuated by DIGS commands.

The Guidance Computer, Power and Sequence Box, Engine Battery, Control Battery, and Instrumentation Battery make up the power distribution system and sequencing system. The power distribution system provides power to all electrical components in the guidance compartment. The sequencing system distributes the power and ordnance power at predetermined times by stored discrete commands issued by the guidance computer. The discrete commands drive relays, which in turn switch power to arm ordnance buses and to fire ordnance squibs.

The instrumentation system and range safety system installed in the second-stage guidance compartment consist of two command destruct receivers, transmitter, instrumentation telemetry package, associated antenna systems, and telemetry battery. The instrumentation system provides to ground stations the vehicle status data gathered from transducers and monitor systems aboard the second stage. The range safety system aboard the second stage provides the first- and second-stage engine cutoff and destruct of first, second, and third stages in case of mission failure. The range safety system consists of two command destruct receivers, range safety box and safety box and safe-arm devices.

C5.2.10 Electrical Systems (Third Stage)

All third-stage ordnance sequencings are implemented by the second-stage guidance computer. The ordnance systems receive power from the second-stage engine battery via third-stage ordnance buses.

C5.2.11 Propellant Pressurization Systems (First Stage)

The first-stage propellant pressurization system pressurizes the start and main propellant tanks to permit flow into the combustion chamber before startup of the turbopump. After MECO the pressure in the start tanks supply the vernier engines for an additional nine seconds to depletion. The propellant pressurization system consists of gaseous nitrogen (GN_2) tanks, valves, regulators, and pneumatic lines and is shown schematically in Figure C5-9. The component data in Table C5-2.

Table C5-2 Propellant Pressurization System Component Data

Item	Material	Pressurant	Operating	Proof	Calculated Minimum Burst	Tank Capacity
GN2 Tanks(4)	4130 Stl	GN2	3000	4000	4800	2600 in. 3
FABU Tank	6061 T 6510 Al. Alloy	Fuel	650	1200	N/A	156 in. 3
LOX Start Tank	7075-T6 Al. Alloy	GN2	650	1125	2090	1517 in. 3
Fuel Start Tank	Al. Alloy	GN2	650	1125	1500	1306 in. 3
Fuel Tank	2014-T6	GN2	30	43.65	57.8	1130 ft3
LOX Tank	2014-T6 Al. Alloy	GN2	36	62.55	72.6	1753 ft3
Pneumatic Regulator	2014-F Al. Alloy	GN2	650	4500	N/A	-----
Pneumatic Lines	CRES	GN2	3000	4500	12,000	-----

C5.2.12 Propellant Pressurization Systems (Second Stage)

The second - stage propellant pressurization system uses gaseous helium (GHe) stored at 4400 psig for 391X vehicles and 4415 psig for 392X vehicles. The helium system consists of a helium fill quick-disconnect assembly and check valve used to fill the gaseous helium spherical bottles, a pressure regulator that provides a common pressurization supply to both propellant tanks, and both an oxidizer and fuel tank pressurization shutoff valve. This system also includes a helium vent valve (HVV) used to vent the helium pressurization system through the vent QD at the umbilical on receipt of a ground command; a squib-actuated retrovalve is used for retro function after the third stage separation. The oxidizer and fuel tank ground service system is used during propellant loading to provide monitor capability for propellant overflow. All the plumbing on the helium system is interconnected with brazed joints except for a few connections that require disconnection to support checkout. The helium schematic and some of the major component schematics of this system are shown in Figure C5-4 for the 392X vehicle.

A gaseous nitrogen (GN_2) system provides for attitude control of the second stage through the Redundant Attitude Control System (RACS). The nitrogen system consists of a gaseous nitrogen fill QD, fill check valve and one 1.27-ft^3 spherical bottle that stores gaseous nitrogen at a pressure of 4200 psig for 391X vehicles and 4415 psig for 392X vehicles. The Attitude Control System (ACS) consists of a regulator to provide 255 $\pm$ 20 psig nitrogen gas pressure, a relief valve set at 380 to 400 psig, two control modules (each of which contains six solenoid-actuated control valves and nozzles) and a propellant settling system containing two integrally mounted valves and two settling nozzles. A schematic of the RACS is shown in Figure C5-5. The component data are presented in Table C5-3:

Table C5-3 RACS Component Data

Item	Material	Fluid Pressurant	Operating	Pressure Level - psig Proof	Calculated Minimum Burst	Tank Capacity
Helium Tanks (2)	Titanium	GHe	4,415	5,065 ± 15	5,910	6,440 in.3
Helium Tank (1)	Titanium	GHe	4,415	5,200	5,904	2,200 in.3
Nitrogen Tank (1)	Titanium	GN2	4,415	5,200	5,904	2,200 in.3
Fuel Tank	410 CRES	GHe	230 ± 8	256 ± 2	289	82.89 ft.3
Oxidizer Tank	410 CRES	GHe	230 ± 8	256 ± 2	289	98.02 ft.3
GHe Regulator**	CRES	GHe	4,415 259	7,250; 800	11,000 4,800*	----
GN2 Regulator**	CRES	GN2	4,415 300	7,250; 800	11,000; 4,800*	----

C5.2.13 Vehicle Hydraulic Systems (First Stage)

The first stage hydraulic system supplies the hydraulic power required to position the first-stage main and vernier engine thrust chambers during powered flight. The hydraulic system consists of a pressure-compensated variable displacement hydraulic pump; accumulator/reservoir assembly (a module containing a bootstrap reservoir, accumulator, filter, high pressure relief valve, low pressure relief valves, check valve, and associated instrumentation); six hydraulic actuator assemblies (two for each engine); hydraulic fluid; and hydraulic lines, fittings, and control components as required. Quick disconnect fittings on the airframe aft closure structure and associated hydraulic lines are provided for connecting the hydraulic system of the vehicle to ground source of hydraulic power.

Each of the six actuator assemblies consists of a servo valve, an actuator (cylinder and piston), and a position feedback potentiometer. The hydraulic actuators can gimbal the main and vernier engines in the pitch and yaw planes in response to control signals to the servovalves from the dc amplifiers of the flight control system. Actuator position feedback voltage is provided to the flight control system by the feedback potentiometers mechanically driven by the actuators.

C5.2.14 Vehicle Hydraulic Systems (Second Stage)

The second-stage hydraulic system provides the power to position the engine thrust chamber in response to commands from the flight control system. This positioning is accomplished by two linear-motion hydraulic actuators. The hydraulic system for the 391X vehicle also contains a hydraulic actuator used to position the propellant flow control valves in open or closed positions. This actuator is delivered to MDAC-HB as part of the second-stage engine, manufactured by TRW. All other components are manufactured or subcontracted by MDAC-HB.

The two gimbal actuators consist of a cylinder, piston, servo-valve, and potentiometer. The flow control valve actuator for the 391X vehicle consists of a cylinder, piston, and solenoid valve. The other hydraulic system components are an electric-motor-driven hydraulic pump, a reservoir, an accumulator, connecting lines, a check valve, a relief valve, and two quick disconnects.

C5.2.15 Ordnance Systems (First Stage)

The first-stage ordnance systems perform three functions: destruct, solid motor ignition and separation, and turbopump startup. The destruct system is initiated by an electrical signal to redundant safe and are devices. The explosive output from the safe and arm devices is transmitted by various detonating fuse harnesses to the various destruct charges located around the booster tanks and in the solid motors. The detonating fuse circuits are duplicated and interconnected for redundancy.

C5.2.16 Ordnance Systems (Second Stage)

The second-stage ordnance systems perform the following functions:

a. Second-stage destruct;

b. Second-stage destruct;

c. Attitude control system isolation;

d. Retro-valve operation;

e. Spacecraft fairing separation;

f. Third-stage spin-up;

g. Third-stage separation.

C5.2.17 Ordnance Systems (Third Stage)

The third-stage ordnance systems perform the following functions: destruct, third-stage solid motor ignition, and spacecraft separation.

C5.3 MISSION SCENARIO

The main engine provides pitch and yaw control by gimbaling in the pitch and yaw planes during powered flight phase. The vernier engines are gimbaled to provide roll attitude control prior to main engine cutoff (MECO). The vernier engines also provide roll, pitch, and yaw attitude control from MECO through vernier engine cutoff (VECO).

Six of the Castor IV solid propellant motors (TX526-2) are ignited at lift-off. The three Castor IV (TX526-3) motors that are not ignited at lift-off are ignited at a predetermined time after lift-off. The six motors ignited at lift-off are jettisioned during the early portion of the air-start motor operation. After the air-start solid-propellant motors burn out, they are jettisoned simultaneously.

The booster liquid propulsion system has a burn time of approximately 226 seconds. Each TX526-2 solid motor has a total burn time of 57.4 seconds. Each TX526-3 solid motor has a total burn time of 57.4 seconds.

The second-stage sequence of events, for the 392X vehicles, begins at Main Engine Cutoff (MECO) plus 13 seconds. At MECO plus 13 seconds, the tank shutoff pilot valve (TSPC) is energized and pneumatically opens the oxidizer and fuel tank shutoff valves for pressurizing the second-stage propellant tanks.

At the same time, the second-stage engine is given a start command. The oxidizer and fuel tank shutoff valves remain open throughout the first burn (approximately 7 minutes). At the first second-stage engine cutoff (first SECO), the tank shutoff valves are closed and remain closed for the duration of the mission. The retro function, after third-stage separation, reduces the helium supply bottle pressure to zero.

The current start sequence is:

MECO + 13 SEC - Tank shutoff valves open and engine start.

The third-stage sequence of events is dependent on the third-stage motor employed.

The TE-M-364-3 third stage consists of a Thiokol Corporation TE-M-364-3 solid propellant rocket motor with an action time of 44.1 seconds.

The TE-M-364-4 third stage consists of a Thiokol Corporation TE-M-364-4 solid propellant rocket motor with an action time of 43.6 seconds.

The TE-M-364-4 third stage consists of a Thiokol Corporation TE-M-364-14 solid propellant rocket motor with an action time of 39 seconds.

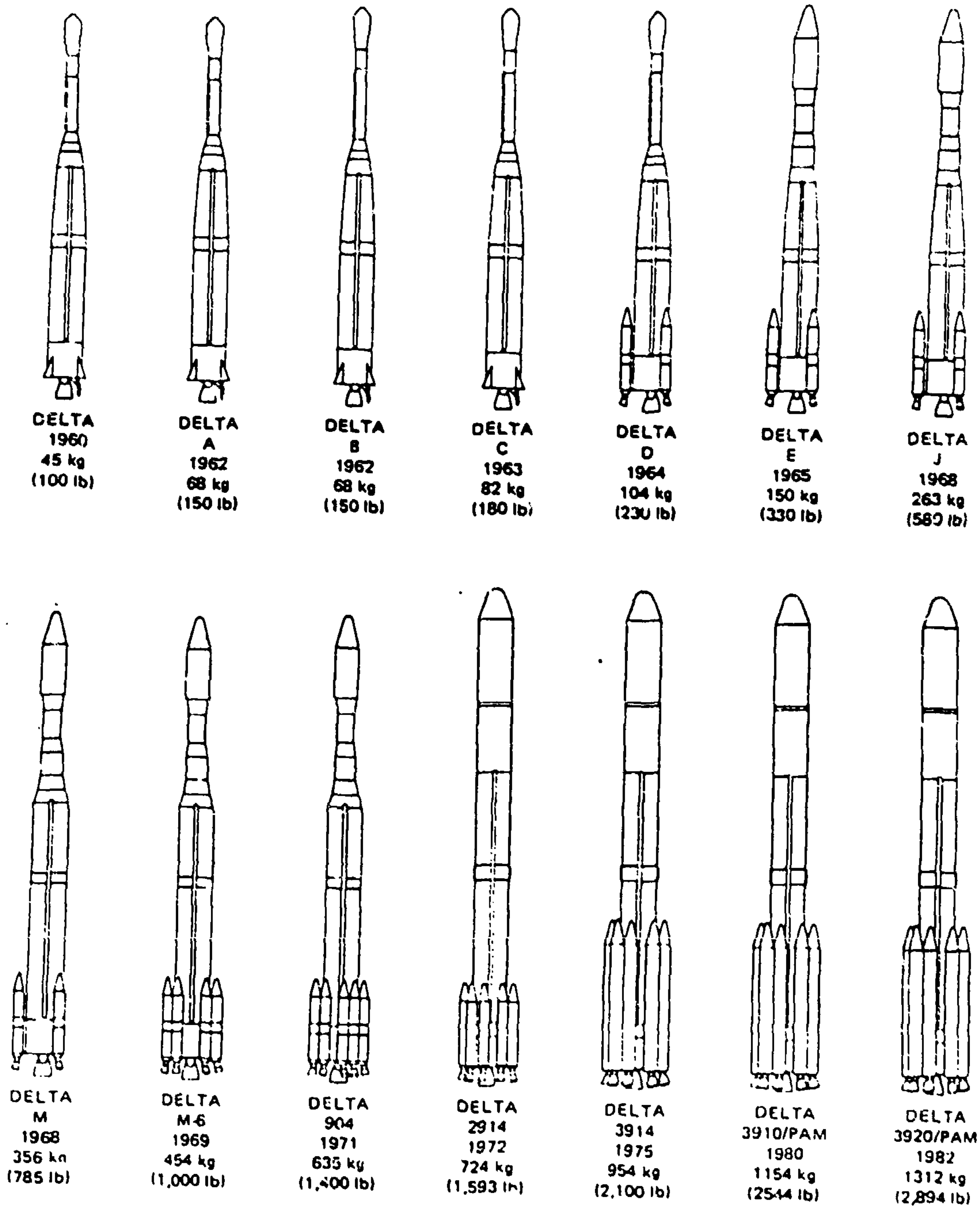

Figure C5-1 Geosynchronous Transfer Orbit Capability - ESMC

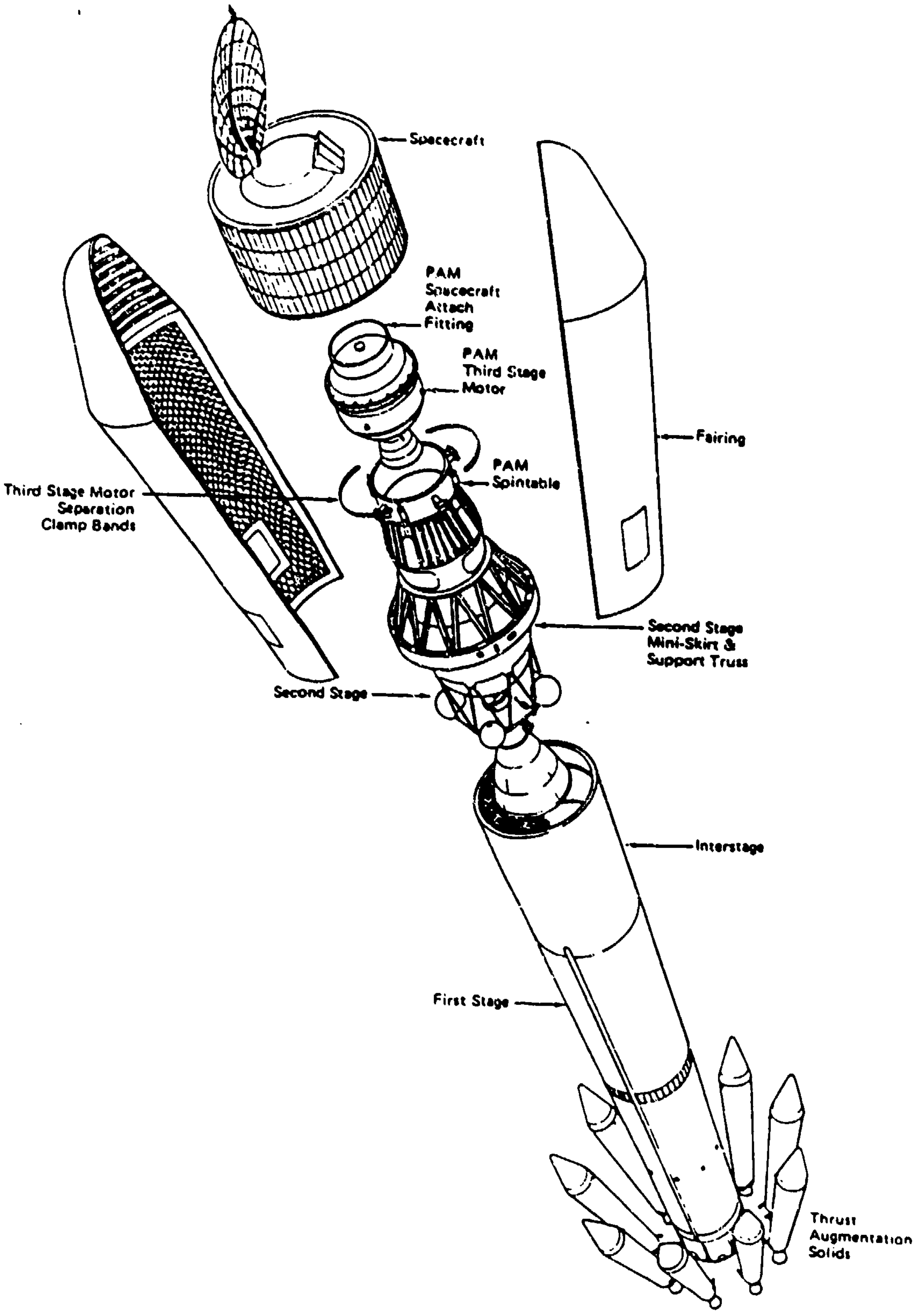

Figure C5-2 Delta Launch Vehicle

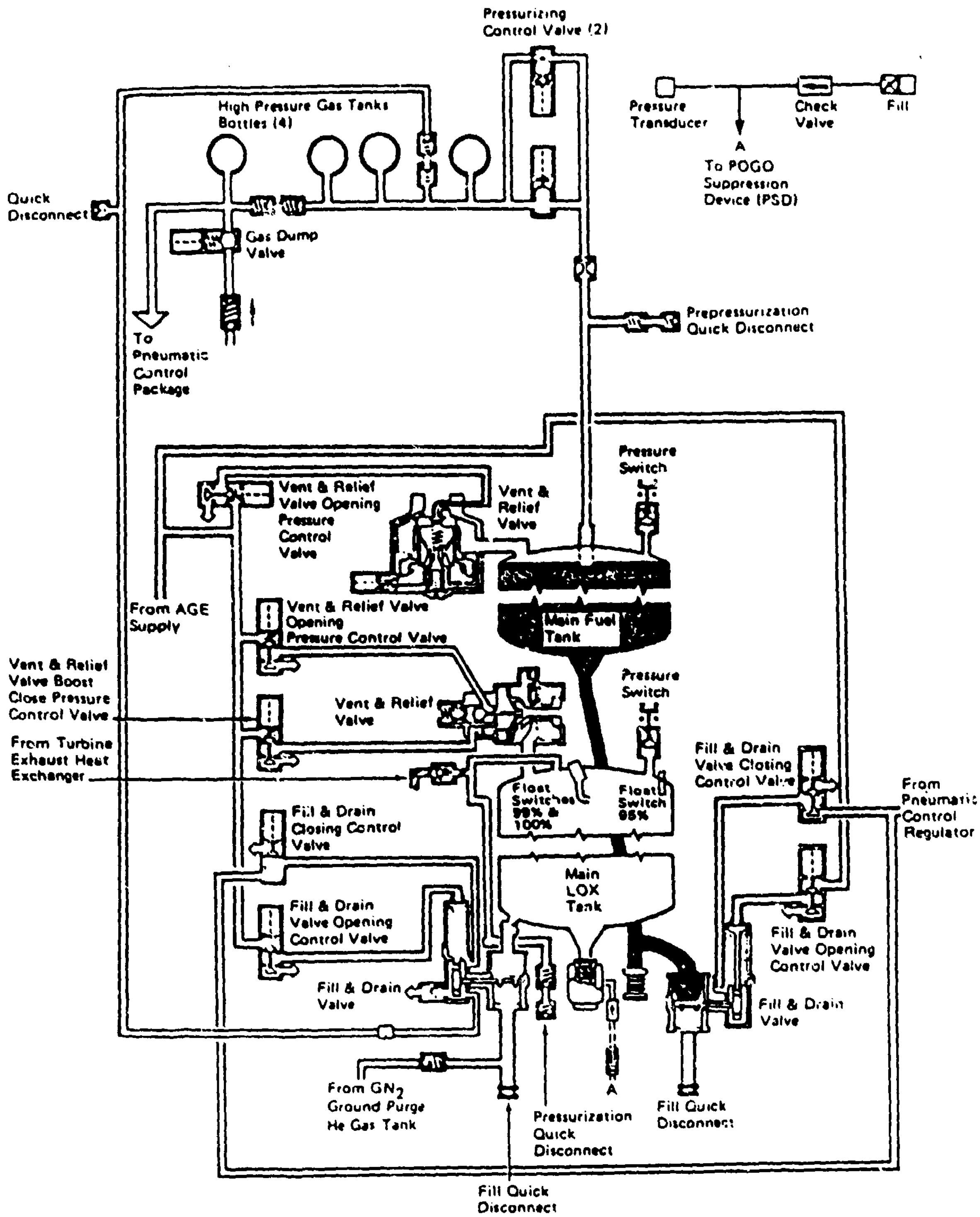

Figure C5-3 First-Stage Propulsion System - High-Pressure Schematic

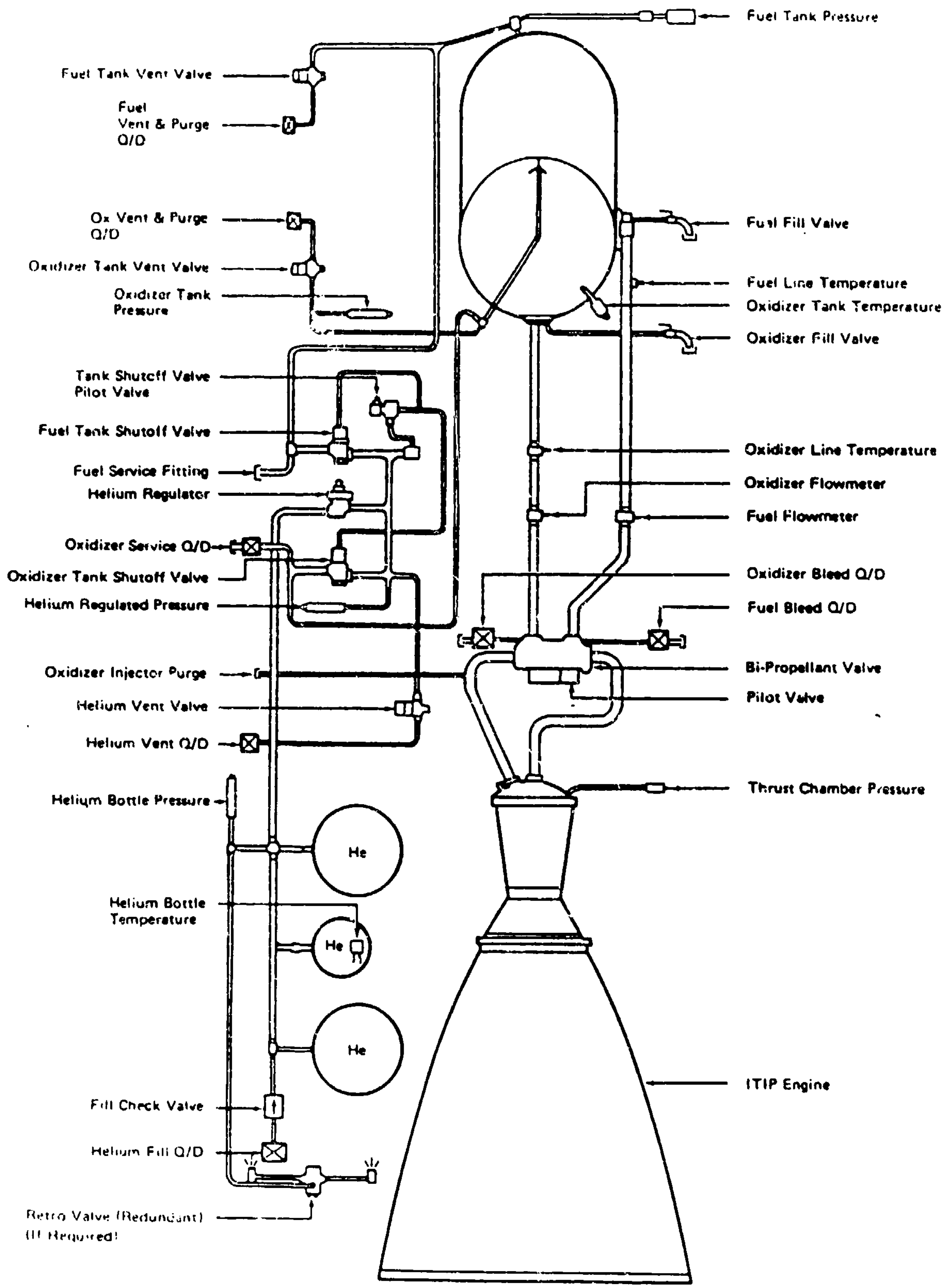

Figure C5-4 Schematic of 392X Propulsion System

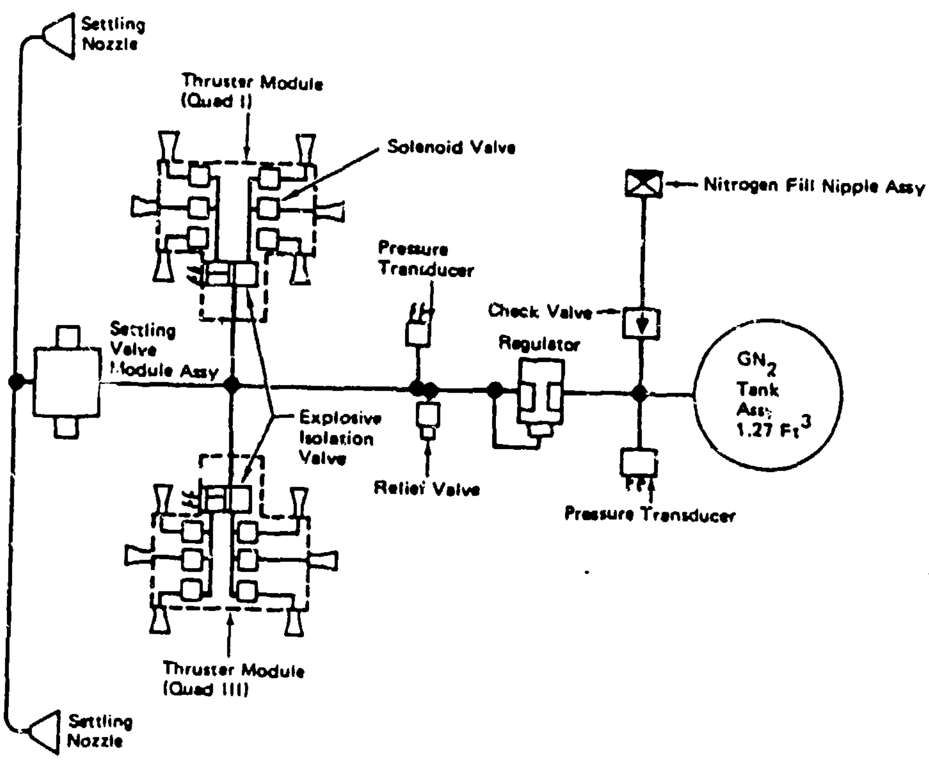

Figure C5-5 391X and 392X Vehicle Nitrogen System Schematic

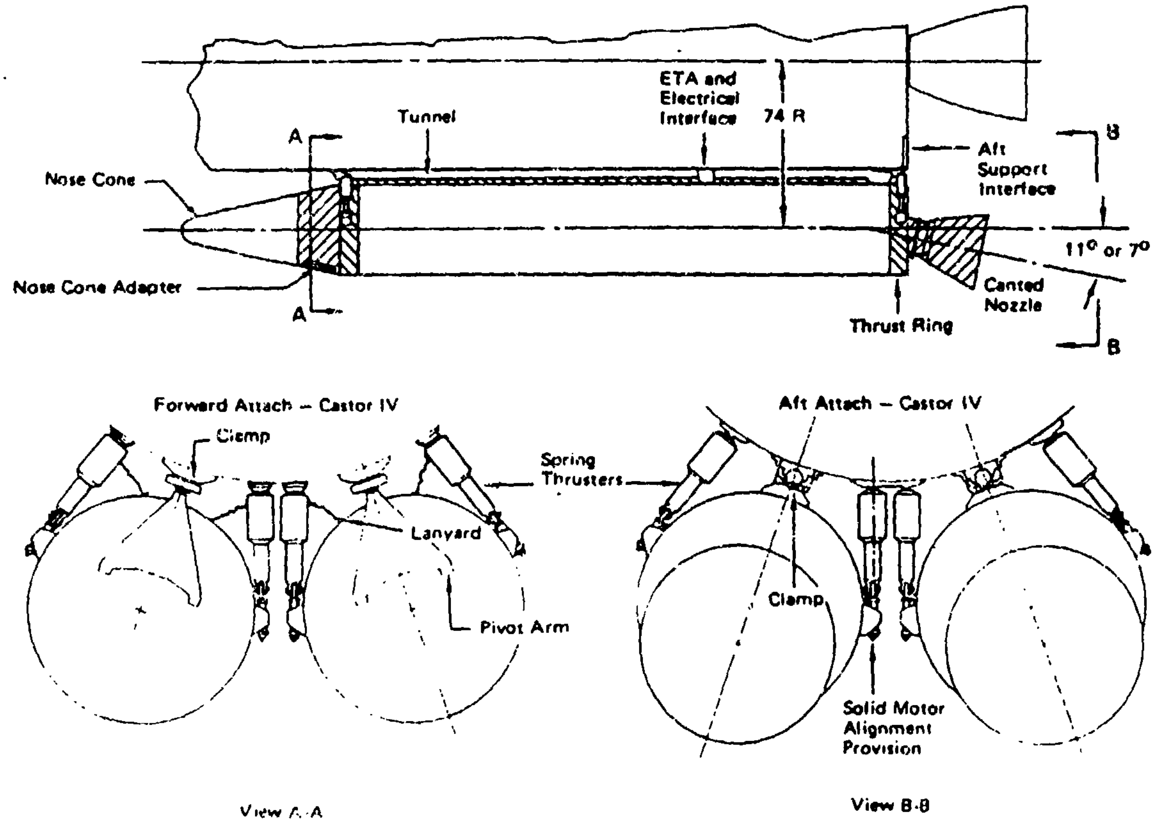

Figure C5-6 Solid-Propellant Motor Mounting

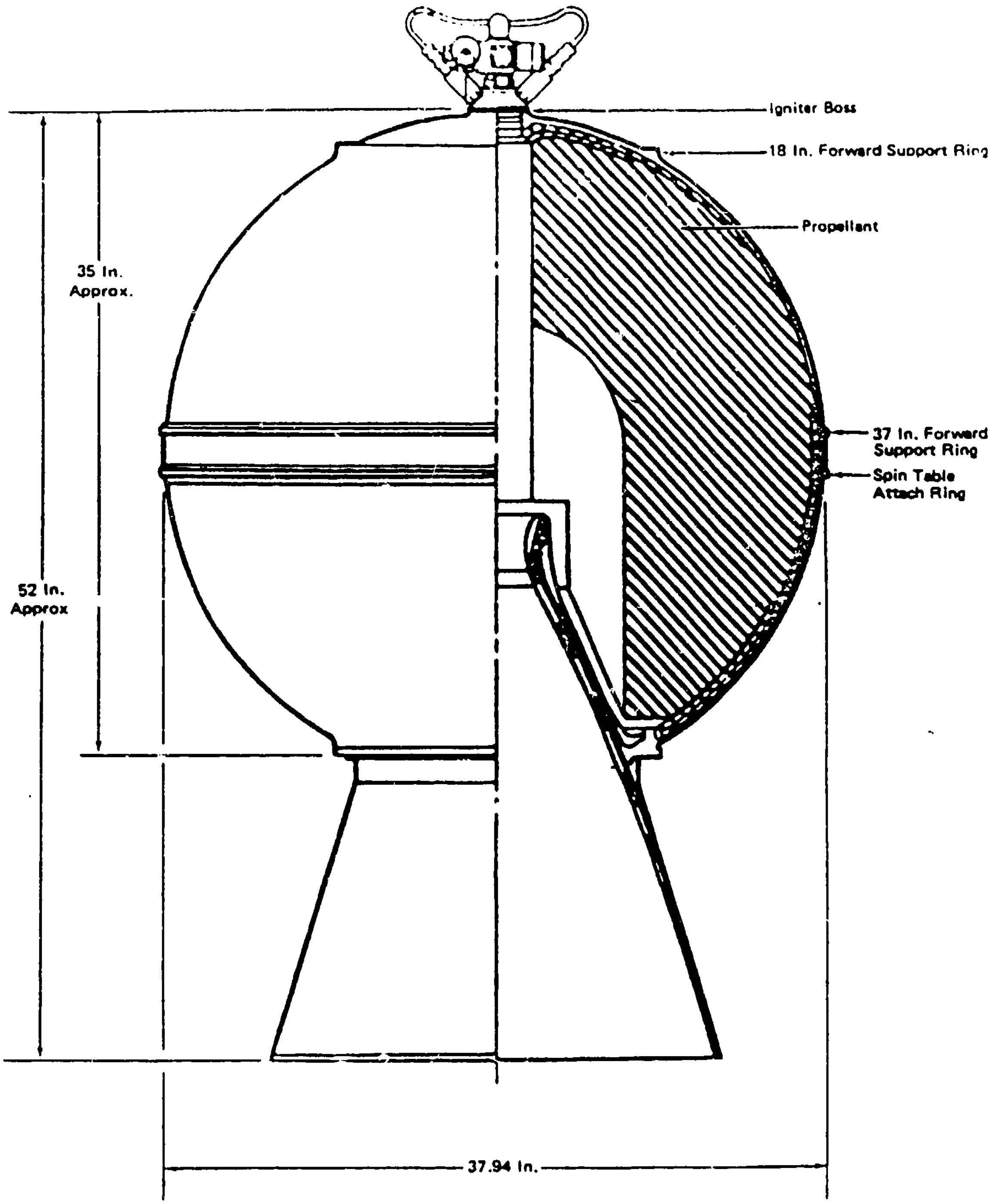

Figure C5-7 TE-364-3 and -14 Motor Configuration

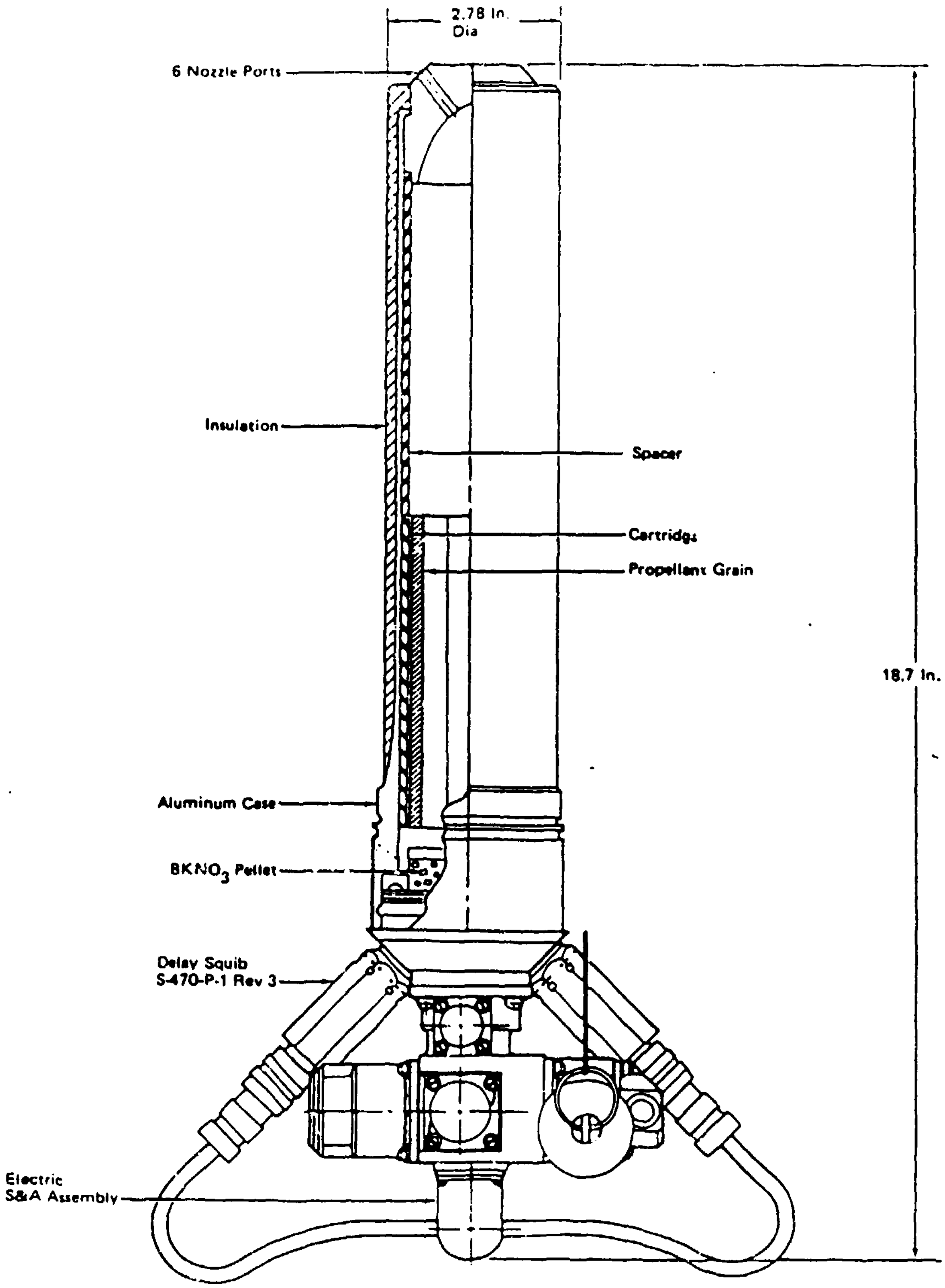

Figure C5-8 Third-Stage Solid Propellant Motor Igniter

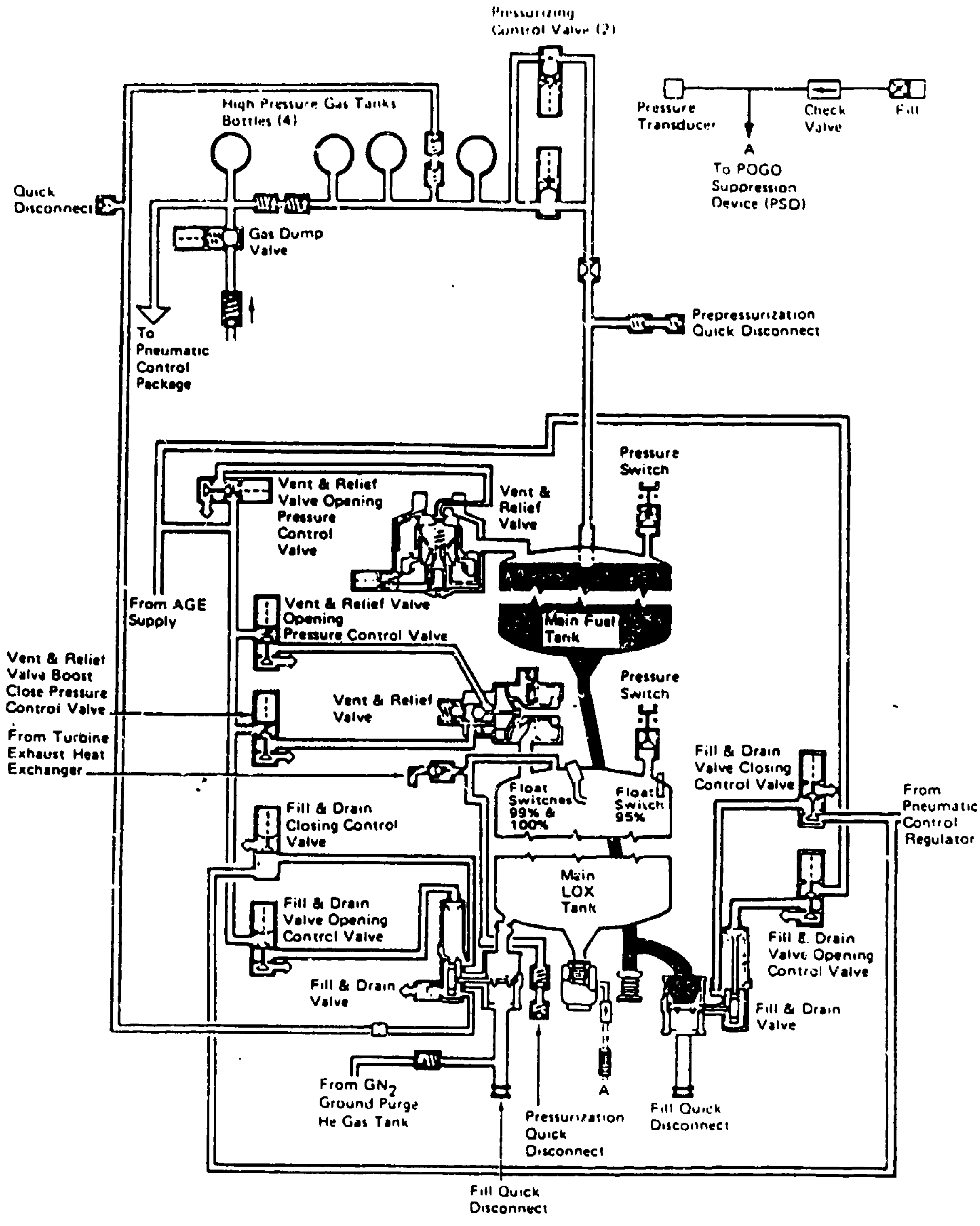

Figure C5-9 First-Stage Propulsion System - High-Pressure Schematic

Appendix C6
Titan II

APPENDIX C6
TITAN II (USAF Model LGM-25C)

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LIST OF FIGURES

C6.0 INTRODUCTION

The following data were extracted from Technical Manual, USAF Model LGM-25C, Missile Weapon System Operation, T.O. 21M-LGM25C-1, 1 Feb 1976, revised 15 May 1984, OO-ALC/MMED, Hill AFB, Utah 84056. For additional information see supplemental publication T.O. 21M-LGM25C-1-2 (classified). Data were also extracted from Technical Manual, USAF Model LGM-25C, Propellant System, Missile Weapon System, T.O. 21M-LGM25C-2-12-2, 7 May 1982, revised 28 Oct 1983, OO-ALC/MMED, Hill AFB, Utah 84056.

C6.1 LGM-25C MISSILE - GENERAL DESCRIPTION

The LGM-25C ballistic missile consists of a two-stage rocket-engine-powered vehicle and a reentry vehicle (RV). Provisions are included for in-flight separation of Stage II from Stage I, and separation of the RV from Stage II. The Stage I and Stage II vehicles each contain propellant and pressurization, rocket engine, hydraulic and electrical systems, and explosive components. In addition, Stage II contains the flight control system and missile guidance set. Figures C6-1 and C6-2 illustrate the major characteristics of the LGM-25C missile.

The launch complex (Fig. C6-3) includes both above-ground and hardened underground facilities. Hardened facilities include the missile silo, control center, blast lock, interconnecting cableways, emergency escape hatch, and hardened communications equipment. Above ground, nonhardened facilities include vehicle parking areas, security fencing and lighting, propellant and electrical connections, static grounding system, commercial power lines, a transformer, cooling tower pits, access portal, nonhardened antennas, soft water storage, area security surveillance system, and weather instruments.

The missile silo (Fig. C6-4) is a reinforced concrete structure with inside dimensions of approximately 146 feet in depth and 55 feet in diameter. A launch duct, constructed with a sound-attenuating lining, is located in the center of the silo. Associated installed equipment and structures include a silo closure door, retractable work platforms, hazard sensing devices, and a small equipment and personnel elevator operating between Levels 2 and 8. Two exhaust ducts carry missile exhaust and ingested air from the flame deflector through deflecting cascade vanes to the surface. Equipment areas are located between the launch duct and the missile silo walls on nine separate levels.

The launch silo closure door consists of the closure door, wheel trucks, track rails, buffers, and a door actuating system. The door actuating system consists of rail bridge jacks; door locks; a drive unit; cables; a pneudraulic power system; and electrical, hydraulic, and pneumatic controls.

Personnel safety is maintained throughout the launch complex by hazard sensing and warning equipment, protective clothing and equipment, safety equipment, and safety requirements. Because of the hazardous conditions that could exist in the launch complex, personnel must be familiar with safety procedures and the use of safety equipment.

Hazard sensing and warning equipment is located throughout the launch complex for hazard sensing and warning of personnel and for controlling propellant vapor level and fires. Hazard elimination, using water spray, air purge, or foam, is actuated automatically by the hazard system in critical areas.

C6.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C6.2.1 Airframe

The airframe (Fig. C6-5) is a two-stage aerodynamically stable structure that houses and protects the airborne missile equipment during powered flight. The missile guidance set enables the shutdown and staging enable relay to initiate Stage I separation. The missile guidance set initiates Stage II and RV separation. Each stage is 10 feet in diameter and has fuel and oxidizer tanks in tandem, with the walls of the tanks forming the skin of the missile in those areas. External conduits are attached to the outside surfaces of the tanks to provide passage for wire bundles and tubing. Access doors are provided on the missile forward, aft, and between-tanks structures for inspection and maintenance. A manhole cover for tank entry is located on the forward dome of each tank.

The Stage I airframe consists of an interstage structure, oxidizer tank forward skirt, oxidizer tank, between-tanks structure, and fuel tank. The interstage structure, oxidizer tank forward skirt, and between-tanks structure are all fabricated assemblies using riveted skin, stringers, and frame. The oxidizer tank is a welded structure consisting of a forward dome, tank barrel, aft dome, and feedline. The fuel tank, also a welded structure, consists of a forward dome, tank barrel, aft cone, and internal conduit.

Stage II airframe consists of a transition section, oxidizer tank, between-tanks structure, fuel tank, and aft skirt. The transition assembly, between-tanks structure, and aft skirt are all fabricated assemblies using riveted skin, stringers, and frames. The oxidizer tank and fuel tank are welded structures consisting of forward and aft domes.

C6.2.2 Rocket Engine System

The missile rocket engine system consists of a Stage I rocket engine (LR-87-AJ-5) and a Stage II rocket engine (LR-91-AJ-5). The Stage I rocket engine is designed to operate with a rated thrust of 430,000 pounds at sea level; Stage II rocket engine is designed to operate with a rated thrust of 100,000 pounds at an altitude of 250,000 feet. An autogenous propellant tank pressurization system is part of the rocket engine system.

The Stage I rocket engine (Fig. C6-6) consists of two independent subassemblies mounted on a single engine frame. Each subassembly contains a thrust chamber assembly, a turbopump assembly, a gas generator, and an engine start system. The two subassemblies have an integrated electrical system for simultaneous operation.

Stage I rocket engine operation is initiated by the launch-command signal from the LCCFC. Prior to Stage I rocket engine operation, the missile prevalves are opened to allow propellant to flow through the suction lines to the rocket engine. At countdown T-zero, a 28-Vdc signal is applied to the two solid-propellant starter cartridge initiators. Ignition and burning of the gas pressure generator produces hot gases that are directed through an inlet nozzle to the turbopump assembly turbine for initial acceleration and running of the turbine. The turbine, through a gear train, drives the fuel and oxidizer pumps. The fuel and oxidizer pumps deliver propellants through discharge lines to the thrust chamber valves. When fuel discharge pressure within the fuel discharge lines reaches 310 psi, a pressure-actuated valve (thrust chamber valve pressure sequencing valve) opens. Through mechanical coupling, the thrust chamber fuel and oxidizer valves open; fuel flows down the thrust chamber coolant tubes and back up into the injector and is emitted into the combustion chamber. Oxidizer flows directly through the injector into the combustion chamber. The propellants ignite hypergolically, and the flow of expanding gases from the nozzle produces thrust. Rocket engine sustained operation depends on bootstrap operation involving the turbopump assembly and the gas generator. Simultaneously with the flow of propellants to the thrust chamber, a small amount of fuel and oxidizer is drawn off below the thrust chamber valves into gas generator fuel and oxidizer lines. Cavitating venturis, located in the gas generator lines, control propellant flow to the gas generator. Propellant pressures open the check valves installed in the lines, allowing propellant to enter the gas generator. The propellants ignite hypergolically, and a fuel-rich gas is produced, which gas enters the turbine inlet, drives the turbine, and thereby sustains turbopump assembly operation. An autogenous pressurization system is used for inflight propellant tank pressurization.

The oxidizer tank is pressurized by directing a small amount of oxidizer into an autogenous system super-heater where the oxidizer is heated and converted to a gas. The expanding gas is directed to the oxidizer tank for inflight tank pressurization. Engine shutdown, at the end of Stage I flight, occurs when either oxidizer or fuel is depleted, causing a subsequent drop in thrust chamber pressure that is detected by the thrust chamber pressure switch. When this lower pressure is sensed, a signal is sent to the thrust chamber pressure sequencing valve to initiate closing of the thrust chamber valves, thereby terminating rocket engine operation. Simultaneously, a signal is sent to the Stage II separation nut squibs and gas pressure generator to initiate Stage II separation and rocket engine start.

The Stage II rocket engine (Fig. C6-7) consists of a thrust chamber assembly with ablative skirt, turbopump assembly, gas generator, fuel tank autogenous pressurization system, roll control assembly, and engine control system.

Except for minor differences, the Stage II rocket engine operates the same as the Stage I rocket engine. The initial start signal for the Stage II rocket engine is transmitted from the thrust chamber pressure switches located on the Stage I rocket engine. At Stage I engine shutdown, a signal is transmitted from the Stage I pressure switches to initiate the Stage II solid-propellant starter cartridge and separation nut squibs. The rocket engine self-sustaining and shutdown operations are similar to those of Stage I; however, the shutdown signal for Stage II is initiated by the missile guidance set. A roll control nozzle using exhaust gas from the gas generator is incorporated in the system to provide roll control of the missile during Stage II operation. The roll control nozzle is connected to a hydraulic actuator installed between the missile and the roll control assembly. The flight control system receives guidance signals from the missile guidance set and sends control signals to the actuator. Pressurization of the Stage II oxidizer tank is not required in flight. The Stage II fuel tank is pressurized using cooled exhaust gas.

C6.2.3 Airborne Propellant System

The airborne propellant system (Fig. C6-8) consists of fuel and oxidizer tanks, disconnects, pressure transducers, storage valves, and pressurization and vent piping. The primary purpose of the propellant system is to load the missile with propellants and initially pressurize the missile propellant tanks to lockup pressures. The system has the facilities for unloading the missile propellant tanks; pressure-testing, purging, and blanketing missile piping; and conditioning propellants. Normal and special unloading consists of connecting mobile equipment and transferring propellants.

Fuel supplied by the propellant system is hydrazine-UDMH 50/50 mixture (MIL-P-27402), and oxidizer is nitrogen tetroxide (MIL-P-26539A). Gaseous nitrogen (MIL-P-27401A) is used to pressure-test, purge, and blanket system tanks, lines, and components. Air pressure from the facility air system is used to actuate pneumatically operated valves in the launch silo.

The Stage I fuel and oxidizer fill-drain and Storage Valve 2 drain disconnects are located in the Stage I engine compartment. The remaining Stage I and Stage II disconnects are located at various points on the outer skin of the missile. The disconnect ground halves are connected to these disconnects to direct the flow of propellants and gases to and from the missile tanks. The disconnects are self-sealing and must be manually connected and disconnected.

Pressure transducers are located in the dome of each missile propellant tank. The output of these transducers is connected to a digital meter mounted on a facilities console indicating the selected propellant tank pressure in pounds per square inch gauge (psig). The transducer output is also used to operate related propellant system equipment if structural pressure limits are exceeded during loading or unloading operations. This function is accomplished by conversion of the transducer output to a 28-Vdc control signal in the Propellant Transfer System (PTS) structural pressure control unit.

There are six storage valves: four in the Stage I engine compartment and two in the Stage II engine compartment. These valves are gas-pressure (squib) actuated butterfly valves with zero-leak diaphragms. The diaphragms prevent propellants stored in the missile tanks from entering the engines. Each valve has a positive locking device that automatically locks the valve in the open position when the valve is actuated. Fill-drain disconnects are directly connected to one fuel valve and one oxidizer valve in Stage I. Two other Stage I valves have disconnects used to drain propellants trapped above the valves during unloading operations. Fill-drain disconnects are connected to Stage II valves by flex hoses.

The pressurization and vent piping for the Stage I fuel and oxidizer tanks and the Stage II fuel tank consists of flex hose between the top of each tank and disconnects on the skin of the missiles. The Stage II oxidizer tank pressurization and vent piping is a flex hose between the bottom of the tank and a disconnect on the missile skin. These systems are used to safely vent gases away from the missile during loading operations and to pressurize the missile tanks to flight pressure after loading. They are also used to pressurize the missile for leak check, purging, blanketing, and propellant unloading.

C6.2.4 System Equipment

The propellant system is divided into airborne equipment, fixed equipment, and mobile equipment.

Propellant system airborne equipment consists of disconnects, pressure transducers, storage valves, and piping.

Airborne half disconnects permit connecting ground half disconnects and hoses to the missile and allow flow of propellants and gases to and from the missile propellant tanks. Stage I fuel and oxidizer fill-drain and storage valve drain disconnects are located in Stage I engine compartment. Other Stage I and II disconnects are located at various points on the missile skin. The disconnects are self-sealing and must be manually connected and disconnected from the ground half disconnects.

A pressure transducer mounted on the dome of each propellant tank provides a voltage proportional to tank pressure. This voltage is applied to the Propellant Tank Pressure Monitor Unit (PTPMU) located on top of the Launch Control Complex Facilities Console (LCCFC). The PTPMU enables an operator to monitor any individual propellant tank pressure when the appropriate pushbutton switch is pressed. The tank pressure is indicated in psig on a digital read-out meter. Out-of-tolerance tank pressures may be predicted and subsequently corrected and avoided by scheduled monitoring and recording of tank pressures. To prevent structural damage to the missile during propellant transfer operations, the pressure transducer output voltage is applied to the Propellant Transfer System Control Unit (PTSCU). During propellant loading, structural overpressure is prevented when the pressure transducer output voltage proportional to the structural overpressure limit causes the PTSCU to apply a close signal to the propellant loading control valve. During propellant loading, structural overpressure is prevented when the pressure transducer output voltage proportional to the structural overpressure limit causes the PTSCU to apply a close signal to the propellant loading control valve. During propellant unloading, structural underpressure is prevented when the pressure transducer output voltage proportional to the structural underpressure limit causes the PTSCU to apply a shutdown signal to the propellant unloading pump.

Four storage valves are located in the Stage I engine compartment and two in the Stage II engine compartment. Storage valves are gas-pressure-cartridge actuated butterfly valves with zero leak diaphragms. The diaphragms prevent propellants in the missile tanks from entering the engines during loading, unloading, or storage of propellants. The valves are locked in the open position when actuated. Stage II storage valves are connected to the fill-drain disconnects by flexible hoses. One fuel storage valve and one oxidizer s orage valve in Stage I are connected directly to the fill-drain disconnects. The other Stage I valves have disconnects (dead-leg drains) that drain propellants trapped above the valves during unloading operations. The coded switch system butterfly valve lock (refer to T.O. 21M-LGM25C-2-26) is installed on the valve actuator shaft of the Stage I Oxidizer Storage Valve 2 (oxidizer valve with dead-leg drain). Any maintenance operation on this valve or tasks performed near this valve, after the butterfly valve lock has been installed, must be performed in strict adherence with Stage I/Stage II airborne fuel or oxidizer components replacement sequence requirements of T.O. 21M-LGM25C-1SC-1.

Pressurization and vent piping are used to vent gases from the missile propellant tanks during loading operations and to pressurize the tanks to lockup pressures after loading. The piping is also used to pressurize the missile propellant tanks for leak checks, purging, blanketing, and unloading. Stage I fuel and oxidizer tanks and Stage II fuel tank piping consists of a flexible hose from the top of each tank to disconnects on the missile skin. Stage II oxidizer tank piping consists of a flexible hose from the bottom of the tank to a disconnect on the missile skin.

Stage I fuel and oxidizer and Stage II fuel autogenous system piping consists of pressure fittings, flexible hoses, rigid piping, and rupture discs. Stage II oxidizer tank does not have autogenous system piping. The rupture disc prevents propellant vapors from entering the autogenous system piping.

Fixed equipment consists of stationary propellant system equipment in the launch complex. Fixed equipment is located at fuel and oxidizer hardstands, fuel and oxidizer vent areas, and in the control center and launch silo.

C6.2.5 <u>Propellant System Safety Requirements</u>

Additional requirements for fuel handling and transfer are outlined in AFOSH Standard 161-13 (Attachment 7 for Titan II). These requirements include establishing temporary restricted areas, posting of caution signs, and additional requirements for support agencies. Permissible exposure limits for missile propellants are contained in AFOSH Standard 161-8. Where the M26A1 is used as Category V protection in a confirmed A-50 environment, the canister will be disposed of at the end of the duty shift. For instances where this device is planned for use as protection against NDMA, the medical service will ensure that the users have been properly trained on all aspects of the documented respiratory protection program.

Spills or small puddles of fuel in the silo should be wet down with water instead of being allowed to evaporate, in order to minimize evolution of A-50 to NDMA vapor.

Discourage access to silo air exhaust for a reasonable time during and following any fuel (liquid or vapor) release in the silo and to fuel vent release point during and following venting.

The following chart depicts the minimum required weather conditions for each listed operation. These minimum conditions must exist and should be expected (forecasted) to remain throughout the operations. However, if propellant flow (load/unload) has started and prevailing weather conditions deteriorate below minimum requirements, the operation, including pressure drain of fixed system, will be completed. The 3-30 knots average wind requirements are determined by using a 10-minute recorded wind trace. The average trace during this period must be 3 knots or greater, but must not exceed 30 knots.

Task	Required Conditions
Fuel/Oxidizer Load/Unload	Negative Delta T 3 to 30 knots Wind Daylight Hours No Precipitation No Thunderstorm within 3 Miles
Pressure Drain and Purge of Fueled/Oxidizer Fixed System	Negative Delta T 3 to 30 knots Wind No Precipitation No Thunderstorm within 3 Miles
Purge Oxidizer Fixed System	No Restrictions When OVB Is Used
Rough and Final Adjust	Negative Delta T 3 to 30 knots Wind Daylight Hours No Precipitation No Thunderstorm within 3 Miles

Task	Required Conditions
Propellant Sampling	Negative Delta T 3 to 30 knots Wind Daylight Hours No Precipitation No Thunderstorm within 3 Miles
Liquid Transfer Between Trailers	Negative Delta T 3 to 30 knots Wind Daylight Hours No Precipitation No Thunderstorm within 3 Miles
Missile Oxidizer Tank Purge (Does Not Apply to Purge for Dewpoint on Purged Missiles)	No Restrictions When OVB Is Used When OVB Is Not Used the Following Conditions Apply: Negative Delta T 3 to 30 knots Wind
Missile Fuel Tank Purge (Does Not Apply to Purge for Dewpoint on Purged Missiles)	Negative Delta T 3 to 30 knots Wind No Thunderstorm within 3 Miles
Missile Oxidizer Tank Vent	No Restrictions When OVB Is Used
Missile Oxidizer Tank Vent for Component Replacement on Loaded Missiles	Negative Delta T 3 to 30 knots Wind Daylight Hours No Precipitation No Thunderstorm within 3 Miles
Missile Fuel Tank Vent for Component Replacement on Loaded Missiles/Preparation for Unloading	Negative Delta T 3 to 30 knots Wind Daylight Hours No Thunderstorm within 3 Miles
Fixed System Component Replacement	Venting Controlled To Allow Sufficient Vapor Dispersion
Connect/Disconnect or above Ground Hoses	Daylight Hours No Precipitation Vent to Allow Sufficient Vapor Dispersion

Task	Required Conditions
Component Replacement on Contaminated Trailers	Negative Delta T 3 to 30 knots Wind Daylight Hours No Precipitation No Thunderstorm within 3 Miles
Missile Tank Venting for Overpressure/ Underpressure	Venting Controlled To Allow Sufficient Vapor Dispersion
Purge Oxidizer Fixed System for Deactivation	No Restriction When OVB Is Used When OVB Is Not Used, the Following Conditions Apply: Negative Delta T 3 to 30 knots Winds

The Titan II propellants are powerful corrosive chemicals and quickly degrade most organic materials. Nitric acid is formed in various concentrations when nitrogen tetroxide mixes with moisture in the launch duct air. This combination of missile propellant leaks and launch duct humidity could cause structural damage to missile airframes, engines, and electrical or electronic components. Quick response to propellant leaks, high humidity, clean-up of propellant spills, and treatment of corrosion is required to prevent excessive equipment damage.

Facility vent quick disconnects will not remain connected to a loaded missile tank anytime a qualified Category I team is not on complex. Failure to disconnect vent quick disconnect can result in release of vapors into the launch duct because of overpressurization or component failure.

Area	Hazard
Fuel Hardstand, Fuel Vent Area, and Launch Silo	Contact with liquid fuel (UDMH) and fuel vapors must be avoided. Physical contact with liquid fuel can cause irritation of the skin and eyes. Inhalation of fuel vapors can cause severe injury to the respiratory system. Liquid fuel is particularly hazardous, since toxic effects are encountered by skin absorption as well as inhalation. Toxic effects, encountered through either inhalation or absorption, can result in death.

Area	Hazard
Oxidizer Hardstand, Oxidizer Vent Area	Contact with liquid oxidizer (nitrogen tetroxide) and oxidizer vapors must be avoided. Physical contact with liquid oxidizer can cause severe burns to the skin and eyes. Although inhalation of the oxidizer vapors may cause no immediate discomfort, death or serious delayed damage to the respiratory system can result.
Launch Silo	If fuel and oxidizer lines are disconnected at the same time, fire may result. Products of oxidizer and fuel combustion are toxic. (Refer to safety precautions for oxidizer and fueled hazards.)
Launch Silo, Fuel Hardstand, and Oxidizer Hardstand	Removing components containing fuel or oxidizer can result in injury to personnel and equipment. (Refer to safety precautions for oxidizer and fuel hazards.)
Control Trailer, Propellant Transfer System Receptacle Pit, Fuel Hardstand, Oxidizer Hardstand, Control Center, and Launch Silo	Electrical equipment can cause shocks, burns, electrocution, and fire.
Launch Silo	Fire near warhead or re-entry vehicle causes danger from high-explosive detonation.

C6.2.6 Special Safing Procedures

This section contains special procedures for safing the propellant system when leaks, spills, or other malfunctions occur. However, because of the vast number of malfunctions that could occur, only the most common situations are covered by detail procedures. Personnel involved in propellant-related operations or malfunctions must exercise three basic actions: First, safe the system and evacuate nonessential personnel from the immediate area. Second, contain vapors within the complex area and evacuate personnel from the projected corridor. Finally, ensure that personnel have been evacuated prior to implementation of corrective action directed by special procedures or the Missile Potential Hazard Team (MPHT), which procedures could result in a vapor release.

Special procedures in T.O. 21M-LGM25C-2-12 will be used by the PTS team chief for all unexpected vapor hazards that occur during PTS operations performed in the silo by personnel in Category I clothing. All topside propellant vapor hazards will be handled by the PTS team chief using T.O. 21M-LGM25C-2-12. Hazards other than propellant hazards will require MCCC to use emergency procedures contained in T.O. 21M-LGM25C-1. Emergency procedures in T.O. 21M-LGM25C-1 and special procedures in T.O. 21M-LGM25C-2-12 will not be used for LCCFC indications that normally result from maintenance activities, e.g., OXI VAPOR LAUNCH DUCT flashing on LCCFC when disconnecting Stage II oxidizer disconnect from missile. Prior to performing any action that may result from indications as described above, the MCCC will be advised. The MCCC will announce on the VSS that the KLAXON may sound and that the silo need not be evacuated unless otherwise directed. If the PTS team chief determines that a hazardous condition exists, personnel will immediately initiate the appropriate special procedures. The DMCCC notifies the Wing Command Post (WCP) of the abnormal situation and planned corrective action. The PTS team chief will report the result of team actions and advise the MCCC that the abnormal situation is corrected/controlled or that special procedures must be continued; the DMCCC notifies WCP of these results. Request for Missile Potential Hazard Team assistance is submitted to WCP, if applicable. After MCL 3252, during special procedures implemented in this technical order, the MANUAL SELECTOR switch on VDAP will be positioned as directed by the PTS team chief to isolate vapor hazard and ensure proper automatic corrective action.

If Blast Door 9 cannot be opened because of overpressure condition and Blast Damper 2 has been closed, open and close Blast Damper 2 to relieve overpressure condition.

Oxidizer propellant leaks/spills in the silo and topside present a two-fold problem. If the leak/spill occurs inside the silo and is small enough that oil/foam suppression procedures would be impractical, the oxidizer must be diluted with a minimum of 20 parts water to one part propellant by volume before neutralizing procedures can be performed. This may be accomplished by water hose, fire water, or sound suppression water at the discretion of the PTS team chief. A considerable amount of heat will be generated by the exothermic reaction of water and oxidizer. A large quantity of vapor will be emitted into the silo and above ground area when the water is introduced. The resulting vapor release must be taken into consideration by personnel calculating the Operational Toxic Corridor or other evacuation corridors/cordons. For large oxidizer leaks/spills that occur topside, foam should be used to cover the leak/spill and suppress the oxidizer vapors, and then dilution water should be applied by a gentle spray.

If the leak/spill is of sufficient quantity that water dilution would create a greater hazardous situation, oil suppression procedures may be initiated at the discretion of the MPHT.

Fuel propellant leaks/spills can be diluted with water or covered with foam to control the explosive and flammable properties of the fuel. Under no conditions will the LDAC switch on the FPCB be set to the OFF position during a fuel leak. The Vapor Detector Annunciator Panel will be monitored throughout all in-silo leak/spill situations. All personnel must evacuate the silo area immediately if fuel vapor level approaches 12,000 ppm or the rate of rise of fuel vapor concentration equals or exceeds 1,000 ppm per minute. If vapor concentration reaches 5,000 ppm and the in-silo team has been evacuated, the fire water system should be activated to suppress vapors. When the leak/spill has been contained and diluted, maintain the liquid in the flame deflector and proceed with sampling and fuel neutralization. When the leak/spill has occurred in the fuel pump room, maintain the liquid in the silo equipment area sump; following dilution, proceed with sampling and neutralization.
If the oxidizer/fuel leak/spill has occurred in the pump room, contain and dilute the leak/spill using sump sprays CV-2E-010-0 and CV-2E017-0 on silo equipment area, Level 8.

C6.2.7 Autogenous Pressurization System

Autogenous pressurization systems (Fig. C6-9) are used for the inflight propellant tank pressurization of both stages of the LGM-25C missile. Stage II incorporates a fuel tank autogenous pressurization system only. Immediately after propellant loading of the missile, the Stage II oxidizer tank is pressurized with nitrogen and sealed. No additional inflight pressurization is required. The Stage I autogenous pressurization system consists of a fuel tank pressurization system and an oxidizer tank pressurization system. After loading, the Stage I fuel and oxidizer tanks and the Stage II fuel tank are pressurized and sealed. The fuel tank pressurization system consists of a gas cooler, hot gas bypass orifice, flow control orifice, sonic nozzle, burst diaphragm, and connecting tubing. The oxidizer tank pressurization system consists of a super-heater, oxidizer bypass orifice, cavitating venturi and filter, flow control orifice, burst diaphragm, and connecting tubing.

The fuel tank autogenous pressurization systems used on both stages of the LGM-25C missile are identical. Both systems cool the hot exhaust gas from the gas generator from +1600 degrees F to +200 degrees F in the gas cooler. The cooled exhaust gas is used to maintain the required fuel tank pressure of 24-29 psia on Stage I and 49-54 psia on Stage II. Gas enthalpy control is provided by installing orifices of proper size in the bypass lines of the gas cooler. The amount of gas fed to the fuel tank is regulated by the use of a sonic flow control nozzle installed in the line between the gas cooler and the fuel tank. When pressure within the system reaches approximately 300 psig, the burst diaphragm, installed upstream of the gas cooler, ruptures, allowing flow of cooled exhaust gases to the missile fuel tank.

The oxidizer tank autogenous pressurization system uses hot gases from the turbine exhaust to vaporize oxidizer obtained from the discharge side of the oxidizer pump; this is accomplished by a super-heater. When pressure within the system reaches approximately 300 psig, the burst diaphragm, installed downstream of the super-heater, ruptures, allowing flow of vaporized oxidizer to the oxidizer tank to maintain the required inflight pressure of approximately 24-29 psia. A cavitating venturi, installed in the oxidizer supply line to the super-heater, maintains a constant oxidizer flow rate. Control of the gas enthalpy is accomplished by a bypass orifice in the center-tapped portion of the super-heater. The size of this orifice determines the amount of partially heated liquid to be used as the cooling agent for the hot vaporized gas passing through the upper portion of the super-heater. The amount of gas flowing to the oxidizer tank is regulated by a backpressure orifice installed in the line between the super-heater and engine. A burst diaphragm installed in the oxidizer tank supply line prevents tank pressurization loss during storage and engine start.

C6.2.8 Airborne Hydraulic System

The airborne hydraulic system (Fig. C6-10) supplies pressure to gimbal the thrust chambers of the rocket engines, roll control nozzle, and vernier rocket motors. Each stage has a closed-loop system consisting of hydraulic pumps, accumulators, hydraulic filters, actuators, and associated plumbing to produce the required hydraulic pressure. Each stage has a separate electric motor-driven pump to supply hydraulic power when the rocket engines are not firing.

The Stage I hydraulic system provides hydraulic pressure to the Stage I servo actuators. Prior to flight, the Stage I hydraulic system is powered by an electric-motor-driven pump that receives ground electric power through an umbilical connector. During flight, this system is powered by a pump driven by a turbine pump assembly on the Stage I engine. Hydraulic pressure generated by the Stage I pump is used by the servo actuators to gimbal the thrust chamber during Stage I flight.

The Stage II hydraulic system provides hydraulic pressure to the Stage II engine, roll control, and vernier actuators. During sustainer engine firing, the hydraulic system is powered by the turbine pump-assembly-driven pump. At sustainer engine shutdown, the electric-motor-driven pump starts and supplies hydraulic power to the vernier rocket motor actuators. The electric motor receives power from the vernier hydraulic power supply (VHPS) battery located in the Stage II equipment compartment. The battery power switch is closed by a signal from the Stage II sustainer engine shutdown relay during flight.

C6.2.9 Airborne Electrical System

The airborne electrical system is composed of the accessory power supply (APS) battery, the VHPS battery, motor-driven switches, relays, hydraulic pump motors, wire distribution, connectors, disconnects, and distribution buses. Miscellaneous electrical equipment includes such items as umbilical connectors, diodes, resistors, timers, terminal boards, and wiring. The electrical system is monitored and verified for operational readiness before launch by the LCS.

During a controlled flight, electrical power is supplied by the APS battery and the VHPS battery located in the guidance bay of Stage II. The APS battery is activated by the LCS during launch countdown and supplies 28-Vdc power to missile equipment during launch and flight. The VHPS battery is also activated by the LCS during launch countdown and supplies 28-Vdc power to Stage II hydraulic pump motor, relays, and motor-driven switches. Power from the VHPS battery is routed through the VHPS power switch, which is closed by a signal from the Signal II engine shutdown control relay. Fig. C6-11 displays the sequence of basic functions of the missile electrical system during launch countdown and flight.

C6.2.10 Explosive Components

Explosive components are devices used within the missile to produce thrust, supply ignition, permit missile staging, and activate components. Explosive components include gas pressure generators, vernier rocket motors, pitch rockets, igniters, and initiators. Gas pressure generators are used to open Stage I and Stage II propellant storage valves, release staging nuts, and terminate vernier rocket thrust. Two solid-propellant vernier rocket motors are mounted inside the Stage II engine compartment. The pitch rockets are two solid-propellant rockets mounted in the between-tanks area of Stage II, equidistant from one another. Pitch rocket nozzles are protected by external fairings ejected by the rocket blast at ignition. Initiators are used to start the vernier rocket motors and activate engine gas pressure generators. Igniters are used to start pitch rockets. The APS and VHPS batteries are activated by squibs sealed inside the battery cases.

An ordnance safety switch (OSS) locks out all electrical signals to explosive components except signals to Stage I storage valve gas pressure generators. A signal from the LCS closes the OSS, activating the missile batteries and Stage II storage valve cartridges, and closes the Stage I engine start switch, igniting Stage I engine gas pressure generators. Upon receiving a signal from the thrust chamber pressure switches, the launch control set ignites the hold-down nut cartridges to release the missile. During missile flight, explosive components receive initiation signals from the missile guidance computer.

Another explosive component on the missile is the butterfly valve lock (BVL); however, unlike the other explosive components, the explosive charges within the BVL have no function during a missile launch. The purpose of these charges is to prevent unauthorized entry into the BVL for the purpose of enabling an unauthorized launch.

C6.2.11 Flight Control System

The flight control system (Fig. C6-11) consists of the Stage I rate gyro package, Stage II autopilot system, and servo actuators. The flight control system receives guidance steering signals from the missile guidance set; converts them into stabilized control commands which are transmitted to the appropriate system. Most of the flight control system is located in the Stage II between-tanks area; it contains circuitry necessary to receive pitch, roll, and yaw steering signals from the missile guidance set. It mixes these signals with signals from the rate gyros, and transmit the resulting signals to the appropriate servo actuators to control missile attitude about the pitch, roll and yaw axes. In addition, the autopilot system contains the Stage II rate gyros and an inverter for supplying power to the magnetic amplifier and rate gyros. The servo actuators convert autopilot control signals into mechanical movement for positioning of the Stage I and Stage II rocket engines, the roll control nozzle, and the vernier rocket motors. Each actuator is a self-contained position servo mechanical piston feedback to balance actuator position. During Stage I flight, the guidance signals are mixed with the Stage I and Stage II rate gyro signals and transmitted to the Stage I engine actuators. A gain-change-discrete signal is also sent from the missile guidance set to the flight control system. When a staging signal is received from a thrust chamber pressure switch, the Stage I rate gyro signals are disconnected from the autopilot circuitry. During Stage II flight, the guidance signals are mixed with the Stage II rate gyro signals and transmitted to the Stage II engine actuators. During vernier flight, the signal flow is identical to Stage II flight.

C6.2.12 Missile Guidance Set

The missile guidance set (Fig. C6-12) is located in the between-tanks sections of Stage II and consists of the IMU and the MGC. The IMU is a sensing device that feeds information into the MGC. The MGC, in turn, uses this information to perform guidance and discrete functions.

The IMU is enclosed in a pressurized rigid aluminum structure designed to allow passive cooling by radiation and conduction. It consists of an inertial reference unit and inertial measurement unit electronics. The inertial reference unit uses a four-gimbal assembly to support a rotating platform stabilized at a reference attitude. It contains the gyros, accelerometers, motors, synchros, and resolvers for stabilizing the platform and producing signals proportional to changes in missile attitude and velocity. The inertial measurement unit electronics contains all the power supplies and electronics for the stabilization loops, accelerometer loops, gyro torquing, moding, fault isolation, and temperature control. The inertial measurement unit electronics is composed of printed circuit boards, which are divided by function to simplify maintenance.

Three accelerometers and three stabilization gyros are mounted on the platform axis so that their input axes are mutually perpendicular. The instrument axes are identified as x, y, and z. When the platform is erected, the z axis is vertical while the x and y axes are in the horizontal plane. The construction of the inner gimbal assures that the z instrument axis and inner gimbal axis remain parallel at all times. However, rotation of a portion of the inner gimbal about its axis varies the angular relationship of the x and y axes to the middle and outer axes of the inertial reference unit. The inner gimbal is mechanized as a two-part assembly, the z platform and the x-y platform. During operation, the z platform is prevented from rotation around the inner gimbal axis except for rotation due to earth rate and gyro drift. A synchronous motor, mounted on the z platform, drives the x-y platform around the inner gimbal axis at a constant rate of 1/4 revolution per minute with respect to the z platform. Platform rotation enables separation of inertial instrument error from external influences.

C6.3 MISSION SCENARIO

C6.3.1 Missile System

The LGM-25C Weapon System consists of an inertially guided liquid-fueled airborne weapon and associated ground equipment necessary to maintain and launch the airborne weapon. The weapon system is capable of destroying enemy targets over 5000 nautical miles distant. The launch complex is designed to maintain an operational readiness condition with no outside support after sustaining an attack that destroys all nonhardened facilities. For maximum safety and effectiveness, individual launch complexes are separated by distance of seven to ten nautical miles. All in-commission missiles are maintained in a constant alert condition and may be counted down individually or simultaneously. Safety rules for the LGM-25C (Titan II) MK6 RV/MK53 Weapon System (U) are contained in AFR 122-22.

Squadron maintenance areas provide facilities for supply, administration, operations, and maintenance necessary to maintain the launch complexes in a constant state of readiness.

Target coefficients and IMU coefficients are loaded in the Missile Guidance Set (MGC) memory through the punched tape reader in the MGACG. The target coefficients are based on particular launch complex and target location characteristics such as gravity variations. The IMU coefficients are values that compensate for the varying characteristics of gyros and accelerometers within the unit. These coefficients, either established at the factory or obtained during the calibrate test, together with the attitude and velocity inputs from the unit, enable the MGC to perform velocity-to-be-gained steering computations in the later portions of flight and to accomplish the roll and pitch programs during booster flight. The resulting steering commands are dc voltages. The direction and amount of corrective movement which the missile experiences as a result of these commands depend polarity and magnitude of the commands.

The MGC goes into a flight program as soon as the IGS has progressed to the inertial mode during a launch sequence. This is called time zero and is the base for all timed functions initiated during flight by the MGC. Actual lift-off occurs several seconds later. As the missile rises vertically during Stage I flight, the MGC issues steering signals that roll the missile until the missile is aligned in azimuth with the aim point. When azimuth alignment has been accomplished, the missile is pitched toward the aim point by steering signals from the MGC. Also during Stage I flight, the MGC issues two discrete signals. Like all other discrete signals issued during the flight program, these signals are issued only after the successful accomplishment of prior programmed events. The first signal is the control-system-gain-change-discrete signal, which alters the gain of the engine actuator control circuitry. This gain change is necessary because aerodynamic resistance decreases as altitude is gained. The second signal is a shutdown-and-staging-enable-discrete signal that energizes a shutdown and staging enable relay. When the relay is energized, it applies 28 Vdc to the coil of the staging control relay. The staging control is not energized until a circuit ground is provided. When either of the Stage I engine thrust chamber pressures drops due to lack of propellant, a thrust chamber pressure switch closes and completes the staging control relay circuit by providing the ground. When energized, the staging control relay applies 28 Vdc to the Stage I staging and shutdown switch, which initiates Stage I engine shutdown, stage separation, and Stage II engine start.

Stage II engine operation begins immediately after booster shutdown. During the later portion of Stage II flight and throughout the remainder of powered flight, the MGC performs velocity-to-be-gained steering computations. The MGC compares the actual position and velocity of the missile with the position and velocity which the missile would have in a previously calculated trajectory. On the basis of this comparison, the MGC issues pitch, yaw, and roll steering commands that alter the attitude of the missile and, in effect, change its velocity along the three mutually perpendicular computational axes. When the MGC issues a sustainer-engine-cutoff-discrete signal, Stage II flight is terminated and vernier flight is initiated. During this phase of flight, the MGC continues to make fine adjustments in missile velocity. Vernier flight is then terminated by a vernier-cutoff-discrete signal.

Vernier cutoff is the beginning of ballistic flight. From this point, the RV, after separation from Stage II, pursues a trajectory that will land it on target. Just before separation, the RV receives a prearm-discrete signal from the MGC, which permits arming of the warhead at the proper time. This signal is followed by an RV release-discrete signal that separates the RV from Stage II. After separation, the RV falls free while the remaining Stage II section is sent into a different trajectory to further minimize detection of the RV. This is accomplished by firing pitch rockets. Firing of these rockets is accomplished by two computer-discrete signals, namely, ignite-pitch-rocket-1 and ignite-pitch-rocket-2.

C6.3.2 Re-Entry Vehicle

(For classified information on the RV, refer to the 11N series Technical Orders.)

C6.3.3 Launch and Alert Operations

This section consists of launch and alert operations. Launch operations include procedures for launching a missile and returning the launch complex to a hardened condition. Alert operations include standard operating procedures necessary for normal shift operation. Amplified checklists are provided for those functions that must be performed as a demand-response procedure. Procedural tables are provided for functions that may be performed without direct reference to the respective procedure. Where appropriate, procedures for performing some routine functions have been provided in the descriptive paragraphs below.

Launch operations include EWO or peacetime launch procedures and post-launch shutdown and facility hardening procedures.

The Launch checklist covers operations based on receipt of a valid execution message from JCS, which executes the missile. After the message has been received, acknowledged, and processed, the operate code word is entered into the CSS at the BVLC, and the launch sequence is initiated and monitored by the missile combat crew.

The Post-Launch checklist consists of actions required to shut down equipment and harden the launch complex facilities. The procedure, performed as soon as liftoff is verified, involves removal of operating power and closing of the silo closure door to afford maximum protection to the launch crew. Unit disaster control directives will be followed for crew survival after launch.

Alert operations include: daily entry and shift verifications, crew changeover, briefings, launch complex surveillance, missile and launch verifications, and most other operating procedures required for normal shift operation.

C6.3.4 Checklists

Emergency checklists are the actions that must be performed to ensure safety of personnel, prevent damage to equipment, and ensure launch capability. These checklists are designated as launch emergency checklists and hazard emergency checklists.

Emergency checklists are listed in demand-response format. The STEP columns of the checklists indicate the sequence in which steps are performed.

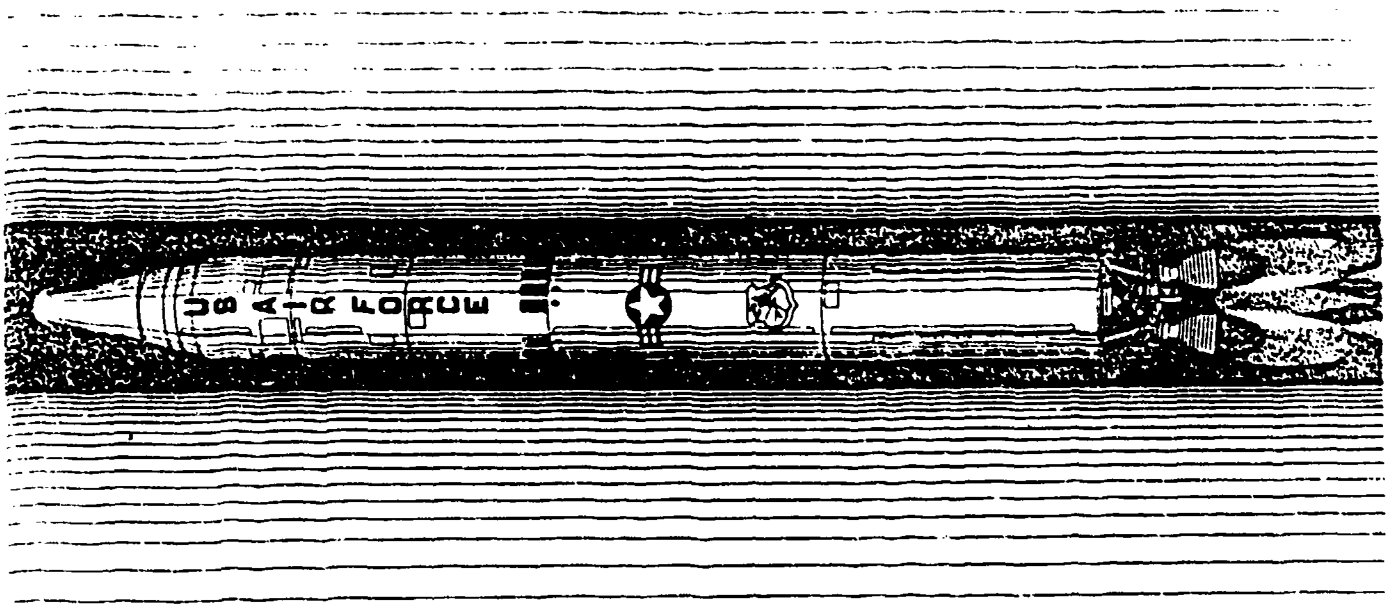
The MCCC from the LCCFC directs the actions of the DMCCC, BMAT, and MFT in performing emergency checklists. The MCCC performs all required actions unless otherwise specified in the checklists. The MCCC dispatches personnel to other parts of the launch complex, as required, to perform corrective actions or to verify or estimate hazard conditions. Personnel will enter hazardous areas only at the direction of the MCCC and will be briefed by him before entering such areas.

Launch emergency checklists are the actions required to return equipment to a safe condition, and provide entry into malfunction isolation and maintenance checklists. Checklists indicate actions for determining whether a hold exists, reporting the malfunction to the command post, and entering into malfunction isolation. Holds are conditions that stop the launch sequence after operate power is on (CMG-1) and before the ABORT or LIFT-OFF indicators light. Normal holds are indicated by red indicators located in the launch control and monitor block of the LCCFC.

When a hold condition is indicated during a launch countdown, the MCCC immediately refers to the Launch Sequence Hold checklist. If any indicators necessary for malfunction analysis fail to light during lamp test, replace the bad indicator prior to shutdown.

When an ABORT occurs, the MCCC immediately refers to the Launch Sequence Abort checklist.

Hazard emergency checklists are the actions required to eliminate hazardous or possible launch degradation conditions within the weapon system. The checklists permit the missile combat crew to monitor and control the hazard until support base personnel arrive. Hazard emergency checklists are classified in three categories: propellant vapor hazards, fire hazards, and miscellaneous hazards.



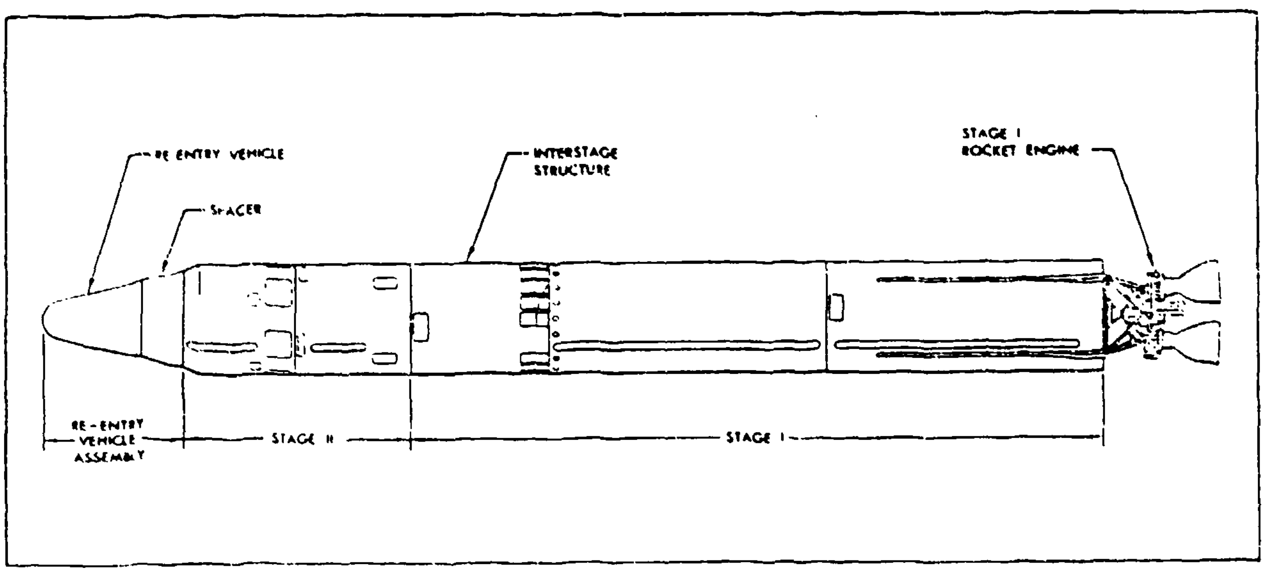


Figure C6-1 LGM-25C Missile

COMPONENT	DIMENSION
Stage I length	67 feet
Stage II length	29 feet
Re-entry vehicle length (including spacer)	14 feet
Stage I diameter	10 feet
Stage II diameter	10 feet
Re-entry vehicle diameter (at missile interface)	8-1/3 feet
Stage I weight (less propellants)	9522 pounds
Stage I weight (with propellants)	267,300 pounds
Stage II weight(less propellants)	5073 pounds
Stage II weight(with propellants)	62,700 pounds
Stage I engine thrust	430,000 pounds (sea level)
Stage II engine thrust	100,000 pounds (250,000 feet)
Vernier thrust (silo)	950 pounds

Figure C6-2 Missile Characteristics

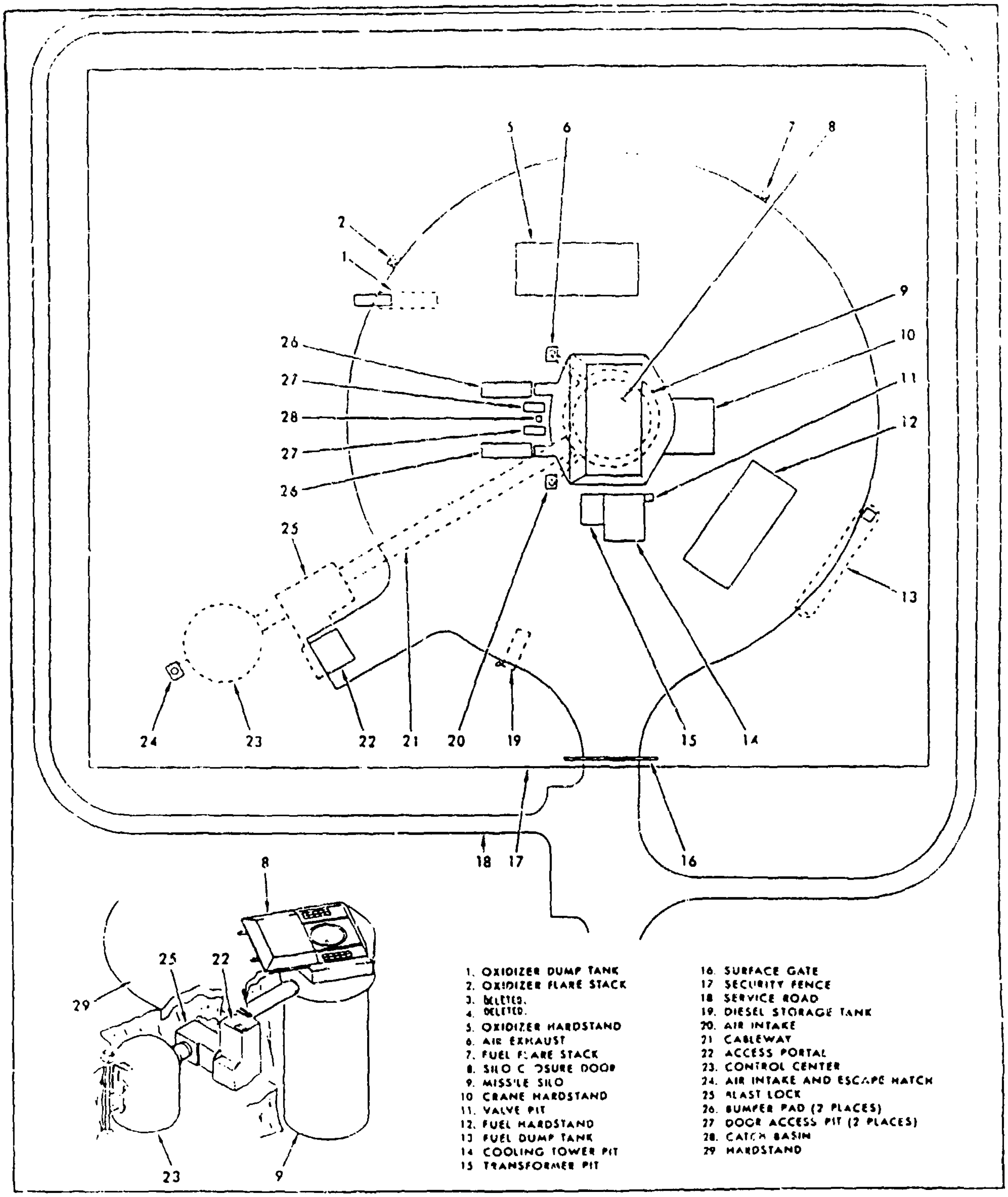


Figure C6-3 Launch Complex

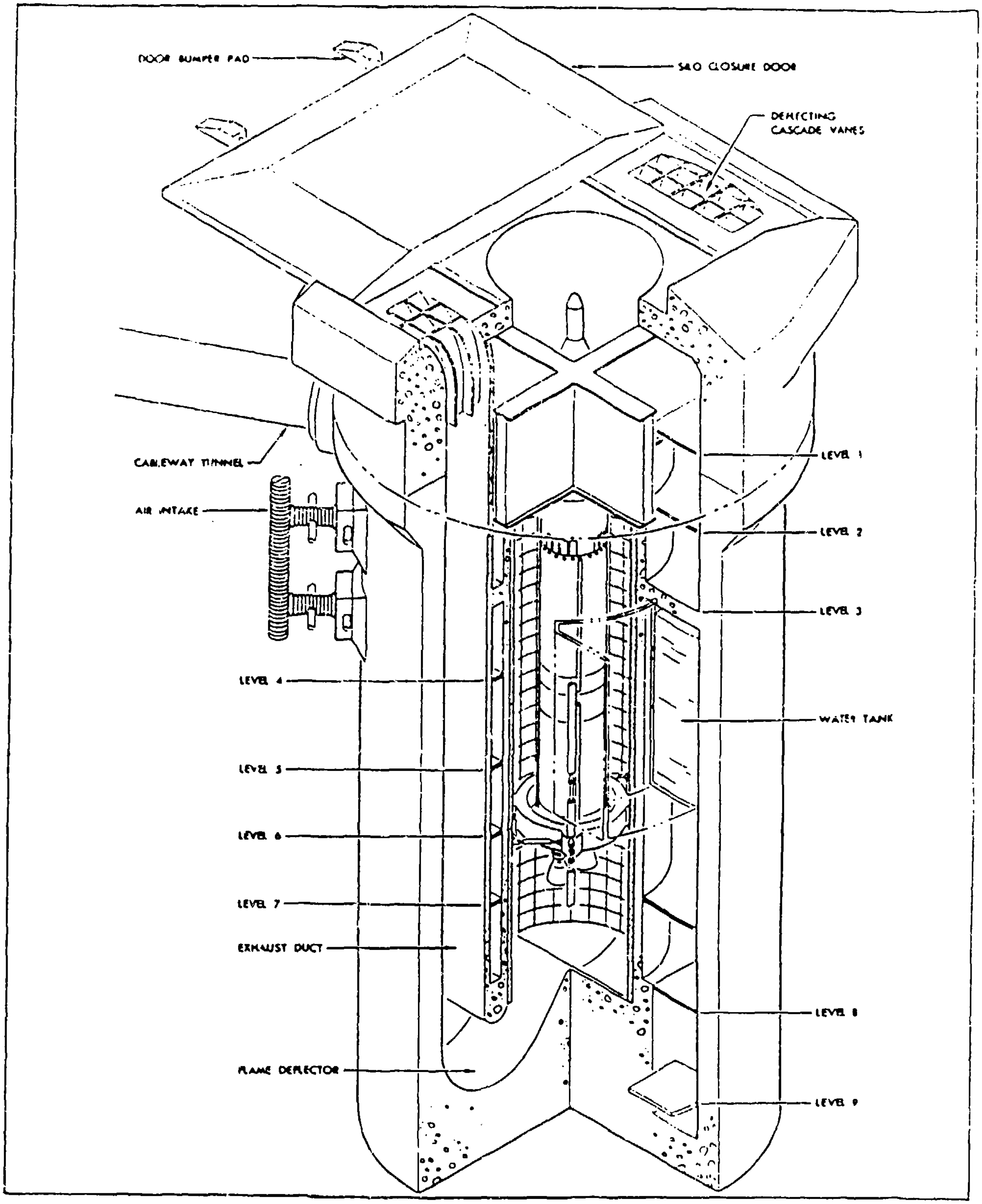


Figure C6-4 Missile Silo

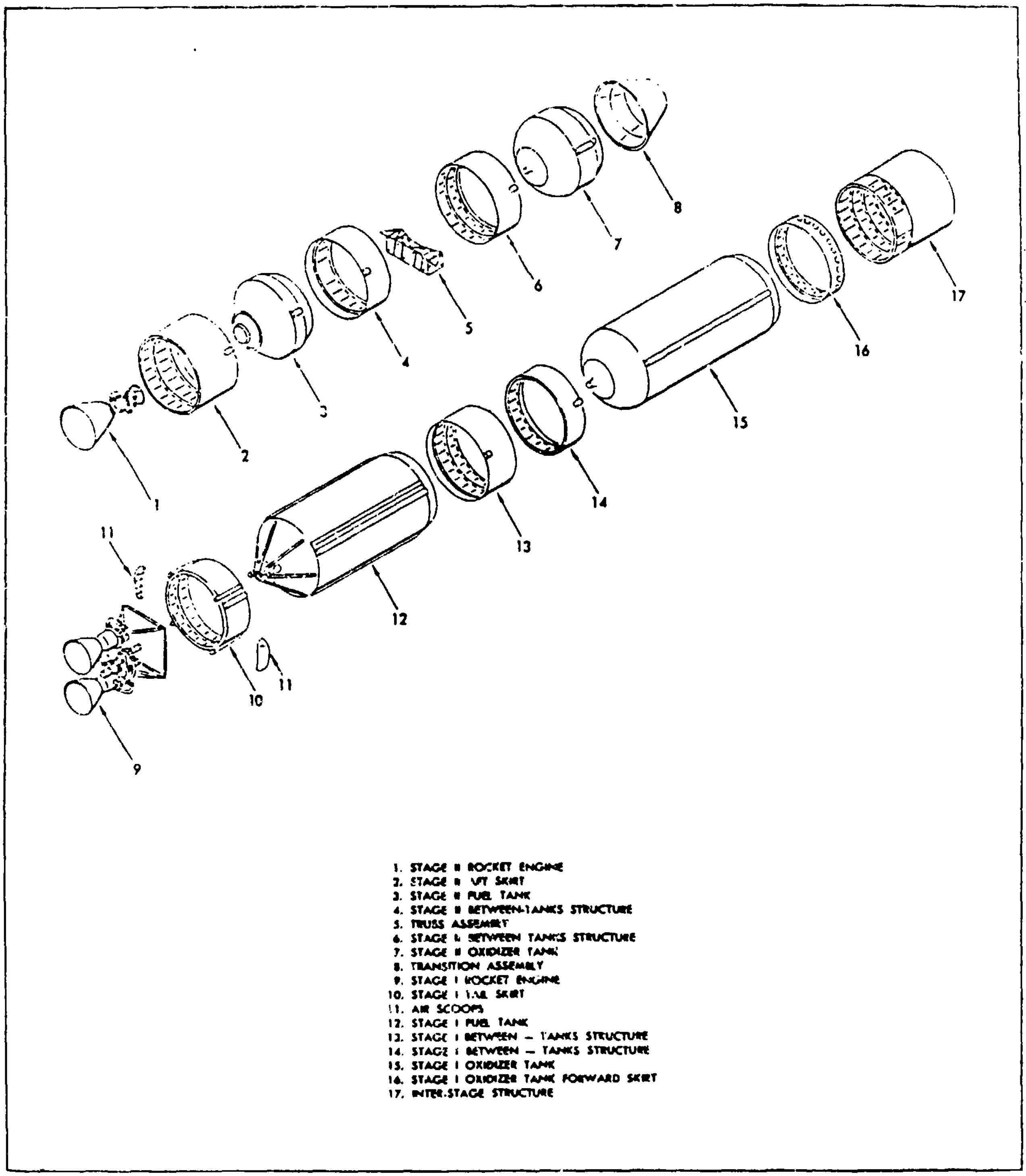


Figure C6-5 Missile Airframe, Stage I and Stage II

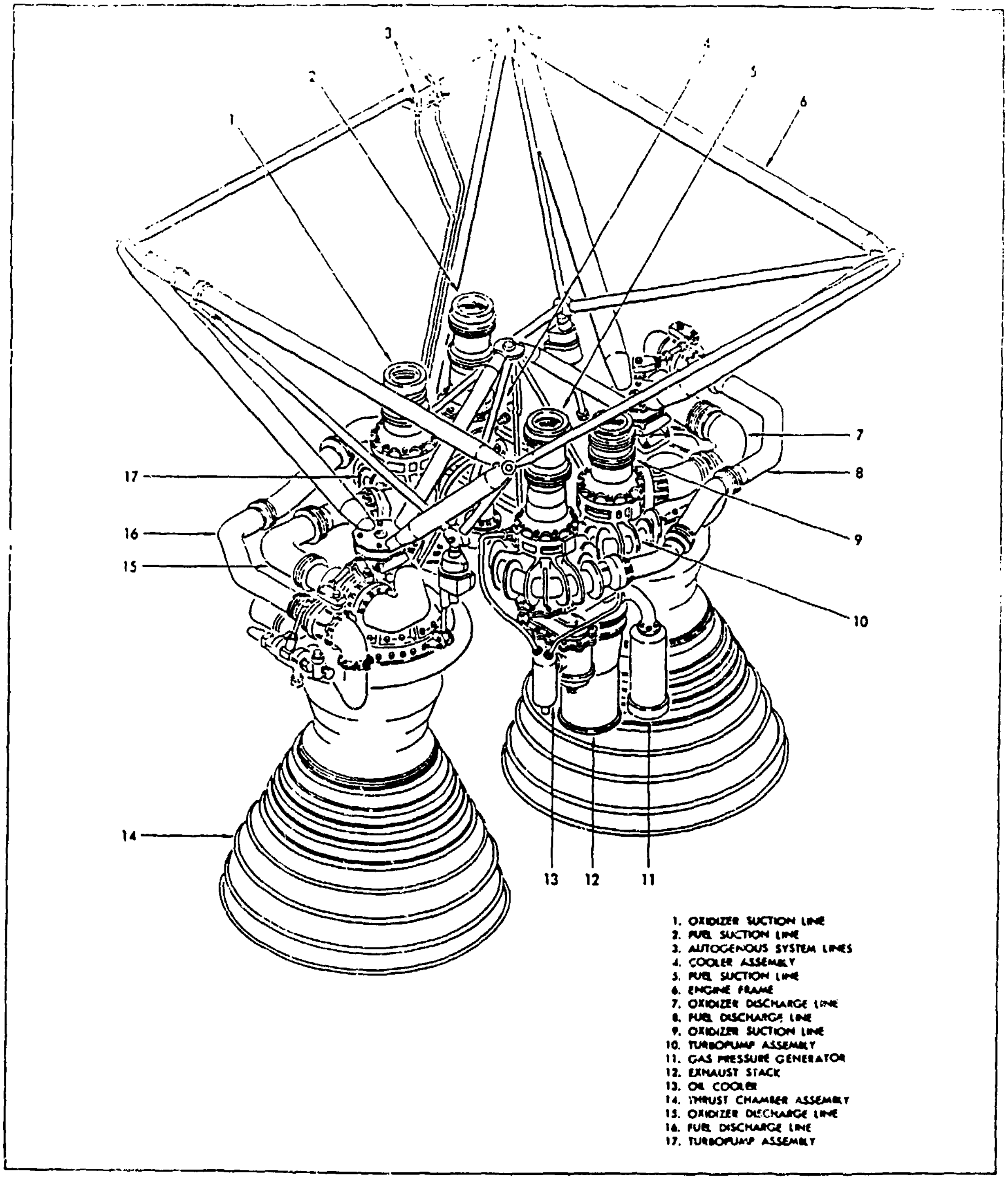


Figure C6-6 Stage I Rocket Engine

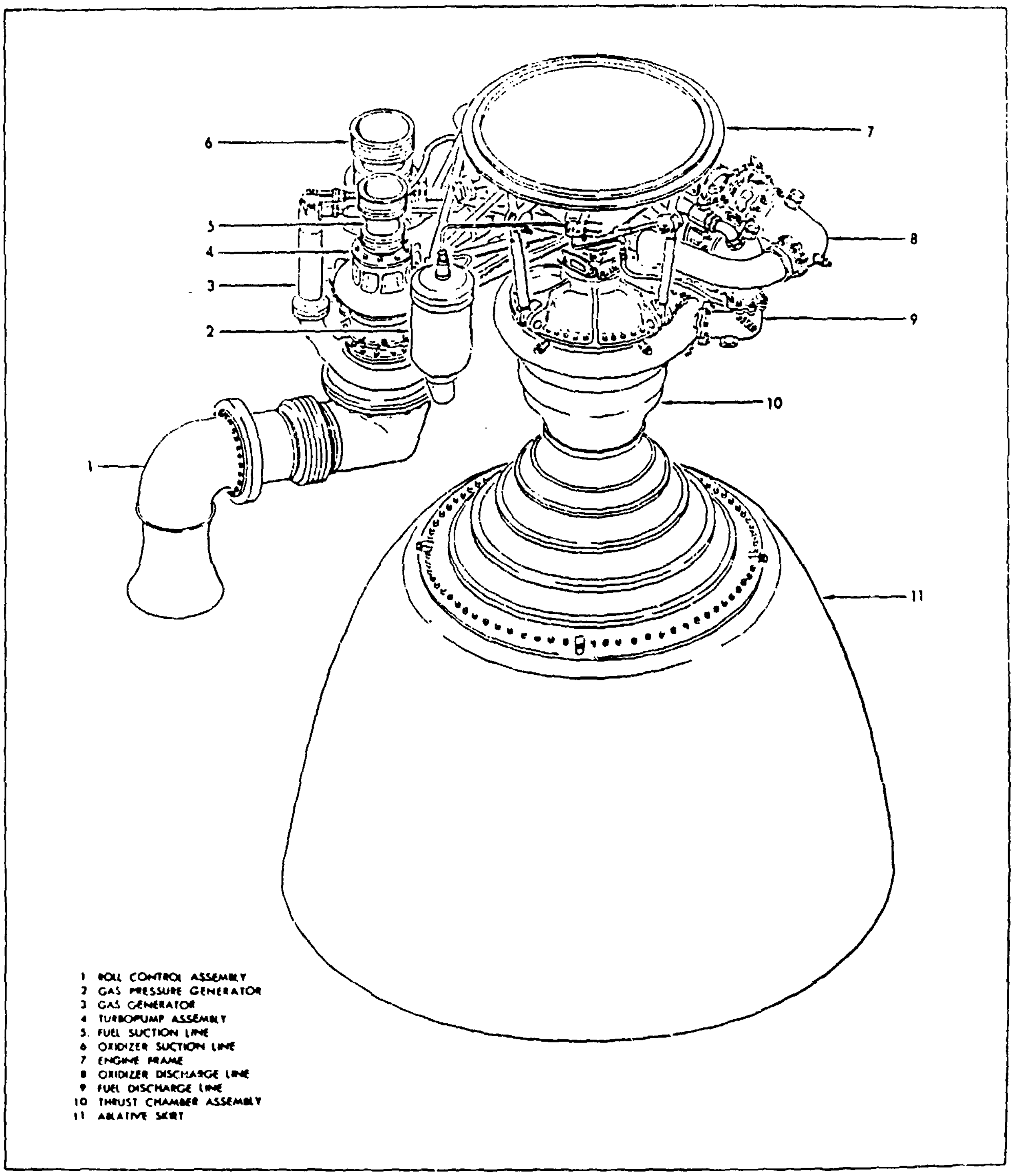


Figure C6-7 Stage II Rocket Engine

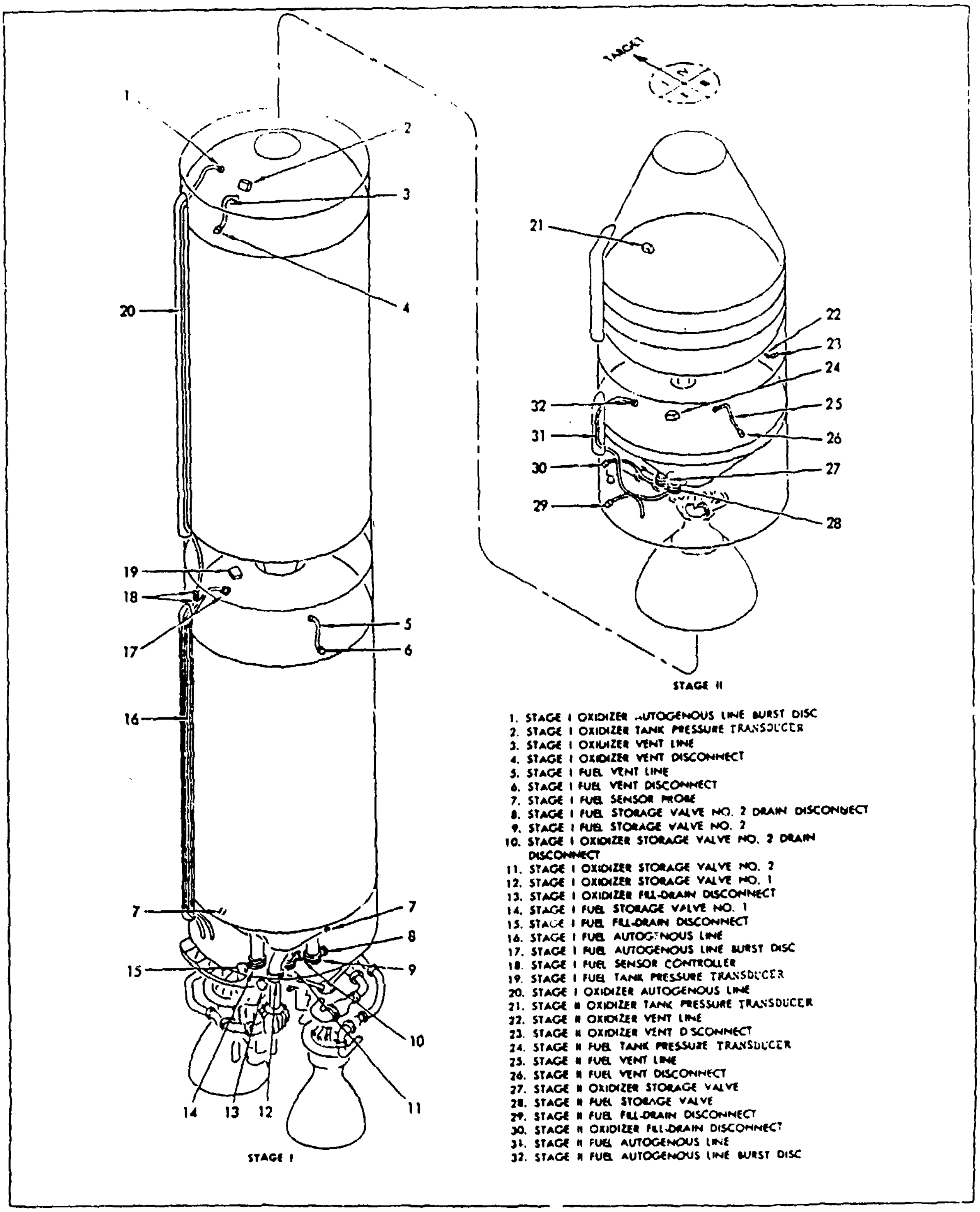


Figure C6-8 Airborne Propellant System

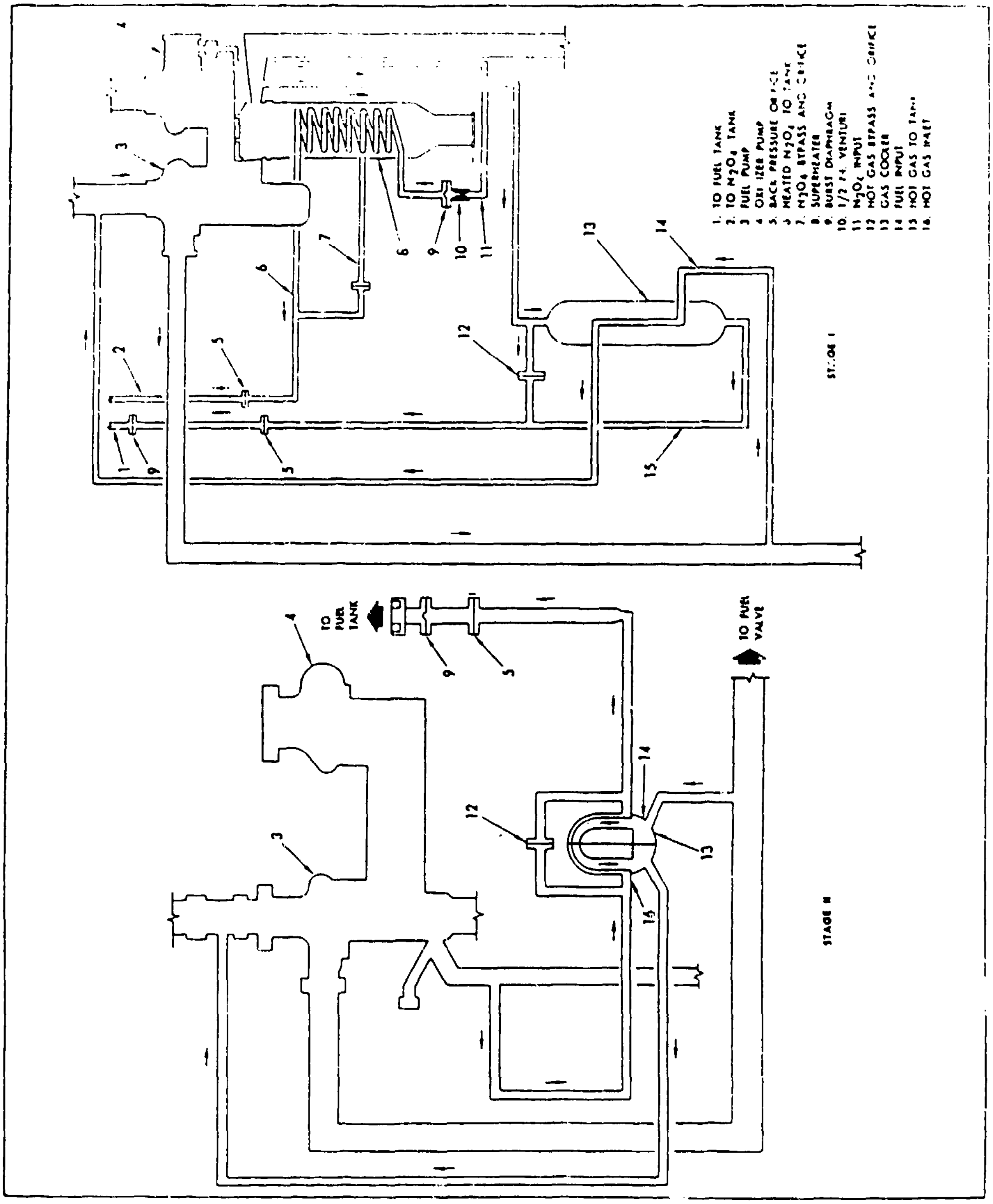


Figure C6-9 Autogenous Pressurization System Fu˙ ɔnal Diagram

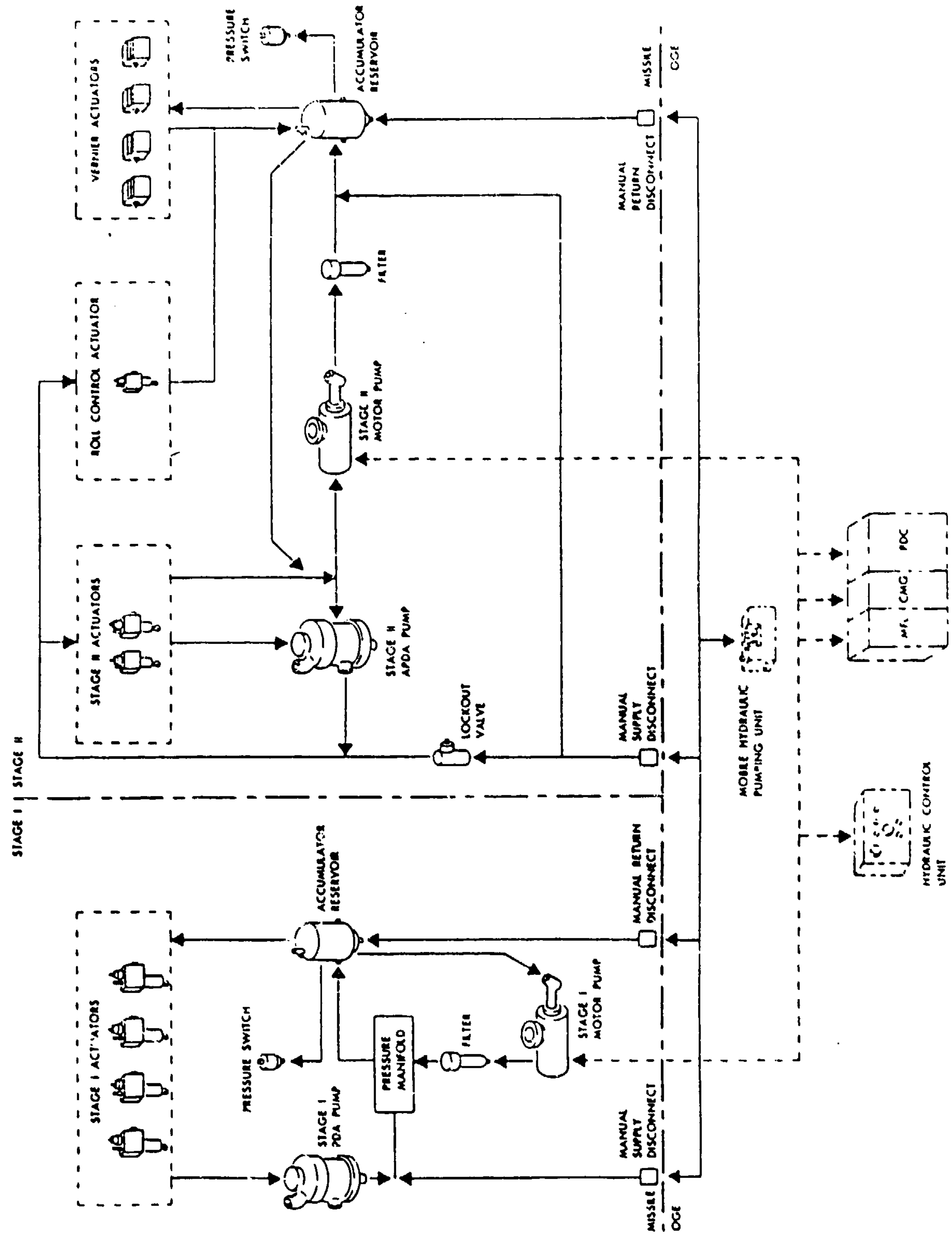


Figure C6-10 Airborne Hydraulic System Functional Diagram

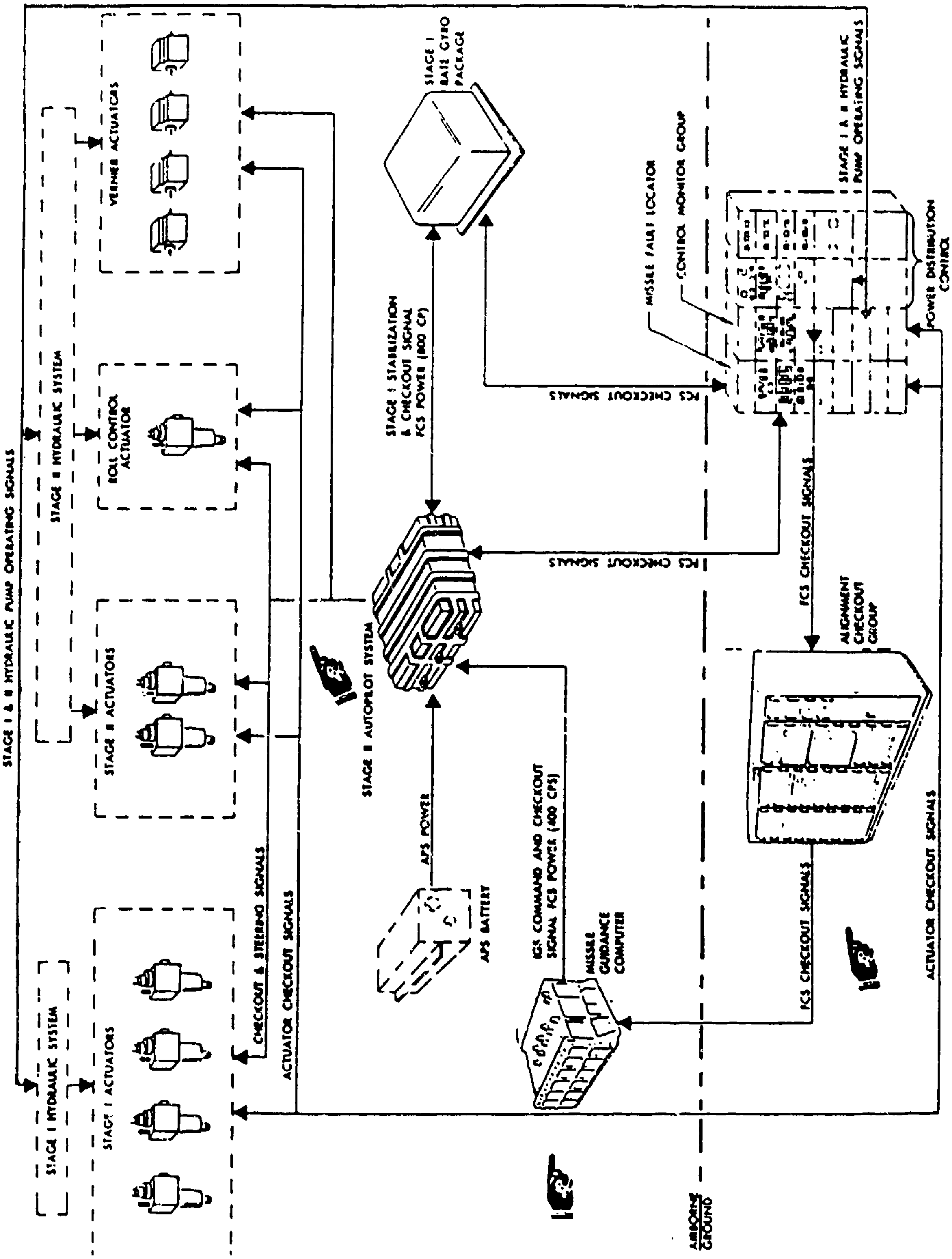


Figure C6-11 Flight Control System Functional Diagram

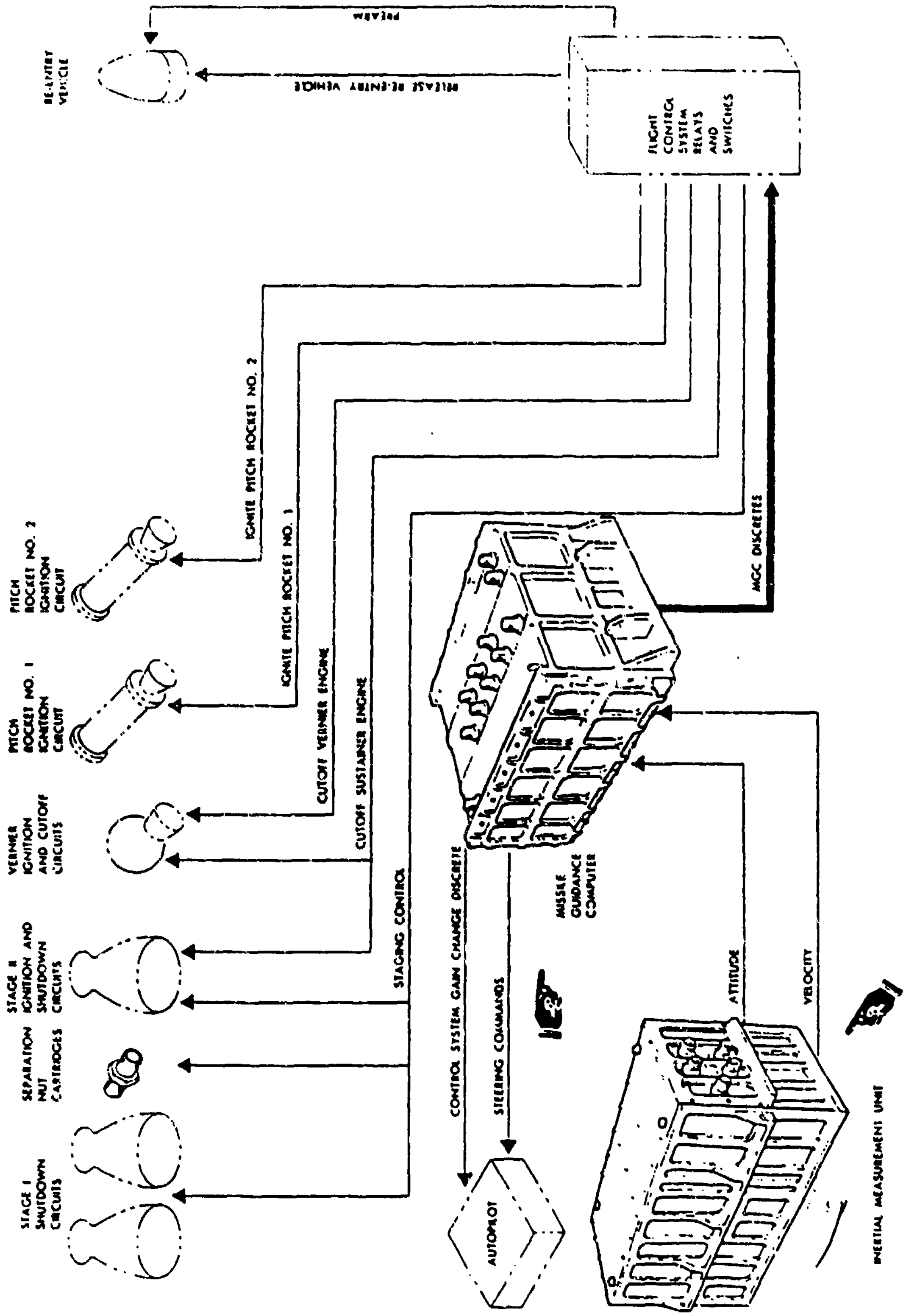


Figure C6-12 Missile Guidance Set Functional Diagram

Appendix C7
Scout

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APPENDIX C7
SCOUT

C7.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref 242.

C7.1 GENERAL DESCRIPTION

The Scout vehicle is a solid-fuel four-stage guided rocket capable of boosting orbital, probe, or re-entry space research payloads of varying sizes. Vehicle equipment includes an ignition system, an automatic and command destruct system, a preprogrammed guidance and attitude control system, a telemetry system, a payload separation system, and a tracking system. Standardized design and component interchangeability permit a variety of vehicle stage configurations to meet a broad range of mission requirements. At liftoff, the vehicle weighs approximately 45,000 lb and is approximately 73 feet long.

The four-stage vehicle as shown in Figure C7-1, consists of a finned Base A assembly at the aft end, four rocket motors, three transition sections connected alternately in tandem, and a payload separation assembly at the forward end. Each section is divided into upper and lower parts at the aft end of the adjacent motor nozzle, and the dividing planes between these parts are the separation planes for each vehicle stage. Frangible diaphragms, which are ruptured by motor blast to provide stage separation at ignition, join Stages 1 and 2 and Stages 2 and 3. Stages 3 and 4 and Stage 4 and the payload separator are joined by clamps secured with explosive bolts that separate when the bolts are ignited.

The vehicle first stage consists of a Base A, a first stage motor, and lower B section. The second stage consists of an upper B section, a second-stage motor, and a lower C section. The third stage includes an upper C section, a third-stage motor, and lower D section. The fourth stage consists of an upper D section, a fourth-stage motor, and a payload separation system consisting of a payload adapter and a fourth-stage module. Systems equipment is installed in the transition sections and Base A. An ejectable fiberglass heatshield equipped with a steel nose cap is attached to the forward end of lower D payload separator and the payload. Fiberglass tunnel covers attached to the exterior of the vehicle cover the connecting wiring between stages and the externally mounted destruct system components.

For specific missions, such as highly elliptical or solar orbits, a standard fifth stage is available and consists of an F transition section, a fifth-stage motor, and a payload adapter. A fourth- and fifth-stage attitude correction system (ACS) is also available for improved accuracy and performance.

C7.2 SYSTEM DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C7.2.1 Ignition System

First-stage ignition is obtained from an external power supply upon command from a launch sequencer. Except for electro-explosive devices (EEDs) in the payload separator, initiation of all EEDs in the first four stages of the Scout vehicle is accomplished by programmed commands from the guidance system timer. Two 30 volt batteries provide ignition power to the dual (redundant) ignition systems. These batteries are dry-charged silver zinc remotely activated units.

C7.2.2 Destruct System

A command destruct system provides thrust termination. Two linear-shaped charges are mounted longitudinally along each of the first three-stage motors (Algol, Castor, and Antares) to provide thrust termination by case rupture upon initiation of destruct action. Ignition circuits for the fourth stage are also interrupted by the detonation of the third-stage linear-shaped charges.

An autodestruct system is initiated by premature separation. Destruct action occurs automatically on the first and second stages in the event of premature separation of the first and second or second and third stages. Stage separation is sensed by switches located in first- and second-stage wiring tunnels. These switches operate when the lanyards routed across the separation planes are pulled at separation. If separation occurs prior to motor ignition or during burning, a headcap pressure switch in series with the lanyard-operated switches allows destruct to occur through closed contacts. During normal operation, the contacts of the pressure switch open as the motor pressure decays so that destruct will not occur at stage separation.

The linear-shaped charges and safe-arm units used in the autodestruct system are the same ones used in the command destruct system. Power for these EEDs is provided by separate batteries located in first- and second-stage wiring tunnels.

C7.2.3 Reaction Control System

Hydrogen peroxide (90% concentration) is the propellant used in the second- and third-stage reaction control systems (Fig. C7-2 and Fig. C7-3). Each system is separate and independent. The H_2O_2 is forced through motor chambers and silver screens, which cause it to decompose into superheated steam and oxygen. The hot gases expand through a Venturi-type nozzle, producing reaction forces that are used to control the vehicle around its three major axes.

Four 40-pound roll motors, two 500-pound pitch motors, and two 500-pound yaw motors are used in B Section. Four 14/3-pound roll motors, two 2-pound pitch motors, two 48-pound pitch motors, and two 48-pound yaw motors are used in C Section. The 48-pound pitch and yaw motors are also used as retro motors to aid separation of the third and fourth stages at the proper times.

If the third stage utilizes the Antares IIB motor, all 48-pound motors are replaced by 60-pound motors.

Flow of H_2O_2 through the B and C section control motors is controlled by on-off solenoid valves, one valve being used for each motor. These valves fire the control motors in response to the guidance system error signals to keep the vehicle within predetermined deadbands around its three major axes.

During flight, the H_2O_2 system operates at a pressure of about 550 psi. This pressure is provided by gaseous N_2 stored onboard in tanks at 3000 psi. A combination on-off vent regulator valve reduces the N_2 pressure to the operating range required and provides a means of controlling the pressurization and venting of the system. Note that this valve allows venting of the low-pressure N_2 system only and does not vent H_2O_2. The H_2O_2 system in each transition section has a relief valve to protect the system in the event of overpressurization. The B section valve relieves at approximately 700 psi and the C section valve at approximately 600 psi.

The reaction control system leak tests on B and C sections are performed in the Ordnance Assembly Building (OAB) during second- and third-stage buildup before these stages are placed on the transporter. Both the nitrogen (N_2) and hydrogen peroxide (H_2O_2) systems are pressurized for these tests with N_2. No H_2O_2 is used in the OAB.

The nitrogen system tubing is tested at the factory to the following pressures:

Working Press., psi	Proof Press., psi	Burst Press., psi	Reference LTV Drawing
3000	4500	7500	23-000321

The volumes and working pressures for the steel tanks used to store N_2 at a pressure of 3000 psi in B and C sections are shown below:

Section	Tanks per Sect	Working Press., psi	Proof Press., psi	Burst Press., psi	Vol per Tank in. 3
B	2	3000	5000	6667	386
C	2	3000	5000	6667	68

The Hydrogen Peroxide System tubing is tested at the factory to the following pressures:

Working Pressure, psi	Proof Pressure, psi	Burst Pressure, psi	Reference LTV Drawing
550	825	1375	23-000321

The aluminum tanks used to store H_2O_2 in B and C sections contain a silastic bladder to separate the N_2 and H_2O_2 and to provide a means for expelling the H_2O_2 from the tank. The volumes and working pressures for these tanks are shown below:

Section	Tanks per Sect	Working Press., psi	Proof Press., psi	Burst Press., psi	Vol per Tank in. 3
B	10	550	825	1375	18.2
C	2	500	750	1250	9.1

C7.2.4 Hydraulic Control System

The hydraulic control system consists of pump, valves, tubing, reservoir, and a servo actuator.

The hydraulic pump delivers 2-gpm hydraulic flow at 3000 psi and can be electrically driven from an internal or external 28-Volt dc source.

The high-pressure relief valve will actuate at 3650-psi maximum and reseat at 3350-psi minimum.

The low-pressure relief valve is rated to provide a flow of 2 gpm at 150 psi and will reseat at 100 psi.

The reservoir has a volume of 50 in.^{3} and fill level of 40 in.^{3} for the hydraulic fluid at 3000-psi rated pressure. The extra 10 in.^{3} provides allowance for thermal expansion.

The servo actuators drive the control surface torque tubes.

Six external bleed fittings are located in the high points of the plumbing to allow purging the system of air.

The test pressures for the lines used in the hydraulic system are shown below:

Material & Spec	Tubing O.D. & Wall, in.	Work Press., psi	Proof Press., psi	Burst Press., psi	Ref Spec LTV
Corrosion Resistant	1/4 x 0.028	3000	6000	12000	308-12-5
Steel Type 304 Seamless	3/8 x 0.035	3000	6000	12000	308-12-5

First-stage control is attained through the use of the hydraulic servo-actuator assemblies, which position aerodynamic and jet-vane control surfaces as directed by guidance signals from the poppet valve electronics package. These signals are first amplified and modulated; then the outputs from the servo amplifier-modulators energize the servo-actuators installed in each of the four fins. Each servo-actuator displaces a set of control surfaces consisting of a movable tip located at the outer end of the fixed fin and a jet vane immersed in the exhaust of the Algol motor. Two servo-actuators located in the pitch plane are combined electrically for synchronized operation and respond to pitch error signals only. The two remaining actuators in the yaw plane respond to both the roll and yaw error signals.

C7.2.5 Payload Separation System

A standard payload separation assembly is provided with the Scout vehicle to separate the payload from the burned-out fourth stage. At fourth-stage separation, microswitches mounted on the fourth stage start timers on the fourth-stage module. After a nominal time period of five minutes, the timers fire explosive nuts on the payload adapter which release the Marman band retaining the payload. Six springs then eject the payload with a velocity of about three feet per second (for a 150-lb payload).

C7.2.6 Attitude Control System

Under certain requirements dictated by the mission of the vehicle, it is sometimes necessary to correct the attitude of the spinning upper stages of Scout through a pre-programmed maneuver after fourth-stage burn-out and before fifth-stage ignition. The addition of the Attitude Correction System (ACS) in the fourth stage provides this capability. It is a self-contained system activated by a single relay closure from the Scout third-stage guidance system.

The fourth-stage attitude correction system is mounted on the fourth-stage TM ring and within the modified "F" transition section. The ACS consists of a sensor assembly, a control electronics package, and a reaction control subsystem (RCS). Figure C7-4 presents a block diagram of the fourth stage ACS. The ACS provides pitch and yaw attitude information, control forces to provide reorientation maneuver and attitude hold capability, the necessary signal processing circuitry for these functions, and telemetry and ground readouts. A standard Scout fourth-stage module (ring) telemetry system is used for vehicle performance and is modified to transmit critical ACS parameters. Timing for the ACS functions is provided by an internal ACS timer initiated by the vehicle third-stage guidance intervalometer.

Control of the spinning fourth stage and fifth stage is provided by a single nitrogen control motor nozzle. The nozzle is fired at the appropriate roll position to provide both pitch and yaw control forces. Switching logic for firing the nozzle at the correct roll position is contained within the ACS.

The nitrogen supply reservoir for the ACS is a toroidal tank that stores 300 cubic inches at 3600 psig. It is mounted just forward of the telemetry ring. The gaseous nitrogen is routed through a pressure regulator to the nozzle.

C7.2.7 Scout Rocket Motors

Summary data for the Scout Rocket Motors are listed below.

1. Algol IIC rocket motor.

2. Algol III rocket motor.

3. Castor II rocket motor.

4. Antares rocket motor.

5. Altair II-(X258) rocket motor.

6. Altair III rocket motor.

7. Alcyone rocket motor.

8. MARC-4 Spin rocket motor.

9. IKS75 Spin rocket motor.

The following definitions will be used in the following discussions:

1. Bridge - a high resistance wire that forms the heating element in a squib.

2. Squib - an electrically initiated device that contains a pyrotechnic charge.

3. Initiator - a squib that is used to ignite an igniter.

4. Igniter - a device that heats the propellant to the ignition temperature and near operating combustion chamber pressure.

C7.2.7.1 <u>ALGOL IIC Rocket Motor</u> - The Algol IIC rocket motor, designated by Aerojet-General Corporation as the 45KS-100,000, is a high-thrust, medium-duration motor. The 45KS-100,000 may be broken down as follows:

45	Duration of thrust, in seconds, at sea level at 70°F
K	Oxidizer and fuel code symbol
S	Solid propellant charge
1000,000	Thrust, in lb, at sea level at 70°F

The rocket motor's major components are a chamber, a single propellant grain, a fixed nozzle, and an igniter.

The chamber consists of a thin-walled elongated steel cylinder with semi-elliptic forward and aft closures. An igniter boss in the forward closure is provided for installation of the igniter. The aft end of the chamber is r 'hined to accept a light-weight nozzle. Welded skirts, forward and aft, have 24 holes for attachment of lifting rings. The skirts also serve as a means of attaching vehicle transition sections.

The propellant charge is an internal-burning, case bonded, polyurethane propellant grain cast in a four-point star core configuration. This grain is bonded to a chamber linear that is bonded to the chamber. Propellant weight is 21,165 pounds. The TNT equivalent weight is 1,480 lb based on 7% propellant weight.

The nozzle, which has a machined graphite/phenolic thrust insert, is constructed of composite materials (steel-glass-epoxy). It is 40.01-in. long and has an exit 33.66-in. diameter. Thirty-six equally spaced bolts secure the nozzle to the aft end of the chamber.

The igniter is sealed into the forward closure with a packing and secured with a snap-ring. It consists of an adapter, two single bridge-wire initiators, a basket, an initiator charge, a booster charge, and a main body charge. Igniter propellant weight is 5.3 lb.

The igniter electrical specifications are:

1.	Bridge Wire Resistance	1.05 ± 0.10 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	1.0 A for 5 min.
	b. Minimum All Fire	3.5 A
	c. Recommend Fire	5.0 A

Storage data for the Algol IIC motor include:

1.	Temperatures	
	a. Motor Storage Range	50°F to 90°F
	b. Igniter Storage Range	30°F to 100°F
	c. Initiator Storage Range	65°F to 160°F
	d. Propellant Auto-Ignition	450°F
	e. Igniter Auto-Ignition	995°F
	f. Flame	548°F
2.	Storage Time Limits	
	a. Motor	5 yr
	b. Igniter	3 yr
	c. Initiator	5 yr

C7.2.7.2 ALGOL III Rocket Motor - The Algol III rocket motor is a high-thrust, medium-duration motor built by United Technology Center.

The rocket motor's major components are a chamber, a single propellant grain, a fixed nozzle, and an igniter.

The chamber consists of a thin-walled elongated steel cylinder with semi-elliptic forward and aft closures. An igniter boss in the forward closure is provided for installation of the igniter. The aft end of the chamber is machined to accept a light-weight nozzle. Welded skirts, forward and aft, have 24 holes for attachment of lifting rings. The skirts also serve as a means of attaching vehicle transition sections.

The propellant charge is an internal-burning, case bonded, polybutadiene-acrylic acid-acrylonitrile terminated propellant grain cast in a four-point star core configuration. This grain is bonded to a chamber linear that is bonded to the chamber. Propellant weight is 27,986 lb. The TNT equivalent weight is 1,959 lb based on 7% of the propellant weight.

The nozzle, which has a machined graphite/phenolic composite insert is constructed with a steel housing, silica/phenolic forward insulator, and compound carbon/phenolic and silica/phenolic exit cone. It is 40.01 inches long and has an exit 33.66-inch diameter. Thirty-six equally spaced bolts secure the nozzle to the aft end of the chamber.

The igniter is sealed into the forward closure with a packing and secured with a snap-ring. It consists of two single bridge-wire initiators, an initiator charge, and a main body charge. The igniter propellant weight is 2.3 lb.

The igniter electrical specifications are:

1.	Bridge Wire Resistance	1.05 ± 0.10 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	1.0 A for 5 min
	b. Minimum All Fire	3.5 A
	c. Recommend Fire	5.0 A

Storage data for motor include:

1.	Temperatures	30°F to 100°F
	a. Motor Storage Range	30°F to 100°F
	b. Igniter Storage Range	-65°F to 160°F
	c. Initiator Storage Range	683°F @ 30s
	d. Propellant Auto-Ignition	577°F @ 30s
	e. Flame	5575°F
2.	Storage Time Limits	
	a. Motor	3 yr
	b. Igniter	3 yr
	c. Initiator	5 yr

C7.2.7.3 Castor II Rocket Motor - The Castor II rocket motor, designated by Thiokol Chemical Corporation as the TS-354-3, is a solid propellant motor. It consists of a motor case loaded with solid propellant (essentially a pressure chamber); insulation that bonds the propellant to the case and protects the case from heat; nozzle assembly that provides proper restriction and subsequent expansion of exhaust gases; and a pyrogen igniter system.

The chamber is a welded assembly consisting of a forward dome section, a cylindrical section, and an aft closure section. Attachment (thrust) flanges welded to the forward dome and aft closure sections are used for missile assembly. At the extreme forward end of the motor case, an opening is provided for insertion of the pyrogen igniter system, and a bolt flange is provided at the aft end for nozzle attachment.

The propellant is polybutadiene acrylic acid, ammonium perchlorate, and aluminum. Its grain configuration is a cylindrical port with two radial slots. This grain is bonded to a chamber liner that is bonded to the chamber. Propellant weight is 8.217 lb. TNT equivalent weight is 575 lb based on 7% of propellant weight.

The nozzle, a convergent-divergent type, is made of steel with pressed-in graphite inserts to form the nozzle throat. An insulation material surrounding the graphite insert acts as a heat barrier between the graphite insert and the steel body. The joint between the chamber and nozzle is sealed by a gasket and TH-L21 (Thiokol) sealing compound. The exit cones are coated with an ablative material commercially named ROKIDE. The nozzle, which has a closure that may be removed for inspection, is attached to the chamber by 32 bolts.

The igniter, designated TK-463-1, contains two 23-003793 initiators. It is a small rocket motor ignited by redundant single bridge wire initiators threaded into its head end. When ignited, the igniter, by means of its exhaust, ignites the propellant in the TX-354 motor. Igniter weight is 3.1 lb.

The igniter electrical characteristics are:

1. Bridge Wire Resistance	1.05 ± 0.1 ohms
2. Bridge Wire Current Requirements	
a. Maximum No Fire	1.0 A for 5 min
b. Minimum All Fire	3.5 A
c. Recommend Fire	5.0 A

Storage data for motor include:

1. Temperatures	
a. Motor Storage Range	30°F to 110°F
b. Igniter Storage Range	30°F to 110°F
c. Initiator Storage Range	-65°F to 160°F
d. Propellant Auto-Ignition	325°F @ 60 min
e. Igniter Auto-Ignition	300°F
f. Flame	6000°F
2. Storage Time Limits	
a. Motor	5 yr
b. Igniter	5 yr
c. Initiator	5 yr

C7.2.7.4 Antares Rocket Motor - The Antares rocket motor is a case-bonded, composite modified double-base solid propellant system. It is built by Hercules, Inc in two configurations, the 259-B3 (Antares IIA) and 259-B4 (Antares IIB). Reinforced plastic and aluminum are used for all major inert components: chamber, nozzle, and igniter.

The chamber is a filament-wound, glassfiber-reinforced epoxy resin structure with ovaloid forward and aft domes. It incorporates integrally wound forward and aft adapters of high-strength aluminum. The chamber is 30.11 in. in diameter and 76.1-in. long. Nominal wall thickness of the chamber cylinder is 0.14 in. Vehicle attachment to the chamber is accomplished with aluminum skirt rings attached to glass fiber skirts at the cylinder-to-dome transition on both ends of the chamber. The forward skirt ring contains 24 equally spaced tapped holes, and the aft skirt ring contains 48 tapped holes.

The chamber is filament-wound directly over a destructible plaster mandrel containing a forward and aft insulator of asbestos-filled Buna-N rubber. The thickness of the insulator in any particular region is proportional to the time that it is exposed to the propellant gases.

The propellant is a double-base, nitrocellulose, nitroglycerin propellant with a single perforated four-slot design with ovaloid-shaped forward and aft ends that conform to the chamber contour. It is cast into and bonded directly to the chamber and insulators. Propellant weight is 2,562 lb. TNT equivalent weight is 2,570 lb based on 100% of propellant weight.

The nozzle, a molded exit cone, has a graphite cloth phenolic liner on its inner surface and an asbestos-phenolic liner on its outer surface. It is also reinforced on the outside with filament-wound structure similar to that used in the chamber. The nozzle has an asbestos-phenolic, nozzle retainer ring with an integral aluminum attachment ring and a graphite (IIA) or graphite-phenolic (IIB) throat insert and is attached to the chamber aft adapter with 24 bolts.

The nozzle closure, an internal plug that is press-fitted and bonded in place, is fabricated from high-density styrofoam (8 lb/ft^3 density-foamed polystyrene).

The Antares II A igniter is a rocket type charged with DGV propellant. Two initiators (Hercules SD60E0) are used in ignite a 25-grain B-KNO_3 pallet initiating charge. Each initiator has two bridge wires and incorporates a 22021-8-4S connector. Ignition delay time is 1.7 $\pm$ 0.2 seconds. Igniter propellant is 7.9 lb.

The Antares IIA igniter electrical characteristics are:

1.	Bridge Wire Resistance	1.4 $\pm$ 0.4 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	1.0 A for 5 mm
	b. Minimum All Fire	3.5 A
	c. Recommend Fire	8.5 $\pm$ 3.5 A

Storage data for motor include:

1.	Temperatures	
	a. Motor Storage Range	40°F to 100°F
	b. Igniter Storage Range	40°F to 100°F
	c. Initiator Storage Range	30°F to 100°F
	d. Propellant Auto-Ignition	392°F @ 12.7 mm
	e. Igniter Auto-Ignition	350°F @ 7.7 mm
	f. Flame	6384°F

2. Storage Time Limits

a. Motor	7 yr
b. Igniter (Antares IIA)	86 mo
c. Initiator	5 yr

Changes that have been incorporated into the Antares IIB igniter include the following: DDP-80 propellant, two single bridge-wire Apollo standard initiators (SBASI) that do not incorporate a delay function, 12-gram $GKNO_3$ pellet charge, and reduction of weight by approximately 60%. Igniter propellant weight is 2.0 lb. Bridge wire resistance is 1.05 $\pm$ 0.1 ohms, and igniter storage time limit is 5 years. Initiator storage time limit is 7 years.

C7.2.7.5 <u>Altair II (X258) Rocket Motor</u> - The Altair rocket motor (manufactured by Hercules, Inc) is a composite-modified double-base solid propellant system. Reinforced plastic and aluminum are used for all major inert components; these components are a chamber, a nozzle, and an igniter.

The chamber is a filament-wound glass fiber-reinforced epoxy resin structure with ovaloid forward and aft domes. It incorporates integrally wound forward and aft adapters of high-strength aluminum. The chamber is 18.04 in. in diameter and 42.82-in. long. Nominal wall thickness of the chamber cylinder is 0.085 in. Vehicle attachment to the chamber is accomplished with aluminum skirt rings attached to glass fiber skirts at the cylinder-to-dome transition on both ends of the chamber. The forward and aft skirts rings contain 24 equally spaced tapped holes.

The chamber is filament-wound directly over a destructible plaster mandrel containing a forward and aft insulator of asbestos-filled Buna-N rubber. The thickness of the insulator in any particular region is proportional to the time that it is exposed to the propellant gases.

The propellant is double-based introcellulose nitroglycerin type of single-perforated design with four major slots and eight additional minor slots. The forward and aft ends are ovaloid to conform with the chamber interior contour. It is cast into and bonded directly to the chamber and insulator. Propellant weight is 506 lb. TNT equivalent weight is 506 lb based on 100% of propellant weight.

The nozzle, a molded exit cone, has a graphite cloth phenolic liner on its inner surface and an asbestos-phenolic liner on its outer surface. It is also reinforced on the outside with filament-wound structure similar to that used in the chamber. The nozzle has an asbestos-phenolic retainer ring with an integral aluminum attachment ring and a graphite throat insert, and is attached to the chamber aft adapter with 16 bolts.

The nozzle closure is fabricated from high-density styrofoam (8 lb/ft^3 density foamed polystyrene). It is an internal plug bonded to the graphite insert and asbestos-phenolic retainer ring with an epoxy adhesive.

The igniter is a rocket-type charged with DGV propellant. Two initiators (SD60A1) are used to ignite a tengrain B-KNO_3 pellet initiating charge. Each squib has two bridge wires and incorporates a PT06-P-8-4P connector. Ignition delay time is 6.35 ± 85 seconds. Ignite propellant weight is 2.1 lb.

The igniter electrical characteristics are:

1.	Bridge Wire Resistance	1.4 ± 0.4 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	1.0 A for 5 mm
	b. Minimum All Fire	3.5 A
	c. Recommend Fire	8.5 ± 3.5 A

Storage data for motor include:

1.	Temperatures	
	a. Motor Storage Range	40°F to 100°F
	b. Igniter Storage Range	40°F to 100°F
	c. Initiator Storage Range	30°F to 100°F
	d. Propellant Auto-Ignition	392°F @ 12.7 mm
	e. Igniter Auto-Ignition	350°F @ 7.7 mm
	f. Flame	6384°F
2.	Storage Time Limits	
	a. Motor	7 yr
	b. Igniter (Antares IIA)	86 mo
	c. Initiator	5 yr

C7.2.7.6 Altair III Rocket Motor - The Altair III rocket motor, manufactured by United Technology Center as the 5A0032, is a composite solid-propellant motor. It consists of a glass fiber epoxy pressure vessel (case), a nozzle, and an igniter.

The chamber is a filament-wound, glass-fiber-reinforced epoxy resin structure with ovaloid forward and aft domes. It incorporates integrally wound forward and aft adapters of high-strength aluminum. The chamber is 19.66 in. in diameter and 40.38-in. long. Nominal wall thickness of the chamber cylinder s 0.080 in. Vehicle attachment to the chamber is accomplished with aluminum skirt rings attached to glass fiber skirts at the cylinder-to-dome transition on both ends of the chamber. The forward and aft skirt rings contain 24 equally spaced tapped holes.

The propellant (UTP-3096A) is a composite PBAN-Ammonium Perchlorate-Aluminum type. It is cast in a transversely slotted circular port design. Propellant weight is 610 lb. TNT equivalent weight is 46 lb based on 7% of propellant weight.

The nozzle has a compression-molded exit cone covered with 0.01-in.-thick stainless steel for reinforcement and a graphite throat insert. It is attached to the aft motor adapter by an integral aluminum attach ring.

The igniter is a rocket type initiated by the two SD60A1 initiators with a 6.35 ± 0.85-sec pyrotechnic delay train. These initiators fire into a B-KNO_3 pellet charge; the charge then ignites the UTP-1095 propellant, which is similar to the propellant used in the motor. Igniter propellant weight is 0.4 lb.

The igniter electrical characteristics are:

1.	Bridge Wire Resistance	1.4 ± 0.4 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	1.0 A for min
	b. Minimum All Fire	3.5 A
	c. Recommend Fire	8.5 ± 3.5 A

Storage data for motor include:

1.	Temperatures	
	a. Motor Storage Range	30°F to 110°F
	b. Igniter Storage Range	30°F to 110°F
	c. Initiator Storage Range	30°F to 100°F
	d. Propellant Auto-Ignition	635°F @ 30s or 775°F @ 10s
	e. Igniter Auto-Ignition	657°F @ 10s
	f. Flame	5400°F
2.	Storage Time Limits	
	a. Motor	3 yr
	b. Igniter	1 yr
	c. Initiator	5 yr

C7.2.7.7 Alcyone (BE-3-A9) Rocket Motor - The Alcyone rocket motor was designed and developed by Hercules Incorporated, Bacchus Works, Magna, Utah. The BE-3-A9 Rocket Motor is a composite modified double-base solid propellant system capable of vacuum ignition and operation and will operate while being spun on its longitudinal axis at speeds up to 600 rpm. The motor has an overall length (with igniter) of 33.84 inches and a total weight of 216.2 pounds.

The chamber is a filament-wound fiberglass structure with a cylindrical length of 5.25 inches, skirt to skirt length of 10.24 inches, and outside diameter of 18.25 inches. The forward and aft closures are ovaloid shaped with integrally wound aluminum adapters for installation of the igniter and nozzle assemblies. Vehicle attachment is accomplished through aluminum skirt rings at the cylinder-to-closure transition at both ends of the chamber. Each skirt ring contains 24 equally spaced holes.

The propellant grain configuration is case bonded, six slotted. The propellant used is a modified double base designated DDP-50. Propellant weight is 191 lb. TNT equivalent weight is 191 lb based on 100% of propellant weight.

The nozzle is conical with an initial expansion ratio of 18.6 to 1 and a 15° half angle. A molded phenolic-silica fabric exit cone structure is overwrapped with epoxy fiberglass. The HLM-85 graphite throat is pinned with phenolic-silica fabric pins to retain the throat in position after burnout. The nozzle is attached to the motor by 16 steel 1/4-in.-diameter roll pins. The exit diameter is 12.20 inches. The length is 18.04 inches of which approximately 3.9 inches is submerged in the aft end of the motor.

The igniter is sealed into the forward closure with a packing and secured by igniter shaft nuts. The igniter is an assembly containing two Apollo Standard Initiators (ASI), an initiating charge, a pentaerythritol tetranitrate (PETN) mild detonation fuse, and a propellant basket containing boron potassium nitrate pellets. The ASI assemblies are located external to the motor and are removable for shipping and storage. Each initiator has two bridge wires and incorporates a PT-C6E-8-4P connector. The igniter does not have a built-in delay. From application of power to pressure rise is normally 0.0035 seconds. Igniter propellant weight is 0.81 lb.

The igniter electrical characteristics are:

1.	Bridge Wire Resistance	1.05 ± 0.10 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	1.0 A for 5 min
	b. Minimum All Fire	3.5 A
	c. Recommend Fire	5.0 A

Storage data for motor include:

1.	Temperatures	
	a. Motor Storage Range	30°F to 110°F
	b. Igniter Storage Range	40°F to 100°F
	c. Initiator Storage Range	25°F to 105°F
	d. Propellant Auto-Ignition	390°F
	e. Igniter Auto-Ignition	400°F
	f. Flame	6562°F
2.	Storage Time Limits	
	a. Motor	8 yr
	b. Igniter	3 yr
	c. Initiator	5 yr

C7.2.7.8 MARC-4 Spin Rocket Motor - The MARC-4 spin rocket motor, manufactured by Atlantic Research Corporation, is a miniaturized high-precision control rocket. Two types (1KS40 and 0.6KS40) are used on Scout.

The motor case and nozzle are an integral unit machined from lightweight, high-quality steel; case thickness in the cylindrical section is 0.04 in. A graphite insert prevents excessive erosion of the nozzle throat during burning. Overall length of the motor case and nozzle is approximately 5 in., and the diameter of the cylindrical section is approximately 1.5 in. A silicone rubber nozzle plug is press-fitted into the nozzle and retained at the throat diameter. Its function is to keep foreign material from entering the interior of the motor.

The propellant is high-performance Arcite 362M. Its form is an inside-outside burning cylinder bonded to the head of the motors. Weights for the 1US40 and 0.5US40 are 0.21 lb and 0.12 lb, respectively.

The igniter (U.S. Flare 3025C) is mounted in the head-end of the motor case. It consists of two U.S. Flare 90SA squibs that detonate a main charge of 1.4 g of $B-KNO_3$ pellets when initiated. Each 908A squib has an external case of aluminum, two lead wires, and a single 2.5-mil Ni-Chrome bridge-wire. The lead wires are shielded and sleeved with a mylar outer jacket that terminates in a PT06P-8-4P connector.

The igniter electrical characteristics are:

1.	Bridge Wire Resistance	0.78 ± 0.12 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	0.7 A for 15 s
	b. Minimum All Fire	1.2 A
	c. Recommend Fire	3.0 A

Storage data for motor include:

1.	Temperatures	
	a. Motor Storage Range	10°F to 120°F
	b. Propellant Auto-Ignition	350°F for 8 hrs
	c. Flame	4263°F
2.	Storage Time Limits	5 yr

The above data apply to both the 1US40 and the 0.6US40 except that the Bridge Wire Resistance is 0.79 ± 0.14 ohms for the 0.6US40.

C7.2.7.9 1KS75 Spin Rocket Motor - The 1KS75 spin rocket motor, manufactured by Atlantic Research Corporation, is a miniaturized high-precision control rocket. Its major components are a steel motor case with integral nozzle, nozzle plug; a slotted, inhibited solid propellant grain; and an igniter assembly.

The motor case and nozzle are an integral unit machined from H-11 or D6A steel. Case thickness in the cylindrical section ranges from 0.028 in. to 0.030 in. A graphite insert prevents excessive erosion of the nozzle throat during burning. Overall length of the motor case and nozzle is approximately 5 in., and the diameter of the cylindrical section is approximately 1.5 in. A silicone rubber nozzle plug is press-fitted into the nozzle and retained at the throat diameter. Its function is to keep foreign material from entering the interior of the motor.

The propellant (Arcite 428B) is an extruded composite type shaped in tubular form. It has a single longitudinal keyhole slot along the full length of the grain to provide required performance characteristics. The outside diameter and a portion of one end of the grain is inhibited with gum rubber stock. A rubber insulator embedded in the inhibitor at the slot area provides insulation, and a cellular rubber bumper bonded to one end of the grain keeps it under compression within the operational temperature range of the assembly. Weight is 0.31 lb.

The igniter (U.S. Flare 501B) is permanently installed as the forward closure of the motor. Two parallel-wired squibs (U.S. Flare 908B) are used to ignite approximately 0.5 g of B-KNO_3 pellets. The igniter cable is shielded four-wire 22-gage copper and is Teflon insulated and jacketed.

The igniter electrical characteristics are:

1.	Bridge Wire Resistance	0.83 ± 0.17 ohms
2.	Bridge Wire Current Requirements	
	a. Maximum No Fire	0.7 A for 15 s
	b. Minimum All Fire	1.2 A
	c. Recommend Fire	3.0 A

Storage data for motor include:

1.	Temperatures	
	a. Storage Range	10°F to 120°F
	b. Auto-Ignition	300°F @ 2 hrs
	c. Flame	5627°F
2.	Storage Time Limit	5 yr

C7.2.8 Scout Explosive Devices

The explosive devices used on Scout consist of shaped charges, safe-arm units, and various electro-explosive devices (EEDs). A brief description of these explosive devices and their functions is presented here.

The shaped charges, which are manufactured by the Vought System Division, a division of LTV Aerospace Corporation, contain 180 grains of RDX (cyclotrimethylene-trinitramine) per inch. They are formed by loading the RDX into standard 3/8-in. aluminum hydraulic tubing of the required length and then rolling the tubing to form a 60° angle. After rolling, which nearly doubles the density of the RDX, retainer caps are pressed into the ends of the shaped charges to obtain the desired end density and to seal the ends; then standard hydraulic nuts and sleeves are attached to the ends of the shaped charges as required.

The Thiokol Model 2125, electromechanical safe-arm unit features as a safe-arm mechanism, an explosive firing train with dual electric primers, and explosives capable of detonating two attached shaped charges. The safe-arm mechanism can be safed either electrically or manually, but it can only be armed electrically. The cross-firing train is provided by a booster assembly of RDX, which increases the reliability of the unit.

The safe-arm unit is installed in the vehicle with the indicator flag fully retracted into housing. It is armed by removing the safety pin and then stepping the rotor to the arm position by commanding an electrical arm signal from the blockhouse. After arming, the safe-arm unit can be fired by initiating the primers with an electrical signal. The ignition of the primers results in the ignition of the explosive detonators and cross-firing train; the detonators then ignite the booster charges.

Electro-Explosive Devices (EEDs) are used to perform various functions during flight. They are electrically initiated and include power cartridges, separation cartridges, ballistic cartridges, gas generators, and initiators. Electrical specifications of the EED primers are shown in Table C7-1.

Table C7-1 Specifications for Scout Electro-Explosive Device

DEVICE APPLICATION	MANUFACTURER & PART NUMBER	BRIDGE RESISTANCE	INSULATION PINS TO CASE	CURRENT - FIRING DATA			RATED TEMP LIMITS	STORAGE TEMP LIMITS	SHELF LIFE
				MAX NO FIRE	MIN ALL FIRE	RECOM ALL FIRE			
SAFE ARM FOR DESTRUCT	23-003933	.19 ±.03 OHMS	100 MEGS MIN @ 500 VDC	1.0 A for 5 min.	2.3 A	5.0 A	+176°F 0°F	+120°F 0°F	5 YR
HEAT SHIELD SEPARATION CARTRIDGE	3398670055-1	1.05 ±.10 OHMS	2 MEGS @ 250 VDC	1.0 AMP	3.5 AMP	5.0 A		+200°F 0°F	5 YR
4TH & 5TH STG SEPARATION CARTRIDGE	23-002981	1.05 ±.10 OHMS	2 MEGS @ 250VDC	1.0 AMP for 1 min.	3.5 AMP	5.0 A		+160°F 0°F	5 YR
POWER CART-RIDGE FOR PAYLOAD SEPARATION	HISHEAR PC-33 POWER CART	1.1 ±.1 OHMS	2 MEGS @ 500 VDC	1.0 A for 5 MIN.	3.5 A	5.0 A	+175°F 0°F	+120°F - 20°F	5 YR
DESPIN CABIE CUTTER, EG & G PAYLOAD SEPARATION	SPACE ORDNANCE S01-266-4	1.05 ±.10 OHMS	2 MEGS @ 500 VDC	1.0 AMP FOR 5 MIN.	3.5 AMP	5.0 A		+1.05°F +25°F	5 YR
IGN-DEST BATTERY INITIATOR	23-002583-2	.65 ±.10 OHMS	500 MEGS	0.01 AMP	2.0 AMP	5.0 A	+170°F + 32°F	+100°F + 35°F	4 YR
5TH STAGE IGN BATTERY	23-003665-1	.05 TO .18 OHMS	10 MEGS MIN @ 500 VDC	2.0 A	9.0 A		+176°F 70°F 0	+160°F 0°F	2 YR
POWER CART-RIDGE FOR PAYLOAD SEPARATION	HISHEAR PC 19-009 POWER CART	1.1 ±.1 OHMS	2 MEGS FOR VDC	1.0 AMP FOR 5 MIN.	3.5 AMP	5.0 A		+120°F - 20°F	5 YR

C7.2.9 Guidance and Control System

The Scout Guidance Control System provides an attitude reference and stabilizes the vehicle around its three major axes. It consists of:

1. Guidance unit whose three single-axis miniature integrating gyros are secured directly ("strapped down") to the vehicle airframe.

2. Rate gyro unit to provide rate signals.

3. Power Control Relay Box to provide power and ignition switching.

4. Intervalometer to provide scheduling of events during flight.

5. Programmer to provide pitch or yaw gyro torquing voltages.

6. 400-cps inverter and battery power supplies.

7. Poppet valve electronics to switch the valves that control the "on-off" reaction jet stabilization system.

8. Hydraulic servocontrol amplifier to amplify the stabilization system error signals for first stage control.

9. Yaw-roll compensation unit to compensate for errors in roll due to yaw displacements and in yaw due to roll displacements that occur during pitch program.

The guidance unit provides three gain conditions to maintain stability and attitude control from lift-off to fourth-stage separation: Gain I during first-stage control; Gain II during second- and third-stage burn control, and Gain III during third-stage coast control.

C7.3 MISSION SCENARIO

The Mission Scenario was covered only briefly in the reference document (reference the General Vehicle Description, Paragraph C7.1.)

Activation of the ignition batteries is accomplished during countdown by firing the gas generators with a command from the blockhouse. Gas pressure then forces electrolyte from the reservoir into the cells. A relief port is provided in each battery case to vent gases when the differential pressure exceeds 5 psi.

Ignition times vary depending upon mission requirements, but the following sequence will always be valid:

T_0	-	First-Stage Ignition from Ground Command
T_1	-	Second-Stage Ignition
*T_2	-	Heatshield Ejection
**T_2	-	Third-Stage Ignition Command and Heatshield Ejection
*T_3	-	Third-Stage Ignition
**T_3	-	Third-Stage Ignition (through a 1.7 $\pm$ 0.2 second squib delay)
***T_4	-	Establish Fourth-Stage Attitude Correction System Reference
T_5	-	Fourth-Stage Spin Motor Ignition and Fourth-Stage Ignition Command
T_6	-	Fourth-Stage Separation
T_7	-	Fourth-Stage Ignition (through 6.35 $\pm$ 0.85 second squib delay), *** and start of Fifth-Stage Timer
T_8	-	Fourth-Stage Burnout
***T_9	-	Payload Separation Activate Attitude Correction System
***T_{10}	-	Deactivate Attitude Correction System
***T_{11}	-	Fifth-Stage Ignition
***T_{12}	-	Payload Separation

* Applicable only when Antares IIB motor is used for third stage.
** Applicable only when Antares IIA motor is used for third stage.
*** Applicable only when Fifth Stage and ACS is used.

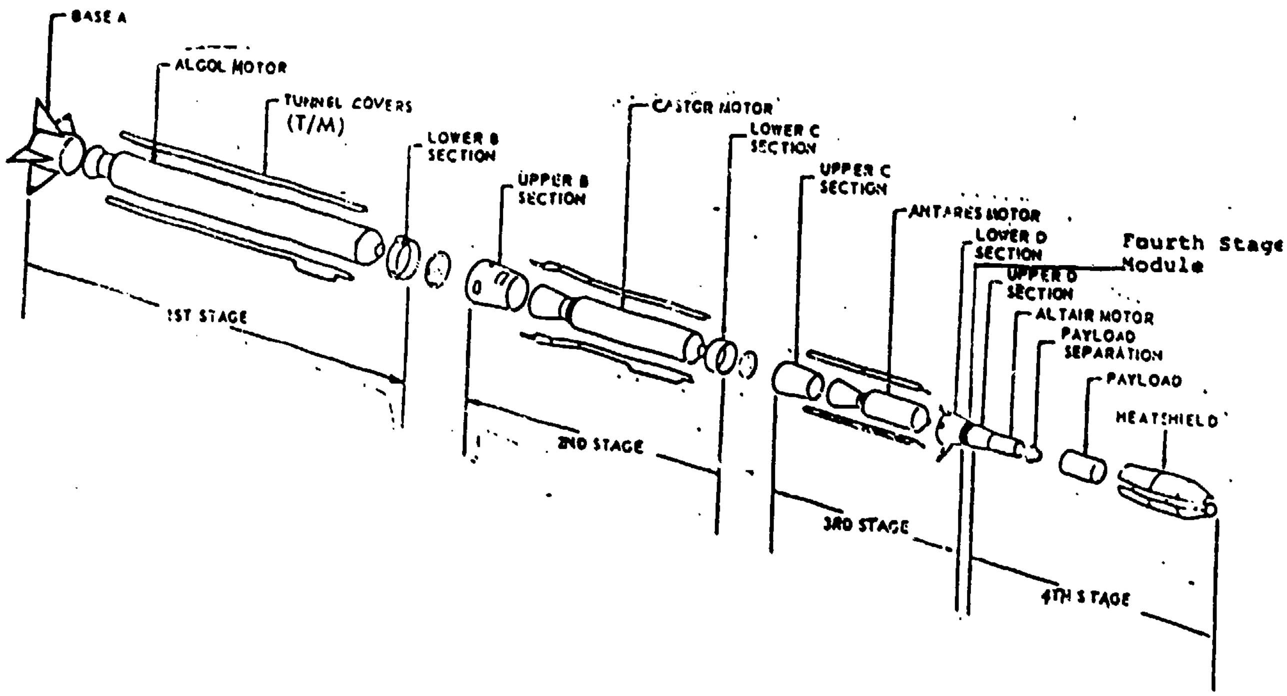

Figure C7-1 Scout Vehicle Exploded View

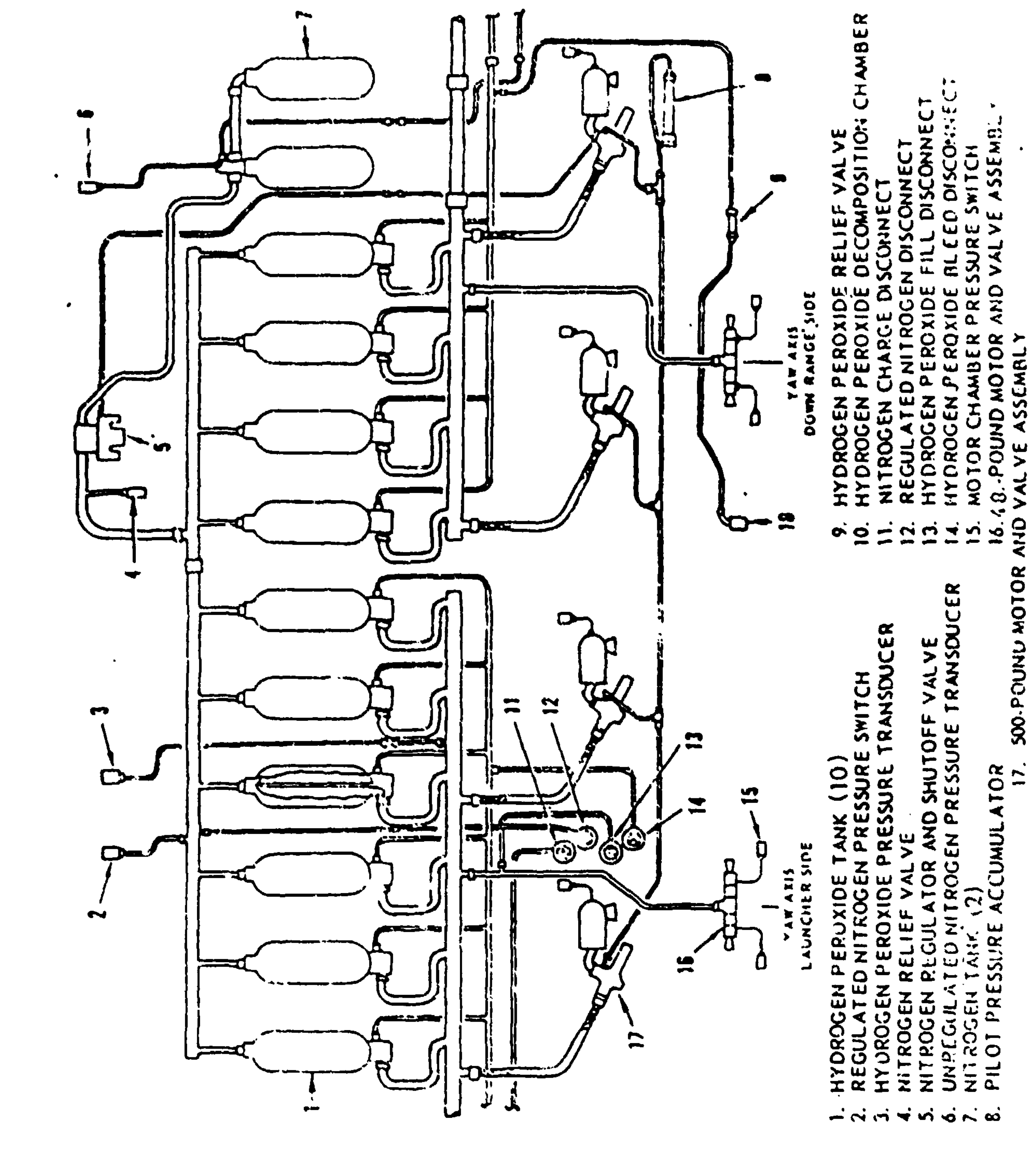

Figure C7-2 Reaction Control System, Second Stage

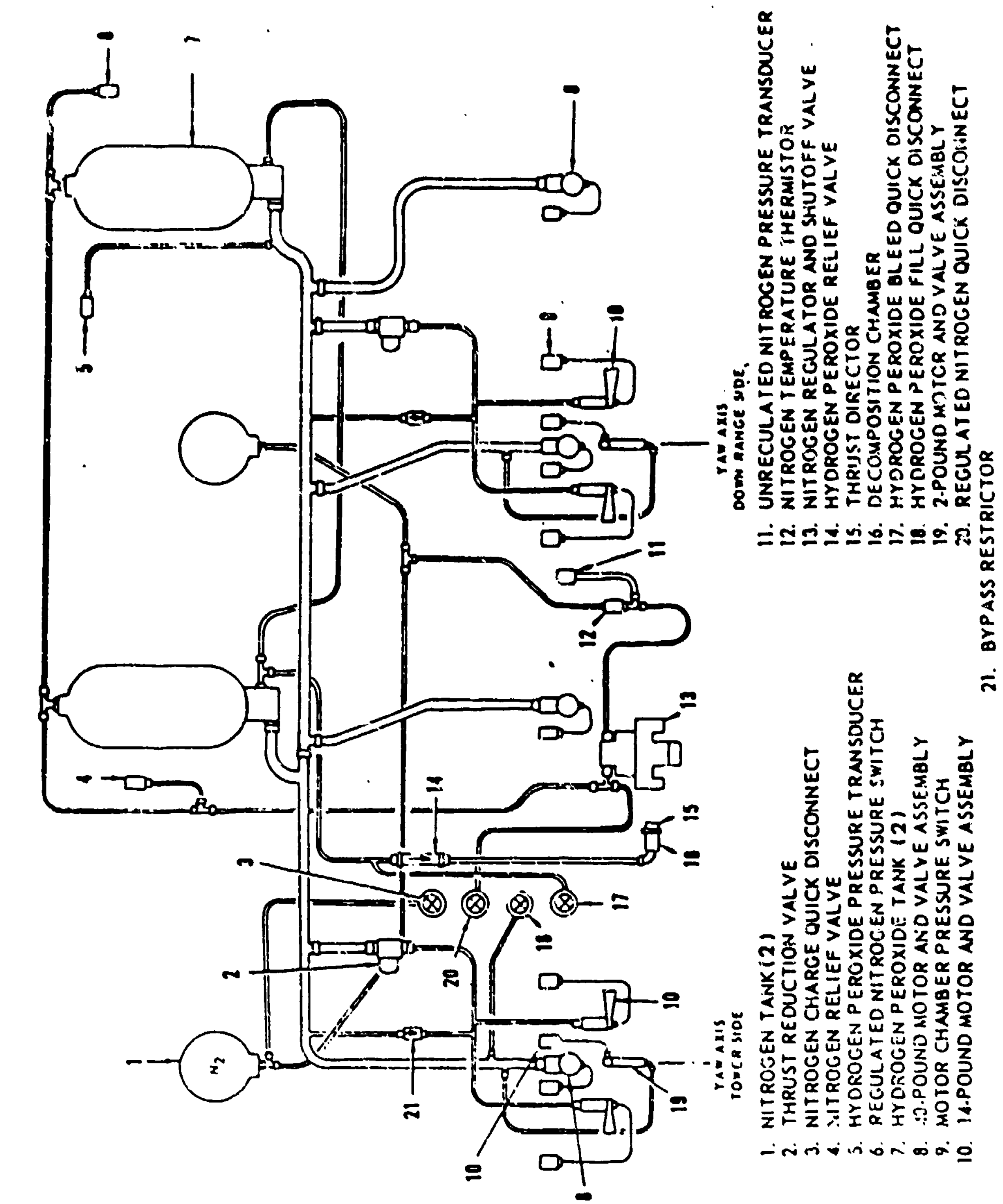

Figure C7-3 Reaction Control System, Third Stage

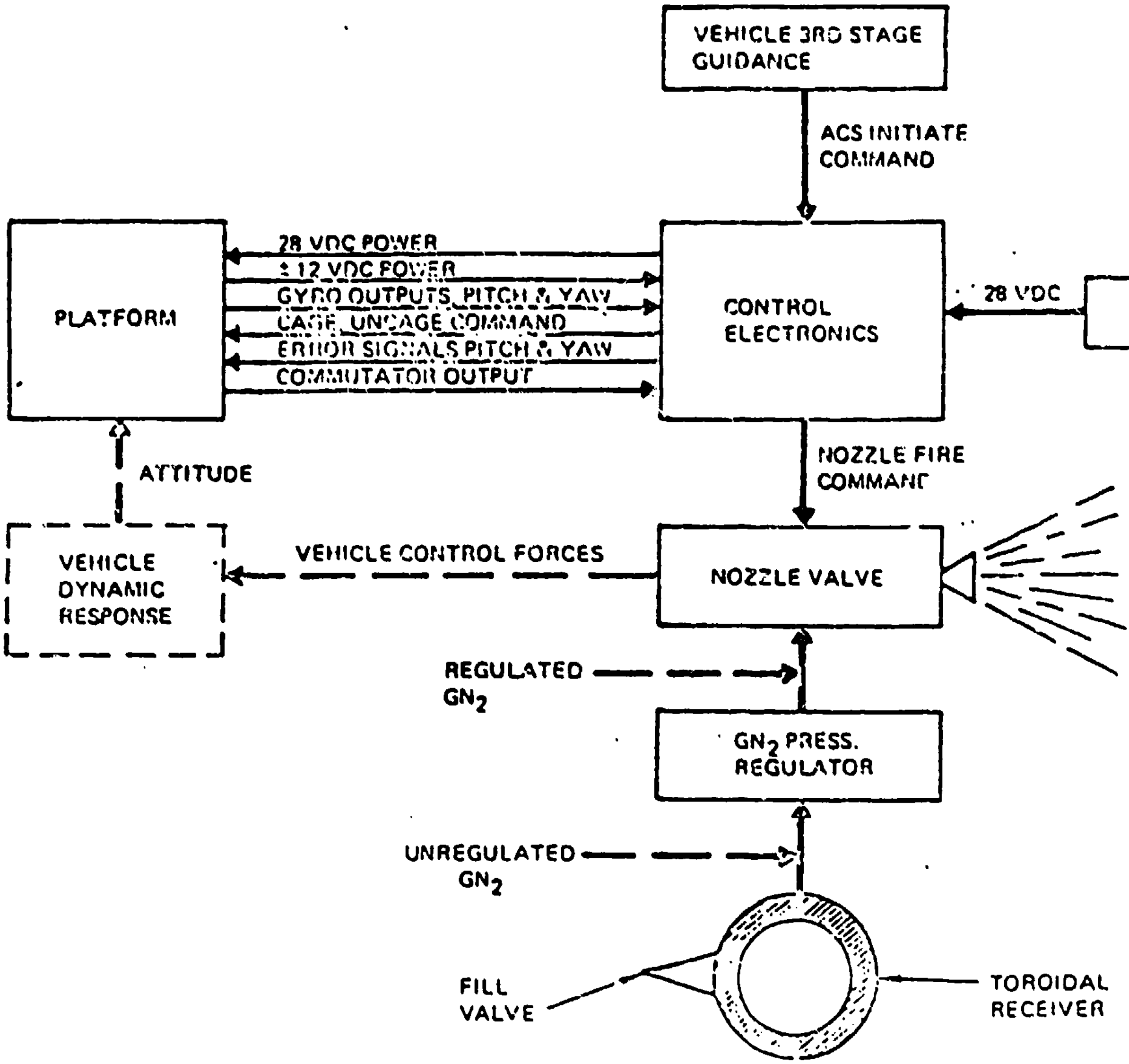

Figure C7-4 ACS Functional Block Diagram

Appendix C8
Titan IV

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TITAN IV

C8.0 INTRODUCTION

The following data were extracted from the Standardized Launch Vehicle X documents, "Executive Summary, Volume I," P84-45031-1, and "Technical Proposal, Volume III," P84-45031-3, August 1984, Martin Marietta Aerospace, Denver Aerospace, P.O. Box 179, Denver, CO 80201.

C8.1 GENERAL DESCRIPTION

A three-body launch vehicle was selected to meet TIV requirements as shown in Figure C8-1. The vehicle consists of a two-stage liquid propellant core vehicle, two strap-on solid rocket motors (SRM), a high-energy Centaur upper stage, and a payload fairing (PLF). The overall vehicle length is 204 ft, and lift-off weight is 1,910,499 lb.

The TIV vehicle performance is 10,565 lb to GSO (requirement is 10,000 lb). This is accomplished by using the boost vehicle (SRMs and core vehicle) to position the Centaur and payload on a trajectory from which the Centaur provides the final velocity to achieve an 80 x 95-nmi park orbit and then delivers the payload to the GSO. The PLF protects the payload and Centaur during the atmospheric portion of the boost flight and ensures that TIV payload environments are comparable to the Space Transportation System (STS). In addition, the Centaur upper stage physical and functional interfaces with the satellite vehicle (SV) are equivalent to the STS.

The TIV airborne vehicle configuration (Fig. C8-2) is a standard three-body vehicle consisting of a core vehicle, dual strap-on solid rocket motors (SRM) and a high-energy Centaur upper stage. The TIV vehicle includes a 200-in.-diameter by 86-ft-long payload fairing that encloses the Centaur stage and provides a 15X40 ft dynamic payload envelope.

C8.1.1 Core Vehicle

The core vehicle consists of two 10-ft-diameter liquid propellant stages. Stage I is 86.5 ft long and contains 341,000 lb of propellant (nitrogen tetroxide and Aerozine 50). Stage II is 32.6 ft long and contains 77,000 lb of the same propellants. The Stage I dual-subassembly engines provide 546,000 lb of thrust and are gimbaled to provide pitch, yaw, and roll steering control. The Stage II single subassembly gimbaled engine provides 104,000 lb of thrust and, in conjunction with a roll control nozzle, provides the stage-steering capability. Both stages use an autogenous pressurization system that contains no operating components. A four-axis gimbaled inertial measurement unit and digital computer provide the boost vehicle guidance and control functions. Other core vehicle avionics include a remote multiplexed instrumentation system, a 10-W S-band transmitter and antenna, a dc battery-powered sequence system, and a range safety destruct and inadvertent separation destruct system (ISDS) in each stage.

C8.1.2 Booster Modifications

The low-risk booster TIV design is an extension of the operationally proven Titan 34D vehicle and the STS Centaur design. The TIV boost vehicle is a natural growth configuration of the Titan family that started with the Titan II Weapons System and grew through the Titan IIIC (added SRMs), Titan 24B (stretched core vehicle), Titan IIIE (Centaur upper stage) and 14.5-ft-diameter payload fairing (PLF) and T34D (stretched SRMs). The TIV Centaur upper stage is a modified version of the STS Centaur G Prime configuration with Centaur G avionics. Figure C8-3 shows an outboard profile of the boost vehicle along with an identification of major changes from established vehicles.

The strap-on SRMs are 10 ft in diameter by 112.9 ft long and are loaded with UTP-3001B solid propellant. The total thrust of the two SRMs is 2.95 x 10^6 lb. The SRM web action time is 110.5 seconds, and each SRM provides a total impulse of 159.7 x 10^6 lb-s. The SRM thrust vector control, which is driven by the core guidance system, injects nitrogen tetroxide into the motor exhaust stream, thereby producing angular displacement of the exhaust streams to produce vehicle steering. The SRMs carry remote multiplex units (instrumentation pickup units) and command and inadvertent separation destruct hardware.

The Centaur is a single-stage liquid-oxygen and liquid-hydrogen vehicle that is 170 in. in diameter (max) and 29.3 ft long. The stage consists of oxygen and hydrogen pressure-stabilized propellant tanks that contain 46,136 lb of propellant. The stage contains a dual-subassembly engine with multiple-start capability that delivers a 33,000 lb thrust. A hydrazine reaction control system provides for coast-phase steering control. A gimbaled inertial platform and digital computer provide guidance and control functions. Command control receivers enable range engine shutdown and destruct, and a C-band tracking transponder is provided. The stage has an instrumentation system that is compatible with payload user needs, including an encryptor and a multiple antenna S-band telemetry system. This stage also provides the payload physical and functional interface.

The final TIV element is the payload fairing (PLF) that protects the upper stage and payload from aerodynamics during the boost phase. The PLF is an all-metal isogrid construction that is coated externally and internally to reduce the effects of aeroheating and flight acoustics and vibration. The PLF environment and spatial conditions are compatible with the STS.

The TIV design incorporates payload accommodations equivalent to the STS (Fig. C8-4). These accommodations assure that DOD payloads designed for STS/Centaur missions will require, at most, minimal design modifications to fly on the TIV and are key to assuring that the TIV complements the STS. The TIV payload capability of 10,565 lb for the GSO mission is equivalent to the predicted Centaur G STS performance of 9,413 lb. The TIV/Centaur payload interfaces are the same as the STS/Centaur interfaces because identical interface hardware is used.

A modified ITL concept for the flow of TIV segments from receipt at CCAFS through assembly and launch from LC-41 is shown in Figure C8-5.

C8.1.3 Alternative Upper Stages

A trade study will determine criteria and criteria rank and will evaluate potential alternative upper stages.

Candidate upper stages are shown in Figure C8-6 which considers upper stages that could perform the TIV ILC and TIV reduced mission payload. The selection criteria and alternative upper stage candidates will be reviewed with the Air Force in April 1985.

After determining which alternative upper stages meet the mandatory criteria, the relative performance of each stage will be evaluated against the other criteria. A significant portion of the effort will required to refine structural changes (Figure C8-7) and to update the estimated payload load capabilities shown in Table C8-1.

Table C8-1 Payload Capability for Other Upper Stages

Mission	Centaur G Prime	Alternative Upper Stage Centaur G	TOS/AMS	4-Tank Transtage	Transtage AMS	IUS
Geostationary	10,56[illegible]	8,050*	5,773*	**	5,970	4,950
12h, 63.4-deg Inclination	17,819	15,190*	11,021*	**	**	**
24h, 65-deg Inclination	10,918	2,548*	6,103*	**	**	**

* Capability will be refined by IR&D study.
** Capability will be defined by IR&D study.

C8.1.4 Loads

TIV equivalent axial loads range from no change up to 64% greater than for Titan 34D. TIV maximum airloads analysis results indicate that the Stage I and II structure must be strengthened to withstand the increased loads caused by the larger PLF and stretched core. A static elastic loads analysis was performed for a Q_T of 1000 psf-deg using a rigid Centaur upper stage and a 10,000-lb payload.

Table C8-2 compares the equivalent axial loads (P_{EQ}) for the TIV with those for the Titan 34D. The increases range from no change to a 64% increase, with the larger increase occurring in the Stage II structure.

PLF loads were estimated more conservatively to protect PLF design deflection requirements and were based on a Q_T of 1400 psf-deg. Table C8-3 presents the PLF distributed loads.

Table C8-2 Equivalent Axial Loads (P_{EQ}) for TIV vs. Titan 34D

Station		P_{EQ} TIV 10^6 lb	P_{EQ} T34D 10^6 lb	% Change [illegible] TIV [illegible] P_{EQ} T34D
108		0.81	1.08	-25
165F	Fwd	0.81	1.08	-25
165A	Oxidizer	1.71	1.08	+58
203	Skirt	1.77	1.08	+64
283F		1.90	1.17	+62
283A	Stage II	1.90	1.17	+62
312F	Oxidizer	1.94	[illegible]	+62
312A		2.07		+59
409		2.13	[illegible]	+56
430F	Stage II	2.12	1.38	+54
430A	Fuel	2.20	1.44	+53
500F		2.17	1.50	+45
500A		2.18	1.50	+45
650		2.04	1.37	+49
812	Stage I	1.94	1.25	+55
975F	Oxidizer	1.70	1.13	+50
975A		2.29	1.66	+38
1070F		2.09	1.61	+30
1070A	Stage I	2.09	1.58	+32
1211	Fuel	1.77	1.86	No Change
1351F		1.46	1.94	No Change
1351A		1.77	2.24	No Change

Table C8-3 PLF Distributed Loads

PLF Station	Limit Shear - V_R (kips)	Limit V_{EQ}* (kips)	Limit P_{Axial} (kips)	Limit Moment - M_R (10^6 in.-lb)	Limit P_{EQ}t (10^6 lb)
1032	2.88	3.17	18.8	0.0	0.0188
950	10.75	11.83	45.0	0.236	0.0553
869	21.95	24.15	60.4	1.107	0.0931
787	31.20	34.43	70.2	2.907	0.1355
705	38.38	42.22	72.2	5.465	0.1913
624	43.28	47.61	74.3	8.574	0.2588
542	45.92	50.51	76.3	12.120	0.3354
460	47.06	51.77	78.5	15.890	0.4169
379	47.42	52.16	81.0	19.700	0.4996
297	47.48	52.23	83.6	23.590	0.5840
215	47.56	52.32	87.4	27.480	0.6697
52	47.18	51.90	107.0	35.220	0.8530
0	46.89	51.58	121.8	37.690	1.4480

*$V_{EQ_{Ult}} = F_{Ult}\left(V_R + \frac{|M_X|}{Dia}\right) = F_{Ult}(1.1\ V_R) = F_{Ult}(V_{EQ_{Lim}})$

t$P_{EQ_{Ult}} = F_{Ult}\ \ 1.06\ P_{Axial} + \frac{2(1.05\ M_R)}{Rad} = F_{Ult}(P_{EQ_{Lim}})$

Figure C8-8 shows the TIV critical flight events that will be analyzed.

C8.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C8.2.1 Solid Rocket Motors (SRM)

The UT-CSD 7-segment SRM was selected to take advantage of the existing qualified design and development testing done for the MOL program.

The TIV Stage 0 consists of two 7-segment SRMs developed and qualified by CSD (Fig. C8-9). Four static test firings, including thrust vector control testing, were completed for the Manned Orbital Laboratory (MOL) program, thus providing a low-cost and low-risk design for the TIV SRM.

Stage 0 consists of two identical segmented 120-in.-diameter SRMs mounted 180° apart on the core vehicle. Each SRM has a sea-level thrust of 1.475×10^6 lb, an I_{sp} of 269.8 s, a web action time of 110.5 s, and a total impulse of 159.7×10^6 lb-s. The propellant (UTP-3001B) is PBAN-based and contains 84% solids (16% aluminum and 68% ammonium perchlorate, plus additives and an iron oxide catalyst). The nozzle assemblies use a 9.18 to 1 expansion ratio and are canted 6 deg. Thrust vector control is provided by a nitrogen-pressurized ullage blowdown liquid injection system. The N_2O_4 injectant is pressure-fed to 24 electromechanical valves mounted on each nozzle. Forward and aft staging rockets provide the separation forces. Each motor segment is 130 in. long and constructed of D6aC steel. The same material is used for the forward and aft closures. Each 7-segment SRM has a dry weight of 94,259 lb and a loaded weight of 695,386 lb.

Increased Stage 0 performance is achieved by replacing 5-1/2-segment SRMs with 7-segment SRMs and by designing a lighter aft support skirt assembly. The transition from 5-1/2 to 7 segments requires the following: Using UTP-3001B propellant rather than UTP-3001 and a 40-in. longer forward closure; replacing the 1/2 segment with two full segments; and using a larger throat and nozzle assembly. The result is the basic configuration that was designed, built, and qualified for the MOL program, (Fig. C8-10).

A modified T34D design for the aft support skirt is shown in Figure C8-11. This design results in a lighter skirt assembly than that proposed for MOL. No TVC modifications are required because the current system was designed for a 7-segment configuration.

Transition to the 7-segment configuration results in additional propellant and dry weight increases associated with the added segments.

The 30% total impulse increase is a result of the 127,500-lb additional propellant per SRM. Burnout weight increases by 13,800 lb (17%), which includes the 2600-lb weight decrease per SRM due to the lighter skirt. Table C8-4 summarizes major SRM characteristics.

Table C8-4 Major SRM Characteristics

The TIV 7-Segment Performance is Comparable to the MOL 7-Segment Design.

Parameter (60°F Nominal)	T-34D 51/2 Segment	MOL *** 7-Segment	TIV 7-Segment
Maximum (Initial) Sea Level Thrust *, lb x 10^6	1.273	1.442	1.475
Delivered I_{sp}*, s (VAC)	265.2	269.8	269.8
Web Action Time, s	104.1	113.4	110.5
Action Time, s	113.7	124.5	121.2
Duration (Total) Impulse* lb-s x 10^6 (VAC)	123.1	158.6	159.7
Max ** Expected Operation Pressure (MEOP), s	935.0	866.0	888.0
Propellant Wt, lb x 10^3	464.3	589.1	592.0
Inert Wt Expended, lb	5000	7000	7000

* Nozzle Centerline
** Value Represents 3-sigma Maximum Pressure at 90°F
*** MOLSRM 1207-3

As shown in Table C8-5, most structural components will require strengthening to meet the higher loads. In all cases, Titan-34D-approved materials will be used for construction.

Table C8-5 Core Vehicle Structural Modification

Design Modification	Description	Design Impact	Step Performance Impact
Stage 1 Fuel Tank	Barrel Section Stretched 43.8 in. & Increase in Strength for New Loads	Skin Length & Thickness	+350 lb
		K Frame Resized Up* Y Frames Resized Up* Forward Skirt Components Resized Up* External Conduit Lengthened* Internal Conduit Lengthened* Oxidizer Feeline Lengthened	+227 lb
[illegible]age 1 Oxidizer Tank Stretch	Barrel Section Stretched 51.2 in & Increased in Strength for New Loads	Skin Thickness, Length, Stringer Webs, & Flanges Resized Up	+409 lb
		Y Frames Resized Up * Aft Skirt Components Resized Up* Forward Skirt Components Resized Up* External Conduit Lengthened*	+532 lb
Re[illegible]le Ski[illegible] Resi[illegible]	Skirt Strength is Increased for New Loads	Skin, Frames, & Stringers Resized Up in Thickness	+ 32 lb
		SRM Attachment Point Structure Deleted at Sta 500 & Moved to Sta 370.6; Basic Internal Structure Remains Same; Local Support Structure Deleted*	-141 lb
Stage I[illegible] Fuel Tank Stretch	Barrel Section Stretched 9 in & Increased in Strength for New Loads	Skin Thickness Resized UP	+ 74 lb

Table C8-5 Core Vehicle Structural Modification (Continued)

Design Modification	Description	Design Impact	Step Performance Impact
		Y Frames Resized Up* Aft Skirt Components Resized Up* Internal Conduit Lengthened* External Conduit Lengthened* Oxidizer Feeline Lengthened* Feeline Dome & Cone Inlet & Outlet Angles Changed* SRM Electrical Cable Umbilical Structure Deleted from Aft Fuel Skirt*	+318 lb
Stage II Oxidizer Tank Stretched	Barrel Section Stretched 9 in & Increased in for New Loads	Skin Thickness Resized Up	+ 83 lb
		Y Frames Resized Up Aft Skirt Components Lengthened* External Conduit Lengthened* SRM Electrical Cable Umbilical Structure Added to Aft Skirt*	+478 lb
		Fourth Retrorocket Package Added to Aft Skirt	+ 21 lb
Centaur Upper Stage Requirements	Increases Length Forward Oxidizer Skirt; Addition of Thermal Barrier to Skirt; Change in Skirt Vending Requirements	Forward Oxidizer Skirt Skin Strengthened 43.1 in. & Resized Up in Thickness	+210 lb
		PLF Separation, Electrical Umbilical Connections & Upper Stage Connection Revised* Provisions for Attaching Thermal Barrier Added to Skirt, Similar to One Used on Titan D-1T	Neglible Weight Change

Table C8-5 Core Vehicle Structural Modification (Continued)

Design Modification	Description	Design Impact	Step Performance Impact
		Existing Triangular Vent Holes Deleted; New Circular Vent Holes Added in New Location Field Splice Added at Two Places Stringers & Frames Resized Up* Payload Fairing Attach Points Added (36 Places)*	+722 lb
7-Segment SRMs	Relocation of SRM Attach Points on Vehicle to Sta 370.6	SRM Attach Fittings, Frame & Local Support Structure Added to Stage II Forward Fuel Skirt	+174 lb
Avionics Revisions	Add two Trusses in Compartment 2A & Delete Truss from Compartment B	Additional Truss Tie Points, Hardware, & Access Doors Added to Forward Fuel Skirt Truss, Tie Points, Mounting Hardware & Access Doors Deleted from Stage II Forward Fuel Skirt	Negligible Weight
		Minitrusses Added to Forward Skirt to Support Destruct Initiator & Destruct Charges*	+10 lb

* Design impacts are secondary effects caused by primary modifications.

C8.2.2 Core Vehicle Structures

The 112-in. tank stretch provides for an additional 46,000 lb of propellant in Stage I and 8300 lb in Stage II. Stage I dry weight increases by 2125 lb and Stage II by 2704 lb to accommodate tank stretch and structure strengthening. These weights also include the relocation of the SRM attach points from Stage I to Stage II, the PLF and Centaur attach points at the Stage II forward oxidizer skirt, and the thermal barrier between Stage II and the Centaur.

The TIV core vehicle is a 10-ft-diameter-by-119.1-ft-long, two-stage liquid propellant vehicle. The primary structure consists of aluminum alloy barrel skins with stringers and ring frames. To minimize cost and risk, the T34D structure was used as the basic design for TIV, and the T34D configuration was extended by 112 in. to accommodate additional propellant requirements.

The core vehicle is the basic TIV building block because the two 7-segment SRMs strap onto the core, and the Centaur upper stage and PLF attach to the Stage II forward oxidizer skirt. In addition, the core vehicle structure provides the tankage for the liquid rocket engines and provides trusses and mounting for avionics equipment.

The Stage I and II rocket engines are the uprated versions of the ATC-LR-87-11 (Stage I) and ATC-LR-91-11 (Stage II) to be used on Titan 34D-16. For TIV, the Stage II engine thrust will be increased 5% for added system performance.

Both stages will retain the Titan III hypergolic propellants and simplified autogenous pressurization system.

C8.2.3 Core Vehicle Propulsion

The TIV core propulsion system uses the latest uprated T34D engines coupled with longer burn durations because of the extended propellant tanks.

Stage I and II propulsion is provided by the LR87-AJ-11 and LR91-AJ-11 engines developed by Aerojet Tech Systems Company (ATC). By using the latest Titan 34D version of these engines coupled with the stretched core propellant tanks, a low-risk, mature TIV core vehicle propulsion system is provided (Fig. C8-13).

The ATC engines meet the requirements of the core propulsion system (Fig. C8-14A) and have exceeded the performance and extended burn durations needed for TIV operation on previous test firings.

The core propulsion is provided by two tandem-mounted liquid stages using storable hypergolic propellants. The fuel is Aerozine 50 (50/50 blend of hydrazine and UDMH) and the oxidizer is N_2O_4. Stage I uses twin turbopump-fed engine subassemblies with 15 to 1 nozzle expansion ratios. Stage I nominal performance is 546,000 lb of thrust, and I_{sp} of 302.6 s, and a burn time of 190 s (341,000 lb of propellants). Gas generators on each subassembly drive the engine turbopumps. A solid propellant start cartridge on each subassembly initiates pump operation. Thrust vector control (pitch, yaw, and roll) is provided by gimbaling the subassemblies. Stage II uses a single gimbaled turbopump-fed engine subassembly with a 49.2 to 1 expansion ratio nozzle. Stage II nominal performance is 104,000 lb, has an I_{sp} of 316s and burn time of 232s (77,000 lb of propellant). The Stage II engine will be balanced to a thrust level 5% higher than the current Titan 34D engines. Roll control is provided by ducting turbine exhaust through a swiveled roll control nozzle. Tank pressurization for both stages is provided by an autogenous (self-generating) system and requires no transducers, valves, or active control systems. Both stages shut down upon propellant depletion. Except for the changes associated with the longer propellant tanks (longer oxidizer feedlines, longer pressurization lines), this is the same flight-proven design used on the Titan III/34D programs.

Additional performance is provided to the TIV over the T34D configuration by stretching Stage I and II propellant tanks 95 and 17 in., respectively. Stage II thrust is uprated 5% by re-orificing the gas generator circuit.

The core propulsion system design is shown in Figure C8-14 with the modifications summarized in Figure C8-14A. Tank stretch is accomplished by additional barrel length. The Stage II 5% thrust increase is accomplished by increasing pump speed. This technique has been successfully demonstrated on the Stage I engines on several previous uprates. This technique has also been successfully demonstrated during previous engine testing when the engine was operated at a thrust level of 105,314 lb (+7%) for 250 seconds. Therefore, the 5% increase is well within the capability of the engine. In addition, the Stage I engine burn time increases 25 seconds to a total of 190 seconds (nominal), and Stage II increases 13 seconds to a total 232 seconds (nominal). Both increases are well within current demonstrated engine capabilities.

Increased tank length results in longer engine burn times and requires corresponding length changes in the oxidizer feedlines and autogenous pressurization lines. Analyses have shown that even with the increased liquid height caused by the stretched tanks, feed system pressures will be at or slightly below current pressure because of reduced acceleration levels. The only burn-time-limiting components on the engines are the ablative skirts. To restore operating margin, the Stage II ablative skirt liner will be machined to add approximately 0.020 in. to the nominal thickness (from 0.400 to 0.420 in.). All other Stage II components and all Stage I components are qualified for durations in excess of TIV requirements. The fuel tank aft thrust cone structure will be modified for the higher Stage II engine thrust, and some minor increases in engine propellant line flange thickness and a gimbal bolt material change may be incorporated to restore operating margins.

An additional 46,000 lb of propellant is added to Stage I resulting in +25 seconds of burn time. The Stage II propellant increase of 8300 lb, coupled with the increased flow rate from the 5% high, balance results in a net increase of 13 seconds burn time.

Because of the additional line lengths, a pressure drop analysis will be accomplished and the results incorporated into the system performance analysis. A stress analysis of the Stage II combustion chamber will be performed to verify margins because of the added mass on the engine skirt.

The Stage I nominal burn time increase from 165 seconds (T34D) to 190 seconds (TIV) is within the 200-second engine qualification levels and well below the demonstrated level of 300 seconds. The Stage II nominal burn time increase from 219 seconds (T34D depletion shutdown) to 232 seconds (TIV) is slightly longer than the 225-second engine qualification, but well below the demonstrated level of 250 seconds. As mentioned earlier, Stage II operating margins will be restored by increasing the thickness of the ablative skirt liner.

C8.2.4 Core Vehicle Flight Controls

The flight control system (FCS) uses the flight-proven T34D/TS configuration to enhance launch availability while maintaining vehicle stability.

The FCS of the TIV is an adaptation of the flight-proven T34D/TS digital configuration (Fig. C8-15A). The modifications are cost effective and feasible as shown by frequency domain analyses and time domain simulations. A pitch/yaw steering scheme is implemented in the software to maximize launch availability by using launch-day measured winds. Launch availability is enhanced by using a load relief autopilot that complements the wind biasing technique.

This configuration balances the increased aerodynamic loading caused by the bulbous PLF with an increased control capability primarily due to the revised thrust and mass distribution of the 7-segment SRMs (Fig. C8-15B). The potential stability degradation caused by distributed aerodynamics that occur for large flexible boosters is understood, and unique analysis techniques and tools have been developed to analyze this phenomena. For the TIV configuration, analysis has shown this degradation to be acceptable with reductions of 0.46 and 0.98 dB, respectively, in the aerodynamic gain margins of pitch and yaw at the max-Q time point.

Significant parameters affecting vehicle stability show strong similarity between the TIV and T34D/TS (Fig. C8-15D). The TIV open-loop frequency response is shown in Fig. C8-15E for the pitch channel at the max-Q time point. The T34D/TS open-loop frequency response, shown in the same figure, is very similar to that for the TIV. Thus, FCS design tools and techniques developed for Titan are directly applicable to the TIV.

Stability analyses were conducted for all three channels (pitch, yaw, and roll) for the TIV Stage 0 flight phase. Results of these analyses, which included the wind tunnel test aerodynamic parameters, show that the FCS configuration meets or exceeds the stability margin requirements and objectives. For example, the stability analysis results for the pitch plane show that all stability margin objectives are exceeded.

FCS analysis tools are sophisticated and have been verified by continual use on various Titan programs. These tools (Fig. C8-15F) include high-fidelity math models necessary for FCS design for a large, flexible booster.

To alleviate concerns over the validity of the aerodynamic data, a wind tunnel test was completed using a 2.8% scale model of the TIV. These results were used to verify stability margin requirement compliance with SS-ELV-401 and to develop Stage 0 FCS parameters for use in 6-DOF simulations.

C8.2.5 Core Vehicle Avionics

The core vehicle avionics system requires no new design and only minor modifications to existing components.

The core vehicle avionics system must meet the requirements of SS-ELV-401 and derived requirements that allow TIV to perform the specified missions. It must provide guidance, navigation, control, and electrical sequencing during the boost vehicle (BV) portion of flight. It must also provide adequate telemetry information and meet the range safety requirements of AFETRM 127-1.

The TIV core avionics (Fig. C8-16) uses the Delco Systems IGS, consisting of a missile guidance computer (MGC) and a four-axis gimbaled inertial measurement unit (IMU) to provide guidance, navigation, control, and sequencing of the BV independent of the Centaur. Electrical power is provided by silver-zinc primary batteries. The distributed sequencing system is a subset of the Transtage system using identical components, and electrical isolation is maintained between the BV and Centaur. The SCI remote multiplexed instrumentation system (RMIS) provides 384 kbps of BV telemetry to allow accurate analysis of all BV systems. The telemetry data are transmitted via a 10-W S-band transmitter and a single broad beam antenna. The telemetry system is also completely independent of the Centaur. TIV tracking is provided by a C-band pulse beacon in the Centaur. Command control receivers in the Centaur provide shutdown and destruct commands to the BV. The Centaur commands are isolated from the BV electrical system, which provides power to the shutdown and destruct circuits. Inadvertent separation destruct systems are included in both core vehicle stages and in each SRM.

The primary core vehicle avionics modifications (Table C8-6) are from a T34D/Transtage baseline.

Table C8-7 lists the major system components. The majority of the components are located on two trusses in Stage II Compartment 2A.

Table C8-6 Avionics Modifications

Design Modification	Description	Design Impact	Performance Impact	Constraints	Risk
Eqpt Location	Transtage avionics eqpt relocated to Stage II compartment	Transtage trusses must be redesigned to mount in new location.	None-Baseline	Modified guidance truss dynamic charac-teristics must be analyzed for compatibility with the IGS.	Original truss design was subject to the same constraints
Cordage	Modify cordage to accommodate new eqpt locations & stage lengths.	Must resize critical ordnance circuits & analysis new clearane & installation constraints.	None-Baseline	Lack of data on Centaur & PLF inter-face.	Similar designs & installa-tions have been successfully accommodated on T34D.
Flight Control System Sensors	Delete compartment 2B truss, LASS & Stage II RGS	Modify soft-ware to resolve IMU sensed acceleration into body coordinates. Redesign autopilot.	Reduced BV Dry Weight (Approx 90 lb)	Autopilot adequacy must be proved by analysis.	Previous studies & preliminary analysis indicate acceptable margins.
Tracking & Flight Safety System	The BV contains no C-band tracking aid & no command control receivers. These functions are Provides by Centaur	Must add required shutdown & destruct interface with Centaur. Centaur has C-band beacon	Reduced BV Dry Weight (Approx 34 lb)	Lack of data on Centaur interfaces & flight safety system design.	Centaur system is being added for TIV application & will be designed to accommodate this approach.

Note:
Design impacts are secondary effects caused by the primary design modification.

Table C8-7 Use of flight-proven T34D components minimizes development risk.

Component/Assembly	Description	Flight Quantity	Previous Use	Model Number	Modifications Required	Supplier
Missile Guidance Computer (MGC)	Digital Computer Memory: 16k Word Size: 24-bit Type: Fixed Point Throughput: 100 kips I/O: Tailored to T-III	1	T34D/ Transtage	Magic 352	Minor* (HMIL, Cores)	Delco
Inertial Measurement Unit (IMU)	4-Axis Gimballed Inertial Platform AIDR: 6.45 meru (3σ) ASDR: 10.92 meru/g Accel S.F.: 180 ppm Accel Bias: 78 ppm	1	T34D/ Transtage	Carousel-VB	Minor* (HMIL, Thermal Switch, Relay)	Delco
Stage I RGS	2-Axis Rate Gyro Range: ± 20 deg/s	1	T34D	C8801030000	None	Martin Marietta
Hydraulic Actuators	Linear, Mechanical Feedback, Hydraulic Actuators	Stage I:4 Stage II:2 Stage IIR:1	T34D T34D T34D	P04600008 P046S0002 P046S0003	None	Moog
Turbine-Driven Hydraulic Pump	Flight Hyd Pump Stage I: 15 gpm Stage II: 5 gpm	Stage I:1 Stage II:1	T34D T34D	P048S0089 P048S0087	None	New York Airbrake
Motor-Driven Hydraulic Pump	Ground Checkout Hyd Pump, 3 gpm	2	T34D	P048S0084	None	Vickers
Accumulator Reservoir	Maintains System Hydraulic Pressure & Level	2	T34D	P048S0090	None	Cadillac Gage
RMIS Converter	PROM Programmable Instrumentation Converter Unit	1	T34D	-	Minor* (HA2820)	SCI
RMIS RMU	Remote Multiplexer Unit, 32-Channel	9	T34D	-	None	SCI
10-V Power Supply	IPS	1	T34D	P094S0016	None	Gulton
S-band Transmitter	10-W Transmitter	1	T34D	P06400474	None	Teledynamics
APS/IGPS Battery	40-Ah Battery	1	T34D	P09400063	None	Eagle Picher
IPS/TPS/CCPS Batteries	4-Ah Batteries	6	T34D	P094S0028	None	Eagle Picher
Relay Assembly	29-Relay Package	2	T34D	81600025000	None	Martin Marietta
Static Inverter	Provides 800 Hz, 26 Vac Power to RGS	1	T34D	81600027000	None	Martin Marietta

*Minor Modifications to Accommodate Piece-Part Replacements due to Nonavailability of Current Parts

The autonomous IGS was chosen for the BV as a result of a trade study that evaluated the cost, weight, risk, and technical merit of several candidates (Table C8-8). The proposed digital system provides more flexibility in control system and load alleviation design than the analog system, minimizes the interfaces with Centaur, and minimizes the schedule impact on critical Centaur hardware and software. This configuration provides flexibility for integrating and flying alternative upper stages or payloads requiring no upper stage. The BV/Centaur interface is limited to the flight safety system.

Table C8-8 The Delco Systems Transtage IGS

Factors	Delco USGS (Transtage)	Analog FCS (T-IIIE)	Centaur IGS	Centaur IGS/MGC
Cost	$5.5M	$1.0M	$2.2M	$5.2M
Weight	246 lb	130 lb	109 lb	168 lb
Schedule Risk	Low	Moderate	High	High
Autopilot/Load Relief Flexibility	Good-Digital A/P has max flexibility	Poor-Analog A/P is inflexible	Good-Digital A/P has max flexibility	Good-Digital A/P has max flexibility
Complexity of Centaur I/F	None-No Functional Interfaces	Moderate-9 Functional Interfaces	High-33 Functional Interfaces	Moderate-3 Functional Interfaces
S/W Impact	Low	Low	High	Moderate
S/W Margins	Good	Good	Poor	Good
New H/W	None	Added Relays in Centaur	Major Mod to Centaur DCU and/or New I/O Unit	Major Mod to Centaur DCU or IMG

C8.2.6 <u>Core Vehicle Software</u>

The TIV core vehicle software evolves from the flight-proven Titan 34D/Transtage flight plan XV.

The core vehicle software is derived from the T34D/Transtage software by deleting Transtage-unique functions and adding pitch and yaw bias equations. The transition to TIV software will employ proven code development methods and verification and validation testing tools.

The core vehicle software provides guidance and control to the boost vehicle and is designed to meet all mission requirements, maximize day of launch availability, and support the launch timetable and window.

The design approach is the Titan "flex launch" concept that was proven with 14 flights using a single flight program version that flew five different mission-peculiar parameter sets.

New aspects to the software include a technique adapted from the Titan IIIE program for tailoring the Stage 0 steering profile to the premeasured winds aloft and derived body rates and accelerations from the core IMU data that permit the removal of the Stage II rate gyro and LASS equipment. These modifications provide pitch and yaw bias that enhances performance and alleviates aerodynamic loads. Addition of the identified software features erodes the current available spare memory. To provide safe margins of spare memory and to accommodate any further requirements, all Transtage-unique equations will be deleted. This deletion is minimal risk, and design analyses indicate that this approach will increase spare memory margins from 8 to 11%. This solid foundation, coupled with Titan proven development and test methods, will provide a low-risk flight software package.

Newly programmed equations will be executed in ICS with nominal and max/min data to assure that coding precision is maintained. The integrated boost vehicle flight program and parameters over the range of TIV missions will be validated using MSSS, ICS, and the Scientific Simulation Lab/Guidance Control Lab (SSL/GCL) real-time test bed. Delco will perform the IV&V of core vehicle software.

This design was selected because of the flexibility to minimize vehicle loads through use of a digital autopilot and guidance techniques tailored to the launch-day wind environment. It provides low development risk by minimizing impact on Centaur Software and interfaces (see Tables C8-9 and C8-10 and Figures C8-17 and C8-18).

Table C8-9 Software Changes from Vehicle Configuration and Performance Improvements

Design Modification	Description	Design Impact	Performance Impact	Design Limitations/Constraints	Risk
Delete Transtage	Remove Transtage-related guidance & flight control equations.	- Increases spare memory by 955 words.	N/A	- Affect PRD Vol I, II only.	- Removes possibility of executing dead code.
Stage 0 Steering	Incorporate attitude polynomial as function of altitude for loads alleviation.	- Replaces existing time polynominal. - Requires day-of-launch parameter uplink.	- Minimizes aero loads due to winds. - Enhances launch availability.	- Polynomial Fit through Parameterized Altitude Bands. - Related to Launch-Day Environment.	- Technique is derived from Titan IIIE.
LASS/Stage 2 Rate Gyro Deletion	Derive body rates & accelerations from IMU data.	- Removes LASS & rate gyro hardware. - Compute inertial to body transformation in software. - Software simulates RG for vehicle checkout.	- Provides avionics savings. - Improves payload capability.	- Data Supplied at 40-ms rate. - Lower Quality Rate for A/P Compensation. - Timing & Sizing Impact, But Have Adequate Margin.	- Low, although some increase in lag & signal noise is introduced.
Payload Fairing Separation	Move jettison time com Stage I.	- Changes separation criteria.	- Increases payload capability by 530 lb. - Reduces FMH dispersions at separation.	- Satisfies SV FMH constraint.	- Low because jettison is well before Stage I/II separation.
Parameter Uplink	Uplink day-of-launch parameters derived from wind data.	- Requires off-line generation/verify network - Modifies launchday procedures.	- Enhances launch availability. - Stage 0 steering tailored to winds aloft.	- Requires day-of-launch wind data. - Requires adequate time for generation, uplink, verify.	- Low because is implemented like T340/TS GMT uplink.

Table C8-10 TIV Booster Software

S/W Function	T34D/TS Baseline	TIV Net Increase	New/Modified Items
G&N (PRD Vol I)	4,830	+80 Inst +120 Data No Change +100 Inst -503 Inst -88 Data -291 Net Chg	Stage 0 Pitch/Yaw Steering as Function of Altitude. Modify PLF sequence. Derive body rates & accel due to LASS & rate gyro removal. Delete transtage-unique equations.
Flight Controls (PRD Vol II)	4,400	+45 Inst -243 Inst -121 Data -319 Net Chg	Modify equations affected by LASS, rate gyro removal. Delete Transtage-unique equations.
Ground C/O (PRD Vol III)	5,038	+40 Inst +5 Data +20 Inst +5 Data +20 Inst +5 Data +95 Net Chg	Modify existing uplink routine to accept winds aloft derived Stage 0 steering parameters. Vehicle checkout affected by rate gyro deletion. Add logic to verify uplink parameter load.
Other Program Data Spare Locat+ons	757 1,359 (8.3%)	No Change +515 (11.4% of Memory)	
Total	16,384	16,384	

C8.2.7 Centaur

The TIV Centaur is the NASA STS Centaur G Prime design modified to fly various Earth-orbit missions and to interface with the satellite vehicle, core vehicle, and Launch Complex (LC)-41.

Modifications to the Centaur G Prime include structural modifications to interface with the core vehicle and the SVs; propulsion modifications to provide sufficient commodities for a three-burn mission and to fit launch complex orientation; and avionics modifications to perform multiburn missions and meet tracking and range safety requirements. Software modifications account for the differences in configuration and mission profile between TIV and STS Centaur vehicles.

Table C8-11 is a summary of the modifications required to achieve the TIV Centaur configuration. The Centaur G Prime forward adapter is replaced with the Centaur G forward adapter that is lengthened approximately 17 in. The G adapter accommodates the three SV mechanical attachment interfaces, and the modification incorporates provisions for the forward bearing reactor (FBR) system. An interstage adapter (ISA), based on the Titan IIIE/Centaur D-1T design, provides the interface with the core vehicle and the linear-shaped charge (same as used on Titan IIIE/Centaur D-1T) separation system. The LH2 tank insulation system is modified to take advantage of the reduced TIV thermal environment as compared to the STS operation. One of the two layers of polyimide foam blankets used on Centaur G Prime is removed, resulting in a weight reduction of 30 lb.

TIV Centaur multiburn missions require an additional hydrazine tank and two additional helium storage bottles compared to G Prime. The hydrazine tank, support system, and piping designed for Centaur G are incorporated into the TIV Centaur. The two 26-in.-diameter helium bottles do not fit within the space limitations of the engine compartment and are located external to the LO_2 tank and are supported from LO_2 tank fore and aft rings. Piping modifications are made to connect the added tanks to the existing two helium tanks in the Centaur G Prime pressurization system. For STS operation, all umbilicals are routed to disconnect panels adjacent to the separation plane where they mate with systems on the Centaur integrated support system (CISS). For TIV operation, umbilicals are reoriented to be compatible with the launch complex. Along with the reorientation, a 95-lb weight savings (95-lb payload gain) is realized by eliminating the external fuel and oxidizer fill and drain piping to the CISS interface and using tank-mounted disconnects at the tank penetrations as used on Titan IIIE/Centaur D-1T vehicle. An additional weight saving is realized by deleting the zero-g vent systems from Centaur G Prime and using Centaur D-1T vent systems. Deleting the LH_2 thermodynamic vent system and the LO_2 jet pulse mixer results in a 135-lb weight savings (135-lb payload gain).

TIV Centaur draws heavily from the STS Centaur G and G Prime vehicles. Both of those vehicles are scheduled to complete development and fly missions in one and two years, respectively, before the initial

Table C8-11 Modifications to Centaur for TIV

Design Modification	Description	Design Impact	Performance Impact	Design Limitations/ Constraints
Forward Adapter	Replace with lengthened G adapter with provisions for FSR.	Increase cylinder section by approx 17 in. Add FBR suppt & remove truss support system.	Baseline for SLV-X	Mounting Space for Added Avionics Components & FBR loading
Interstage Adapter	New adapter similar to Centaur D-1T with ordnance separation system & Stage II interface.	Strengthen adapter for higher loads, redesign umbilical supports, & test umbilical & separation system.	Weight Reduction +150-lb Payload	None
LH_2 Tank Insulation	Remove one layer of foam.	Radiation Shield Geometry Changes	Weight Reduction +30-lb Payload	None
Increase Hydrazine Capacity	Incorp second Centaur G hydrazine tank.	Existing Centaur G Design	Baseline for SLV-X	None
Increase Helium Capacity	Add two 26-in. dia helium spheres.	Support Structure & Pipe Routing	Baseline for SLV-X	Locate external to the LO_2 tank. Verify tank capability.
Replace Fill/Drain & Dump Sys with D-1T Umbilical System	Delete external pipe & use tank-mounted disconnects.	Umbilical Chutes in Payload Fairing	Weight Reduction +95-lb Payload	Use D-1T umbilical design concepts. Test to verify.
Use D-1T Vent Systems	Delete internal & external zero-g vent components & vent O_2 & H_2 as for D-1T.	Payload Fairing Vent Fin & ISA Penetration	Weight Reduction +135-lb Payload	Use D-1T venting design concepts.
Delete STS-Unique Avionics Packages	Delete DUFTAS & star scanner.	Wiring, Harness, Routing, & Box Mounting	Baseline for SLV-X	Verify mission accuracy with no attitude update.
Add Pyro Firing Circuits for SV	Replace PICUs with PYCs.	PYCs have more channels.	Baseline for SLV-X	None
Add C-band Tracking Capability	Add C-band system similar to Centaur D-1T & D-1A.	Wiring, Harness, Routing, & Box Mounting	Baseline for SLV-X	Provides 12-dB min signal-to-noise ratio between vehicle & tracking ground station through park orbit insertion.
Add Range Safety Command Capability	Add receivers, antennas, batteries, & destruct package.	Wiring, Harness, Routing, & Box Mounting	Baseline for SLV-X	Space for mounting packages.

operational capability for TIV. The limited modifications required for TIV to the STS G and G Prime versions of Centaur minimize the technical risk associated with the Centaur upper stage.

The TIV upper stage baseline combines the Centaur G Prime tanks, structure, and propulsion with Centaur G forward adapter and avionics. The Centaur G Prime tanks and engines are used without change (Fig. C8-19).

The Centaur G forward adapter, which includes the required multiple satellite vehicle (SV) mechanical attachments, will be lengthened to clear the LH_2 tank dome and will be installed on the Centaur G Prime tank structure. A modified Titan IIIE Centaur D-1T interstage adapter (ISA) will provide the interface with the booster Stage II and include the ordnance system for separating the Centaur stage. Electrical and fluid umbilicals will be modified to be compatible with the orientation of the vehicle on Launch Complex 41.

The TIV Centaur stage design is based on proven hardware and software developed for the Centaur G and G Prime vehicles. Figure C8-20 shows the TIV Centaur configuration with major subsystems and features identified. The stage is 29.3-ft long with a maximum diameter of 170 in. Dry weight of the stage is 6,120 lb, and loaded weight is 52,585 lb. Major hardware items are summarized in Table C8-12.

The stage uses LO_2 and LH_2 cryogenic propellants with two Pratt and Whitney RL10A-3-3A engines delivering 16,500-lb thrust, each with an I_{sp} of 446.3 s. The pressure-stabilized tanks are of welded high-strength stainless steel construction with propellant capacity of approximately 46,000 lb. The hydrazine reaction control system (RCS) consists of twelve 6-lb thrust units and two positive expulsion tanks with 340-lb hydrazine capacity.

The stage is attached to the TIV core vehicle through an interstage adapter. The LH_2 tank insulation system consists of a single 3/4-in. polyimide foam blanket with aluminized Kapton/fiberglass laminate radiation shield.

Missions specified require a three-burn capability for TIV Centaur and derive requirements for additional helium pressurant and hydrazine RCS propellant.

The TIV Centaur uses the DOD STS Centaur G avionics system with minor augmentations to meet range requirements. Use of the Centaur G avionics system (which is configured to meet DOD requirements) for the TIV Centaur assures compliance with DOD functional requirements. The TIV Centaur avionics system includes electrical power, guidance, navigation, control, instrumentation, secure telemetry, tracking, and range safety command destruct subsystems. The system (Fig. C8-21) is configured around the digital computer unit (DCU), the inertial reference unit (IRU), the sequence control unit (SCU), and the servo inverter unit (SIU). Propellant use and tank venting and pressurization are also controlled by the avionics system.

Table C8-12 STS Centaur can be adapted for TIV with only limited modifications

Component/Assembly	Description	Flight Quantity	Previous Use	Model Number	Modifications Required	Supplier
Propellant Tanks	120-in. Dia LO_2 & 170-in. Dia LH_2 Pressure-Stabilized Tanks	One Set	STS Centaur	G Prime	None	GDC
Main Engines	16,500 lbf at 446.3S I_{sp}	Two	STS Centaur	G Prime	None	Pratt & Whitney
Forward Adapter	Modified Centaur G	One	STS Centaur	G	Add 16.88 in. to Length & Prov. for FBR; Delete Truss Supt Sys	GDC
Aft Adapter/ Interstage Adapter	Cylindrical Structure between LO Tank & Stage II with Sep Sys	One	T-IIIE	D-1T	Reduce Length & Strengthen for TIV Loads	GDC
Tank Insulation System	Purged Foam Insulation with Radiation Shield	One	STS Centaur	G Prime	Remove One Layer of Foam	GDC
Reaction Control System	Hydrazine with Twelve 6-lbf Thrusters & 340-lbm Propellant Capacity	One	STS Centaur	G Prime	Increased Hydrazine Storage Same As Centaur G	GDC
Fluid Umbilicals	Piping & Disconnects Interfacing with Ground Systems	One	T-IIIE	D-1T	Reroute Piping & Relocate Disconnects	GDC
Avionics System	Computer-Controlled Inertial System with Battery Power, C-Band Tracking, Encrypted PCM Telemetry, & Range Safety Command	One	STS Centaur	G	Use Centaur G Sys with C-Band & Range Safety Command Destruct from Centaur D-1A	GDC
Flight Software	Modularized Software in Which Each Module Performs Specific Functions	One	STS Centaur	G	Use Centaur G Sys with STS-Unique Modules Deleted & Specific Modules for Configuration & Mission	GDC

Seven pyrotechnic initiator control units (PICU) in the Centaur G system are replaced with a like number of pyrotechnic control units (PYC) used in the Centaur D-1A. PYCs provide more ordnance firing channels for the SV interface and are lighter than PICUs. PICU additional safety circuitry and telemetry interfaces required for STS operation are not necessary for TIV. The C-band tracking system and the range safety command destruct system from the Centaur D-1A are added to provide ground tracking and range safety control capability. Mounting provisions are included for three 150-Ah batteries and a power transfer unit for spacecraft power.

The TIV Centaur uses the basic Centaur G software system and flight modules. Modifications to software modules account for the vehicle configuration and mission differences between Centaur G and TIV Centaur. STS-unique modules, such as navigation update, strapdown torquing, attitude update, and predeployment check, are removed for TIV Centaur. TIV Centaur software module derivation is shown in Table C8-13. The table identifies the modules that (1) normally require mission-peculiar modification; (2) are unique to TIV Centaur; (3) are common to Centaur G and TIV Centaur; and (4) are deleted from the Centaur G baseline.

The flight software is structured in modules where each module performs a unique function. Individual modules are selected from a configuration controlled library and assembled into an integrated flight program. Table C8-14 lists the flight modules for an TIV geostationary mission with their estimated DCU memory requirements. Target orbit constants (such as apogee, perigee, inclination) are loaded into the DCU from a constants tape. This permits the generalized guidance equations to accomplish a broad range of missions by specifying the appropriate target data. The sequencer module issues time-related discretes for both hardware and software. The design is such that most changes can be accomplished by a change of constants in the sequencer table.

Table C8-13 Flight software requires minimum change from STS Centaur G baseline

Function	Mission Peculiar	Mod for TIV	TIV/G Common
Operating System			
- I/O, Discretes, Executive, Math Routines		X	
Navigation Functions			
- Navigation			X
- Navigation Update		Delete	
- Platform Rotation			X
- Strapdown Torquing		Delete	
- Attitude Update		Delete	
- Accelerometer Bias Calibration			X
Guidance Functions			
- Powered Phase			X
- Coast Phase		X	
- Post Injection	X		
- Steering Interface			X
Autopilot Functions			
- Attitude Error & DDR			X
- Powered Phase			X
- Coast Phase			X
Sequencing Functions			
- Sequencer		X	
- Discrete Priority			X
Vehicle Controls & Monitors			
- CCVAPS			X
- Propellant Utilization			X
- Hydrazine Monitor			X
Telemetry			
- PCM Formatter	X		
- Submultiplex Formatter			X
- Antenna Selection			X
- Telemetry Recovery			X
- Vehicle Telemetry			X
Flight Initialization			X
Slack Task			X
Predeployment Checks		Delete	
Constants	X		

Table C.8-14 TIV Centaur software uses modules from configuration-controlled library

Flight Module	Core Storage Required, Memory Cells	
	Nonalterable	Alterable
Attitude Error & DDR	133	19
Coast Autopilot	305	51
Coast Guidance	576	67
Nonalt Constants & Sequence Data Table	1,356	0
Computer-Controlled Vent & Pressurization	889	150
Flight Initialization	174	1
Powered Guidance	1,689	253
Hydrazine Monitor	72	12
Navigation	536	142
Powered Autopilot	234	35
Vehicle Telemetry	250	0
Post Injection	135	3
Platform Rotation	149	22
Propellant Utilization	269	31
Discrete Priority	77	1
Operating System	1,400	640
Steering Interface	776	118
Telemetry Recovery	100	100
Vehicle Sequencer	1,112	53
Antenna Selection	250	5
Accelerometer Bias Calculation	160	25
Slack Task	22	5
PCM Telemetry Formatter	1,024	0
Telemetry Formater (Submultiplexer)	192	679
Total	11,880	2,412
Available	12,288	4,096

C8.2.8 Payload Fairing (PLF)

The TIV payload fairing (PLF) is an all-metal structure of isogrid construction.

The PLF is 200 in. in diameter and 86 ft long (maximum) and weighs 12,074 lb, including 1,089 lb of acoustic blankets.

The PLF is required to provide Shuttle bay equivalency with optional lengths to accommodate 20- to 40-ft-long payloads and to satisfy payload and Centaur requirements.

The PLF is a trisector design that consists of two primary sections (Centaur and payload compartments), a biconic nose, and a contamination-free thrusting-joint separation system (Fig. C8-22).

The fairing is constructed of aluminum isogrid.

Nose, cylinder, and aft-body sections bolt together to form three fairing segments that run from nose to tail. These segments are mated together by the fairing separation system in the same manner as the Titan 34D fairing. In flight, an ordnance-initiated gas-bellows expands and separates the fairing into three sections and provides the required separation velocity. The major design changes from Titan 34D are the addition of a strand of linear charge and additional and larger separation studs at the base section. Chutes will be provided to house Centaur umbilical lines for air conditioning, electrical lines, and propellant fill and drain.

C8.2.9 Payload Environments

TIV payload environments are equivalent to or more benign than the STS/Centaur environments.

TIV is designed to produce payload environments equal to or less severe than those experienced on the Shuttle. Figure C8-23A compares the payload acoustics environment for TIV with the system specification requirement and the STS levels. The predicted TIV levels are below both the STS and the required specifications. The overall level of 136.1 dB versus the STS value of 138.0 dB assures the payload an TIV level that is within the design limits for STS.

The shock level induced on the payload by the separation shock events is the same for TIV as a payload would experience on STS/Centaur. Figure C8-23B shows that the shock level for the TIV will peak at 2000 g's at approximately 5000 Hz, which is lower than the 7500 g's a payload must withstand if flying on the STS/IUS.

The thermal environment produced by the TIV is moderate when compared to the STS. Figure C8-23C compares the TIV and STS prelaunch and flight payload thermal environments. The TIV prelaunch environment is selectable over the range of 50 to 80°F. The internal wall temperature of the PLF reaches a moderate maximum temperature of 98°F at the 250s PLF jettison time. This compares to the 150 to 210°F orbiter payload bay temperature that a payload must experience for a minimum of 1 h and up to 3 h before the bay doors are open. The free molecular heating rate at time of PLF jettison will be less than the nominal value of 100 Btu/h-ft^2. The thermal orientation during park and transfer orbits is selectable by the user and provides flexibility in keeping the payload at moderate temperatures.

The reduced exposure time and contamination sources of TIV minimize the contamination concern for the payload (Fig. C8-23D).

The TIV Stage II retrorockets are approximately 43 ft aft of the TIV/SV interface, and the payload will be blocked from the impingement by the 14-ft diameter of the Centaur. The remainder of the flight is similar to the free-flight portion of the STS/Centaur mission.

Figure C8-23E compares the pressure decay rates of the payload compartments for the TIV and the STS. Because the STS vent door does not open until 10 s into flight, the STS maximum decay rate of 0.5 psi/s is calculated for the TIV. Therefore, payloads that can tolerate the STS rate will be compatible with the TIV pressure decay rate.

The limited number of RF sources (Fig. C8-23F) on the TIV simplifies the EMC/EMI concerns for the user. While the Shuttle uses numerous transmitters and antennas, the TIV will employ only the BV S-band, the Centaur S-band and the Centaur C-band sources. The maximum expected level of 11.3 V/m for the TIV is less than the 18.0 V/m expected on the STS.

Dynamic loads are another environment that is a concern to the user. Figure C8-23G provides flight event load factors for the TIV payloads. The -3.2-g axial load factor for the STS is more severe than the -3.0 g for the TIV, while the lateral load factors are identical. Other TIV load events are potentially less severe than STS loads. Satellite vehicle basic structures designed to withstand the STS loads will be compatible with the TIV.

C8.2.10 Payload Interfaces

TIV uses interface hardware identical to the STS/Centaur G. The TIV payload mechanical and electrical interfaces are essentially identical to those provided on the STS/Centaur G. The three options for the mechanical payload adapter interfaces of the TIV are presented in Figure C8-24A.

The eight-hole pattern accommodates payload designed for either the Transtage or the IUS.

The physical access to the electrical interfaces is provided by the 10 electrical interface connectors listed in Figure C8-24B. These connectors provide the interface for the mission-unique harness.

The electrical interfaces are summarized in Figure C8-24C. The TIV provides electrical interfaces for payload power, communications, discretes, ordnance firing, telemetry, and power and data to the payload AGE through a TIV umbilical.

TIV accommodates up to 8.4 kWh total of SV electrical power compared to the requirement of 6.0 kWh. This power is provided by three batteries that are single-failure tolerant, and weight of the batteries is chargeable to the SV.

To provide event command to the payload, the TIV has the capability to issue up to 16 discretes. These discretes can be referenced to mission time or to other mission events. The TIV also provides 10 primary and 10 backup ordnance firing circuits to activate payload pyrotechnics. Details of the discrete and the ordnance power are shown in Figure C8-24C.

C8.2.11 TIV Mass Properties

There is high confidence in mass property predictions because 80% of the TIV dry weight is based on extensions to existing systems.

Table C8-15 summarizes TIV weights and percent change from previous existing Titan programs. As shown, the TIV lift-off weight is 1,910,499 lb and represents an increase of approximately 25% over a typical Titan 34D launched from CCAFS.

Table C8-15 TIV Weights and Percent Changes from Previous Existing Titan Programs

System	Existing Titan Wt. lb		TIV-X Wt, lb		Existing System, %
	Item	Total	Item	Total	
SRM		1,104,038		1,390,772	
- Dry Wt	157,088		188,518		83
- Expendables	946,950		1,202,254		
Core (Step 1)		314,136		359,155	
- Dry Wt	16,270		18,395		88
- Expendables	297,866		340,760		
Core (Step 2)		74,101		85,913	
- Dry Wt	6,280		8,871		71
- Expendables	67,821		77,042		
Centaur Upper Stage		34,984		52,585	
- Dry Wt	3,795		6,120		62
- Expendables	31,189		46,465		
Payload Fring		4,759		12,074	39
Satellite Vechicles				10,000	
Total Vehicle				1,910,499	

TIV mass properties comply with MIL-M-38310B requirements. The approach to mass properties control for TIV is the same as that used on current Titan programs. The TIV mass properties reference diagram (Figure C8-25) complies with MIL-M-38310B and defines the step/stage relationship used in defining mass properties.

There is high confidence in TIV mass property predictions because most component/assembly weights and dimensions are based on calculated or measured values.

Table C8-16 Predicted TIV Mass Properties

TIV Dry Weights	Weight, lb — Dry Weight — Reference Measured Weight	Delta Weight	Total Dry Weight	Remarks
Step 0 (Solid Rocket Motors)	157,088(1)	(+31,430)	188,518	Note:
- 5 1/2 to 7-Segment SRM		+31,430		
Step I (Core 1st Stage)	16,270(1)	(+2,125)	18,395	(1) Reference weights extracted from T34D RGS status report no. 13 (MCR-80-113).
- Oxidizer & Fuel Tank Stretch		+759(2)		
- Structural Beef-Up		+1,507(3)		
- SRM Attach Kit Removal		-141		
Step II (Core 2nd Stage)	5,972(1)	(+2,899)	8,871	(2) Estimate based on 51.2 in. increase in Oxidizer tank length and 43.8 in. increase in fuel tank lengths.
- Forward Skirt Stretch & Beef-Up		+932(4)		
- Oxidizer & Fuel Tank Stretch		+157(5)		
- Structural Beef-Up		+796(3)		
- SRM Attach Kit Addition		+174		
- Centaur I/F Hardware		+213		(3) Estimate based on required P_{EQ}/stress analysis beef-up.
- Transtage IGS & Power Gen		+132(6)		
- Instrumentation Additions		+17(7)		
- PLF Backup Structure		+297		(4) Estimate based on skirt stretch of 43.1 in.
- Separation & Destruct Moods		+14(8)		
- Thermal Barrier		+135(8)		(5) Estimate based on 9-in. increase in Oxidizer tank length and 8-in. increase in fuel tank.
- Ablative Skirt & 5% Thrust Inc		+32		
Payload Fairing				
- 86 ft by 16.7 ft dia	4,759	+7,315(9)	12,074	
Step III (Centaur Upper Stage)	6,570(10)	-450	6,120	(6) Extracted from Titan 34D Transtage weights for IGS & power generation, includes 2nd truss.
- Centaur Weight Reductions		-450(11)		
Satellite Vehicle	10,000(12)		10,000	(7) RGS modified to include minitruss.
				(8) Based on Titan 23E weights.
				(9) Estimated based on Titan 34D standard fairing (10 1/2-ft dia) isogrid.
				(10) Centaur G Prime Control Weight.
				(11) Estimate based on centaur G Prime weight reductions.
				(12) Spec required for GSO mission.

Weight growth is included in the performance margin calculations and were established for dry weight as defined in Table C8-17. Because extensions to existing hardware are used, the delta weights between the existing design and new TIV design are known. To protect against uncertainties, weight growths of up to 13% for new or modified TIV designs are provided. Based on previous Titan experience, their growths are more than adequate. However, because Centaur weight growths translate almost directly to payload capability and only a 4% growth is provided for Centaur, the Centaur dry weight has been identified as a medium risk to the TIV program.

Table C.8-17 TIV Weight Growth Values

Item Dry Weight	Weight Growth: Previous Titan Concept-to Actual Percentage	TIV: Delta Wt, lb	TIV: New Design, %	TIV: Wt Growth, lb
Step 0 (SRM)	1	31,430	11	3,369
Step I	6	2,125	7*	151
Step II	4	2,591	9*	223
Step III (Transtage)	7 (Info)	-	-	-
Payload Fairing	No Growth	7,315	13	966
Step III (Centaur)	No Growth	2,325	4	100

*Resultant percent based on 10% of TIV new design without barrel length weights included.

C8.3 MISSION SCENARIO

C8.3.1 Performance

The TIV exceeds the required payload weight capability requirement. The TIV nominal payload weight capability is 10,565 lb for the GSO mission; 17,819 lb for the 12-hour inclined mission; 10,918 lb for the 24-hour inclined mission; and 10,565 to 12,790 lb for the mission range of final orbit eccentricity and inclination as shown in Figures C8-26A and C8-26B. This exceeds the minimum required performance and includes allowances for flight performance reserve, launch temperature variations, 3-DOF to 6-DOF simulations variations, and capability excess for class-to-TAG vehicle variations. The specification columns in Figure C8-26A present performance for missions specified in the RFP. Alternate missions consider conditions listed in the bottom of the table. Alternate 1 for the 12-hour inclined mission used a launch azimuth that minimizes range safety hazards by positioning the impact points farther from the east coast and through less dense population areas on the European continent. Alternate 2 used the same launch azimuth but imposes an argument-of-perigee constraint of 270 degrees. The alternate for the 24-hour inclined mission also used a different launch azimuth and part orbit inclination. A 3-DOF trajectory optimization program, incorporating Titan IIIC and Titan 34D/Transtage trajectory shaping concepts, was used for performance determination.

The flight performance reserve (FPR) 3-DOF analysis considered 11 major vehicle dispersions based on previous Titan studies and an allowance for lesser effect items not simulated. For each dispersed booster vehicle parameter, the change in Centaur first-burn propellant was determined. These were adjusted for 3-DOF to 6-DOF simulation effects and made equivalent to Centaur third-burn propellant. Figure C8-26C shows these quantities and the resulting FPR for the three missions.

A contingency allowance for growth and analysis is shown in Figure C8-26D. The boost-vehicle weight growth is approximately 10% of the modified structure for SRMs, Stages I and II, and PLF. The analysis contingency for the SRMs is equivalent to a 1/2-sigma dispersion in web action time; the liquid rocket engines are approximately 50% of the performance improvements in Stage II thrust from Titan 34D. Total Centaur contingency is 100 lb. These contingency values, when combined, provide the total TIV concept-to-design values at the bottom of the table.

Combining the FPR and contingency requirements with reserves for target-to-launch, 3-DOF to 6-DOF simulation differences, and class-to-TAG effects, the total Centaur propellant margin requirements are as shown in Figure C8-26E. The effects these requirements have on payload weight capability are included in Figure C8-26A.

C8.3.2 24-Hour Geostationary Mission

The TIV configuration and mission profile data established as the performance baseline for the 24-hour geostationary mission are defined in Figure C8-27A. The lofted profile of the TIV allows separation of the payload fairing during Stage I burn and compliance with all of the Figure C8-27B guideline constraints. The angle of attack and dynamic pressure constraints at Stage 0/I separation are values currently used on other Titan programs. The aerodynamic heating indicator (AHI) limit of 100 x 10^6 ft-lb/ft is 10 to 15% lower than the value for other CCAFS Titan launches. This is to ensure that a dispersed heating profile will be within design capability. The 900-psf maximum dynamic pressure constraint is compatible with the hardware design and was selected to satisfy the air-load indicator design constraint guideline of (Q_T) 1000 psf-deg for a nominal 6D vehicle simulation incorporating winds. The Q_T 1000-psf-deg guideline minimizes structural weight required by aerodynamic loads.

Figures C18-27C and C8-27D provide time, altitude, and velocity data for the GSO mission boost and orbit trajectories. This trajectory is similar to other Titan program trajectories except that it separates the payload fairing in Stage I and exhibits a slightly more lofted profile.

The guidance injection accuracy data of Figure C8-27E are from the GDC proposal and incorporate a 4-fps velocity uncertainty to account for system performance dispersion effects.

The TIV ground track for the geostationary mission is provided in Figure C8-27F (boost flight) and Figure C8-27G (orbit flight). This ground track is similar to those of other Titan GSO missions and compatible with requirements for range safety flight plan approval. The vacuum instantaneous impact points of the jettisoned bodies (Figure C8-27F) are within safety requirements for the dray impact footprints for this mission.

The TIV boost flight profile allows tracking station visibility for the required portions of flight with the exception of the last 100 s of the boost flight (Fig. C8-27H). During this time (Centaur burn into park orbit), the tracking station elevation angle is less than 5 degrees, and an ARIA aircraft may be required to support data acquisition. Similarly, the data of Figure C8-27I show that additional tracking support will be required for the Centaur transfer orbit burn (Centaur second burn). The orbit flight after transfer orbit inject through payload separation is visible to tracking stations for this first equatorial crossing trajectory.

C8.3.3 Inclined Missions

The baseline performance vehicle for the inclined missions is the same as the GSO baseline identified in Section C8.3.2. With the exception of flight azimuth, park orbit, and final orbit parameters, the mission profile ground rules of Section C8.3.2 also apply to the inclined missions.

Figure C8-28A presents the boost ascent profile that is representative for both the 12-hour and 24-hour inclined missions, because both missions use approximately the same park orbit. The orbit profiles are different for the two missions, and they are presented in Figure C8-28B (12-h mission) and Figure C8-28C (24-h inclined mission). The data in Figure C8-28D show design trajectories for both the 12-h and 24-h inclined missions to be acceptable for flight environment and vehicle design. They also provide a performance capability that exceeds requirements.

The Centaur injection accuracy data for the inclined missions are contained in Part 3, Volume 4, of the proposal submitted by GDC. These injection-accuracy data incorporate a 4-fps uncertainty in velocity at final orbit to account for vehicle performance dispersions.

The boost and orbit ground trace (Fig. C8-28E) contains vacuum instantaneous impacts points (IIP) for the 12-h mission with a park orbit inclination of 55 degrees. Figure C8-28F contains a boost and orbit ground trace for a suggested alternate flight plan to the specified 12-hour mission. The suggested alternate flight plan achieves the specified final orbit but requires less overflight of land mass and no boost overflight of Soviet Bloc countries. Also, the impact of jettisoned bodies occurs farther from land masses.

Figure C8-28G presents a boost-and-orbit ground trace with vacuum IIPs for the specified 24-hour inclined mission. For the same reasons that pertained to the 12h mission, we have presented a suggested alternate 24-hour inclined mission that achieves the same final orbit as the specified mission but uses a different park orbit inclination. The boost and orbit trace for the 24-hour alternative mission is provided in Figure C8-28H.

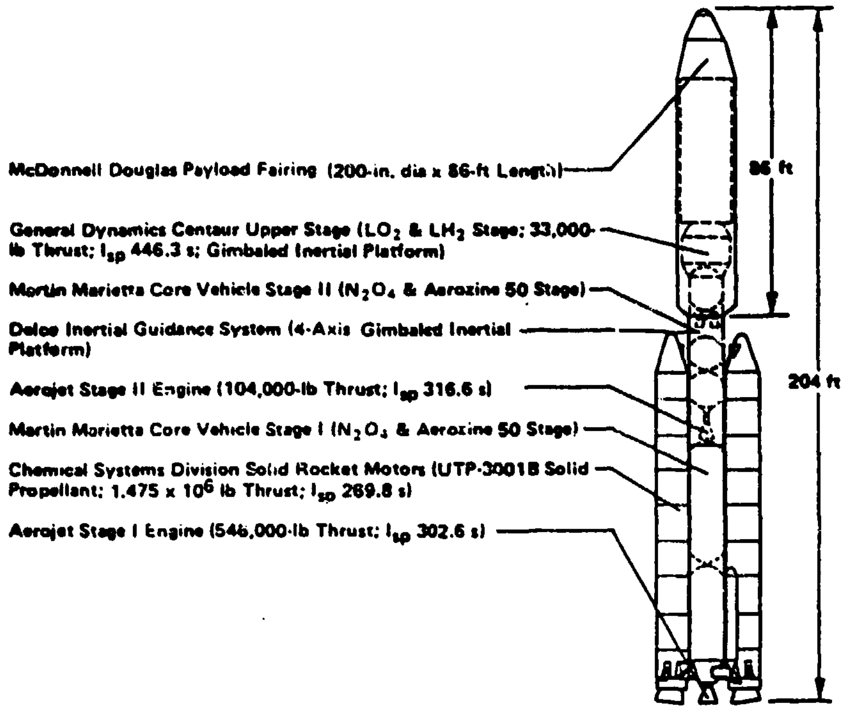
Figures C8-28I and C8-28J provide boost and orbit tracking station elevation angle data for the specified 12-hour mission. Similar data are presented in Figures C8-28K and C8-28L for boost and orbit portions of the 24-hour inclined mission. These charts show that an Advanced Range Instrumentation Aircraft (ARIA) or some other tracking aid will be needed to provide tracking coverage for some portions of the mission. The tracking station elevation angle data for the suggested alternatives to these missions exhibit only slight variance with these data.

C8.3.4 Mission Capability

TIV payload capability to GSO and inclined orbits exceeds STS/Centaur G capability. TIV provides STS/Centaur equivalent payload capability. Figure C8-29 presents the equivalence for payload performance, envelope, and structural capability. The TIV delivers 10,565 lb to a GSO as compared to the 9413 lb delivered by the STS/Centaur G and the TIV specification requirement of 10,000 lb. TIV capability to a 12-hour orbit inclined at 63.4 deg is 17,819 lb compared to the STS/Centaur G capability of 9,459 lb and the TIV specification requirement of 11,500 lb. The TIV will deliver a 10,918 lb payload to a 24h orbit inclined at 65 degrees.

TIV provides an allowable payload dynamic envelope of 15 ft in diameter by 40 ft in length. This is equivalent to the available STS envelope when a payload uses the Centaur G. Figure C8-29 also shows that there is additional payload envelope in the payload fairing nose cone.

TIV can accommodate SVs weighing 5,000-11,500 lb with the cg 150 in. forward of the TIV/SV interface, and 10,000 lb with the cg 160 in. forward of the interface. The SV cg lateral offset can be up to 8 inches from the SV centerline. The STS/Centaur accommodates the same SV mass properties.



Flight Element	Wt, lb	Reliability
SRMs	1,390,772	0.99749
Core Stage I	369,155	0.99385
Core Stage II	85,913	
Centaur	52,585	0.99 (Allocated)
Payload Fairing	12,074	0.9999
Payload (GSO)	10,000	N/A
Total	1,910,499	0.9813

Performance - Payload Weight

GSO – 10,565 lb
12-h Inclined – 17,819 lb
24-h Inclined – 10,918 lb

Performance—Polar Orbit

SLV-X Booster—32,000 lb

Payload Accommodations

Payload Size: 15-ft by 40-ft Dynamic Envelope
Payload Weight: 10,000 lb (GSO); 11,500 lb (Inclined)
Payload Environments: $\leq$ STS
Payload Interfaces: Equivalent to STS

Figure C8-1 TIV Configuration

Payload Fairing

Configuration
- Trisector Configuration with Biconic Nose Cone; Interfaces with Core Vehicle at Stage II Forward Oxidizer Skirt

Size
- 200-in. dia by 86-ft Length; Payload Compartment Dynamic Envelope of 15-ft dia by 40-ft Length. Payload compartment is modular with 20, 30, or 40-ft length capabilities.

Weight
- 12,074 lb (86-ft Length)

Structure
- Isogrid Skin with Internal Ring Frames, Stringers, and Longerons; Aluminum Alloy Materials; Acoustic Blankets

Separation System
- Contamination-Free Thrust Joint

Core Vehicle

Configuration
- 2-Stage Liquid Propellant Vehicle with Interface to Centaur Upper Stage, SRMs, & PLF

Size
- 10-ft dia by 119.1-ft Length (Stage I 86.5 ft & Stage II 32.6 ft)

Dry Weight
- Stage I 18,395 lb & Stage II 9,871 lb

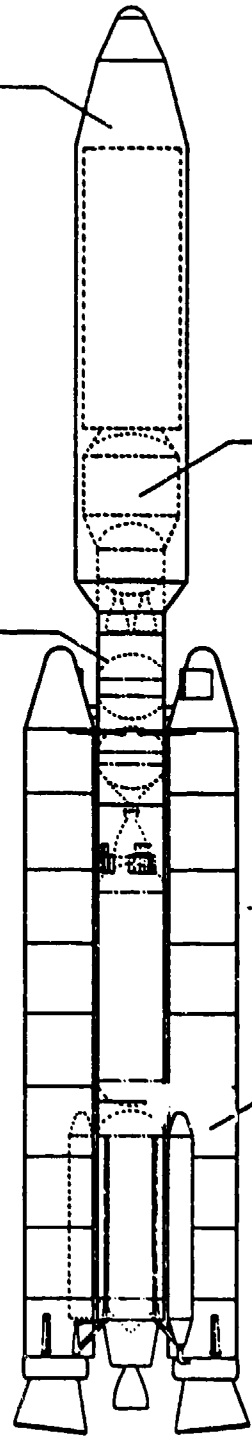
Loaded Weight
- Stage I 359,155 lb & Stage II 85,913 lb

Structure
- Skin Stringer Construction with Ring Frame Cross Sections
- PLF Attaches to Stage II Forward Oxidizer Skirt
- Aluminum Alloy Materials

Propulsion
- Propellants: N_2O_4 & Aerozine 50 (341,000 lb Stage I & 77,000 lb Stage II
- Stage I engines (LR-87-11) provide 546,000 lb of thrust with 1.9 mixture ratio, 190-s burn time & 302.6-s I_{sp}; engines gimbaled to provide steering control.
- Stage II engine (LR-91-11) provides 104,000 lb of thrust with 1.79 mixture radio, 232-s burn time, & 316.6-s I_{sp}. Gimbaled engine & roll control nozzle provide steering control.
- Autogenous Pressurization Systems

Avionics
- 16k Core Memory Digital Computer
- 4-Axis Gimbaled Inertial Platform
- 2-Axis Rate Gyro (Stage I)
- 9 Remote Multiplexer Units
- 10-W S-band Transmitter
- Silver-Zinc Primary Batteries
- Command Destruct System with Command from Centaur
- Inadvertent Separation Destruct System
- Hydraulic System (Engine Gimbals)

Software
- 16,384-Word Memory (14,510 Used)



Centaur Upper Stage

Configuration
- Single-Stage Liquid Oxygen & Liquid Hydrogen Vehicle with I/F to SV & Core Vehicle

Size
- 170-in. dia (Max) by 29.3-ft Length

Dry Weight
- 6,120 lb

Loaded Weight
- 52,585 lb

Structure
- Pressure-Stabilized Propellant Tanks
- Stainless Steel Tanks
- Graphite Epoxy Fwd & Aft Adapters
- Polyimide Foam Blankets (Enclosed by Multilayer Radiation Shield) Tank Insulation

Propulsion
- Two RL10A3-A3 engines provide 33,000 lb of thrust with a 5:1 mixture ratio and an I_{sp} of 446.3 s.
- Multiple Start Engines
- LO_2 & LH_2 Propellants (46,136 lb)
- H_e blowdown pressurization system is augmented by GH_2 from main engines.
- Hydrazine RCS with Twelve 6-lb Thrusters

Avionics
- 16k Core Memory Digital Computer
- Gimbaled Platform Inertial Measurement Group
- Two Remote Multiplexing Units & Two Signal Conditioners
- S-band Telemetry System
- C-band Tracking Transponder
- Command Destruct System
- Silver-Zinc Primary Batteries
- Secure Communications Encryptor

Software
- 12,288 Words of Nonalterable Memory (11,880 Used)
- 4,096 Words of Alterable Memory (2,412 Used)

Solid Rocket Motors

Configuration
- Strap-On, 7-Segment Solid Rocket Motors

Size
- 10-ft dia by 112.3-ft Length

Dry Weight
- 188,518 lb (Total—2 SRMs)

Loaded Weight
- 1,390,772 lb (Total—2 SRMs)

Structure
- 120-in. dia by 130-in. length motor segments are forged D6aC steel.
- Fwd & aft closures are forged of D6aC steel.

Propulsion
- UTP-3001B Solid Propellant (1,184,000 lb Total)
- Total Thrust of 2,950,000 lb (Both SRMs), with Web Action Time of 110.5 s & an I_{sp} of 269.8 s
- Total Impulse of 159,700,000 lb-s (Each SRM)
- Thrust Vector Control (TVC) Provided by N_2O_4 Injection into Motor Exhaust Stream

Avionics
- Remote Multiplexers
- Inadvertent Separation Destruct System

Figure C8-2 TIV Vehicle Configuration

PAYLOAD FAIRING

Modification	Now	Was T34D
Diameter, in.	200	126
Length, ft	86	55
Nose	Biconic	Blunt
Boattail	Have	N/A
Separation System	3-Strand	2-Strand
Umbilicals	Unique	Unique

CENTAUR

Modification	Now	Was G Prime
Structure		
– Forward Adapter	G + 17 in.	G Prime
– SV Mech I/F	3	1
– Interstage Adapter	D-1T Config	STS Unique
– LH_2 Tank Insulation	1 Layer	2 Layers
Propulsion		
– Hydrazine Tanks	2	1
– Helium Spheres	4	2
– Fill, Drain, Dump & Vent Systems	D-1T System	STS Unique
Avionics		
– Basic Avionics	G (DOD Unique)	G Prime (NASA Unique)
– STS-Unique Avionics	Delete	Has
– C-band Tracking	D-1A System	N/A
– Range Safety & Command Destruct	D-1A System	N/A

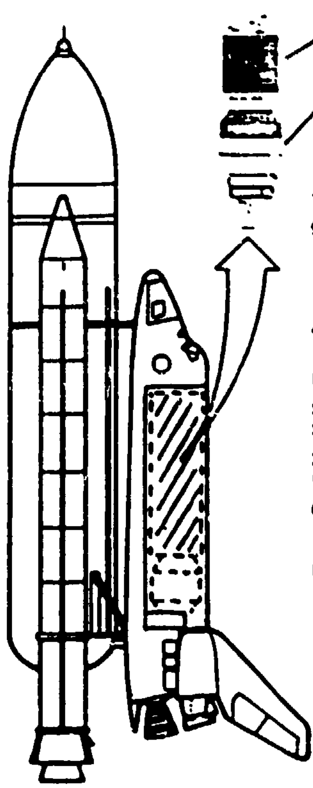
CORE VEHICLE

Modification	Now	Was T34D
Structure		
– Stage I, in.	889.5	794.5
– Stage II, in.	408.5	348.4
– Strengthen (P_{EQ})	≤64%	Std
Propulsion		
– Stage I Prop., lb	341,000	295,000
– Stage I Burn Time, s	190	165
– Stage II Prop., lb	78,000	69,700
– Stage II Burn Time, s	232	219
– Stage II Thrust, lb	104,000	99,000
Avionics		
– Body Rates & Acceleration Derivation	IMU Derived	LASS & Stage II Rate Gyro Derived
– Wind Load Alleviation	Day of Launch Uplink	N/A
Software		
– Memory Margin, %	11	8
– Timing Margin, %	65	65

SOLID ROCKET MOTORS

Modification	Now	Was MOL	Was T34D
Solid Segments (No.)	7	7	5½
– Propellant Type	UTP-3001B	UTP-3001B	UTP-3001
– Propellant (lb x 10^3)	592.0	589.1	464.3
– Thrust (lb x 10^6)	1.475	1.442	1.273
– Total Impulse (lb-s x 10^6)	159.7	158.6	123.1
Forward Closure	+ 40 in.	+ 40 in.	Ref
Aft Skirt Assy	Lightweight Design	MOL Design	T34D Design
Thrust Termination	None	Incorporated	None
Internal Insulation	Increased	Increased	Ref

Figure C8-3 Derivation of TIV Boost Vehicle and TIV Centaur



DOD STS MISSION

Payload (SV) Capability

DOD STS Mission		DOD SLV-X Mission
15 x 40 ft	Dynamic Envelope	15 x 40 ft
9413 lb	Weight to GEO	10,565 lb

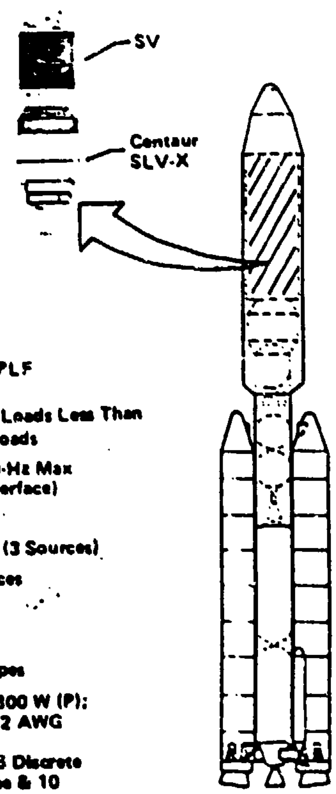
Payload (SV) Environments

DOD STS Mission		DOD SLV-X Mission
138 dB	Acoustics (OASPL)	136.1 dB
40°F to 120°F	Prelaunch Temp	50°F to 80°F
150°F Max (at P/L Bay Door Open)	In-Flight Temp	98°F Max (At PLF Separation)
SV Design Driven by STS Landing Loads	Loads	SLV-X Lateral Loads Less Than STS Landing Loads
2000 g at 5000-Hz Max (at Centaur Interface)	Shock	2000 g at 5000-Hz Max (at Centaur Interface)
0.5 psi/s Max	Pressure Decay	0.4 psi/s Max
18 V/m Max (5 Sources)	RF Emissions	11.3 V/m Max (3 Sources)
Multiple Sources Extended Time in STS	Contamination	– Fewer Sources – Short Time

Payload (SV) Interfaces

DOD STS Mission		DOD SLV-X Mission
3 Mech I/F Types	Physical	3 Mech I/F Types
1200 W (A); 1800 W (P); 6 kWh; 7 TP; 12 AWG Conductors	Dc Power	1200 W (A); 1800 W (P); 6 kWh; 7 TP; 12 AWG Conductors
Two 1.2 kbps Tlm; 16 Discrete Cmds; 10 Prime & 10 Backup Ord	Communications	1 kbps Tlm; 16 Discrete Cmds; 10 Prime & 10 Backup Ord

PAYLOAD ACCOMMODATIONS



DOD SLV-X MISSION

Figure C8-4 TIV Payload Accommodations

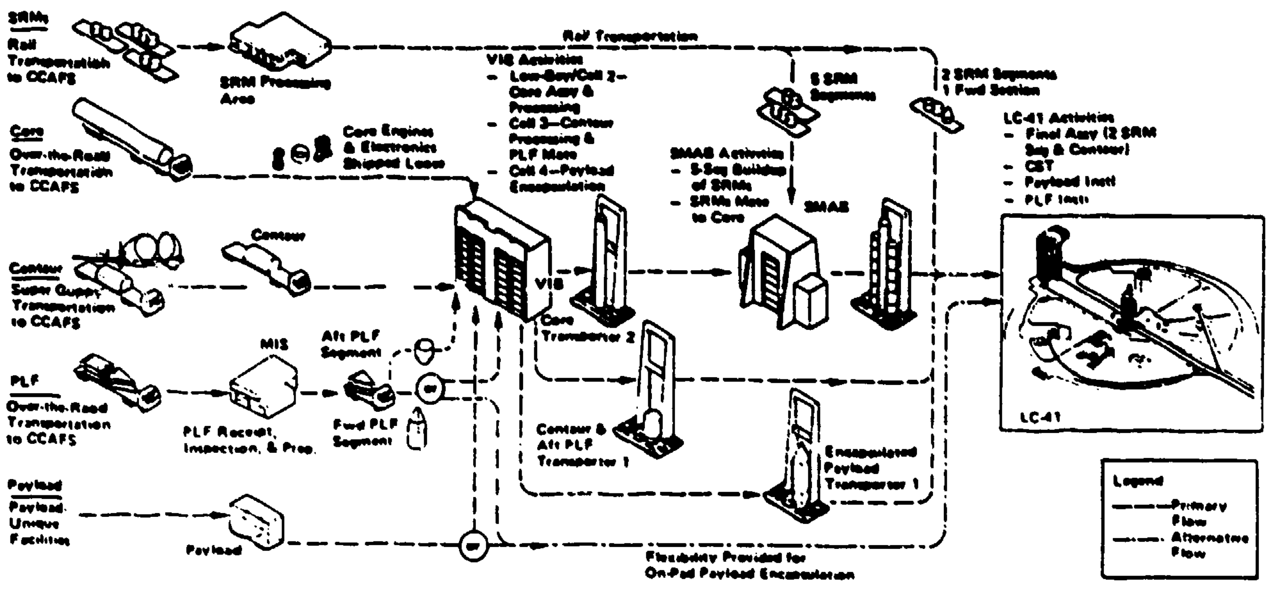


Figure C8-5 Launch Base Operations

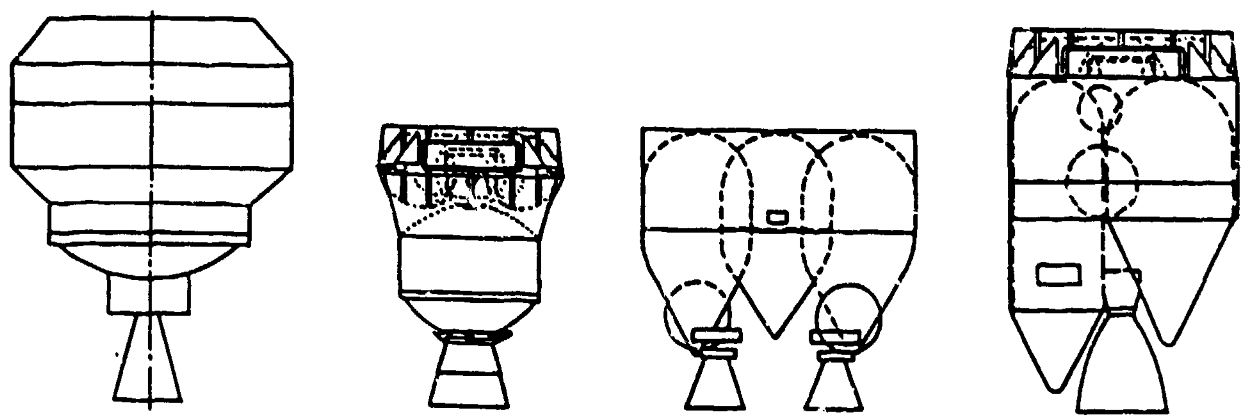


Figure C8-6 Several Alternatives to Centaur G Prime

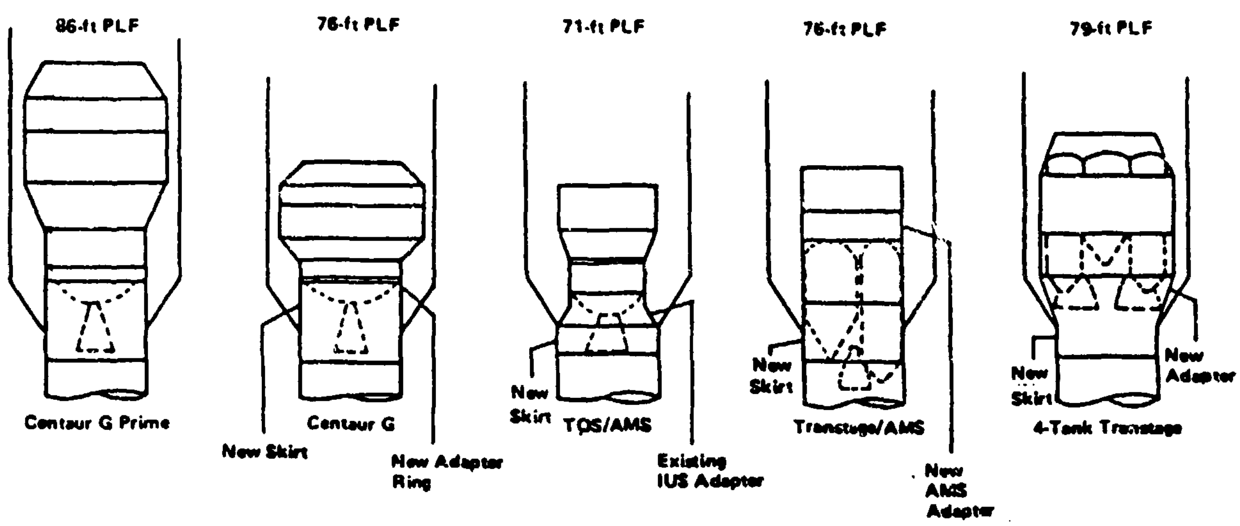


Figure C8-7 Structural Modifications for Alternative Upper Stages

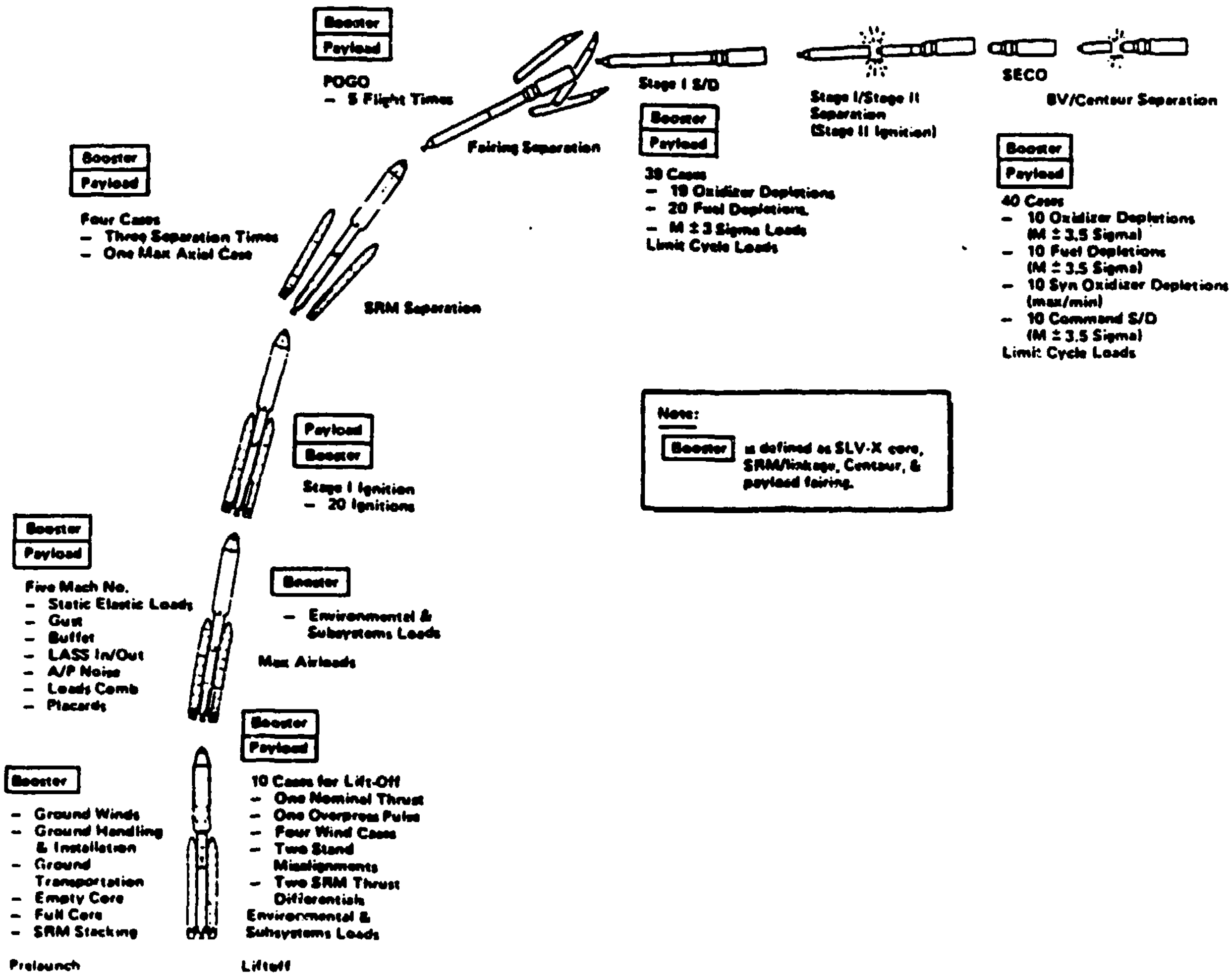
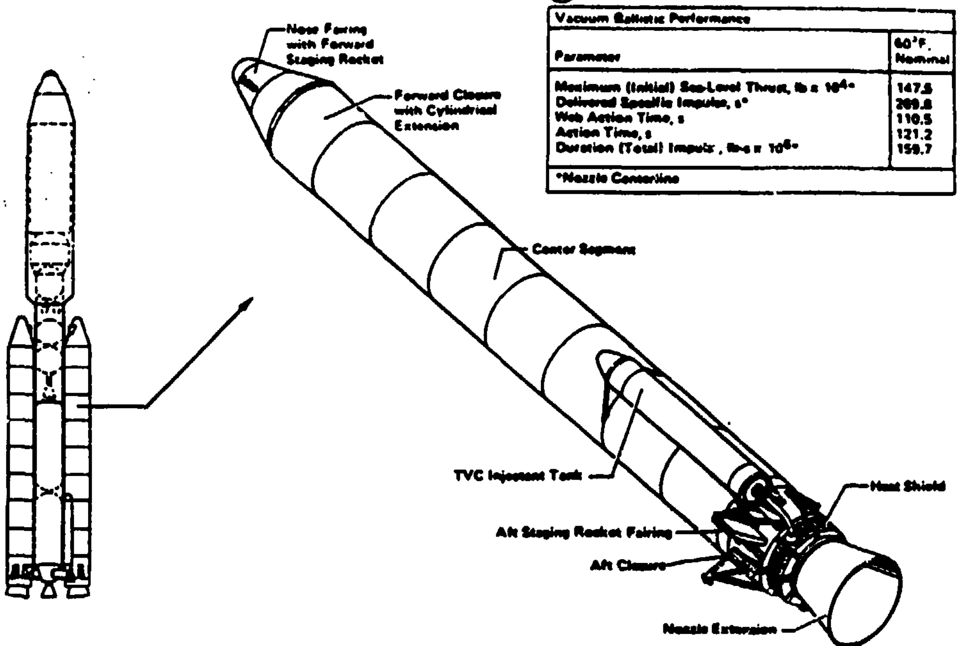


Figure C8-8 TIV Critical Flight Events

(B) SRM PERFORMANCE

Vacuum Ballistic Performance	
Parameter	60°F, Nominal
Maximum (Initial) Sea-Level Thrust, lb x 10^4*	147.5
Delivered Specific Impulse, s*	269.8
Web Action Time, s	110.5
Action Time, s	121.2
Duration (Total) Impulse, lb-s x 10^6*	159.7
*Nozzle Centerline	



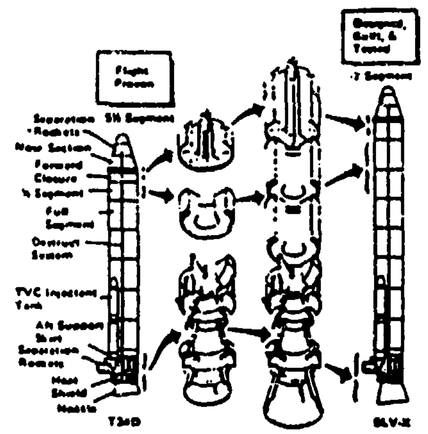
(A) REQUIREMENTS MATRIX

SS-ELV-401 Para 3.7.1.1.2	
Requirement	Compliance Summary
Segmented SRMs	7-Segment, SRMs with Forward & Aft Closures
Thrust Vector Control	N_2 Blowdown Liquid Inject System with Electromechanical Valves
Staging System	Four Forward & Four Aft Staging Rockets
Electrical/ Instrumentation	Battery-Powered Electrical Distribution & Flight Instrumentation System
Ordnance	Provides SRM Ignition, Separation, & Destruct Capability
Mechanical Attach Hardware	Aft Skirt Supports Core & SRM, Shear Tie & Outriggers
Aerodynamic Fairings	Nose Fairings on SRM & TVC Tanks

(C) MAJOR COMPONENTS

Component/ Assembly	Description	Flight Quantity	Previous Use	Modifications Required
Stage 0 Major Components				
Nose Section	Encloses Forward Staging Motors, Instrumentation, Electrical & Ordnance Cabling, & Umbilical Connectors	Two per Vehicle	T-III C, D, E, & 34D	None
Forward Closure	Forward Extension & Igniter Assembly	Two per Vehicle	T-III C, D, E, & 34D	40-in. Longer (MOL Design)
Motor Segment	130-in. Long by 120-in. dia	14 per Vehicle	T-III C, D, E, & 34D	Additional Insulation (Aft 3 Segments)
Aft Closure	Aft Closure & Throat Entrance	Two per Vehicle	T-III C, D, E, & 34D	Additional Aft Insulation
Nozzle Assembly	Throat, Exit Cone, & Extension	Two per Vehicle	MOL Tests	None
Aft Skirt	SRM & Core Support Hardware	Two per Vehicle	New	Modified 34D Design
Propellant	UTP-3001B	1.184 x 10^6 lb per Vehicle	MOL Tests	None
TVC System	Tank, Manifold, & Electromotive Valves	Two per Vehicle	T-III C, D, E, & 34D	None
Electrical System	Instrumentation, Control, & Ordnance	Two per Vehicle	MOL Tests	None

Figure C8-9 Solid Rocket Motor - Stage 0



(A) DESIGN SUMMARY

Design Modification	Description	Design Impact	Performance Impact	Design Limitations & Constraints	Risk
7-Segment SRM	Replace 5-1/2 Segment SRM with 7-Segment SRM	Increased Length & Weight	30% Increase in Total Impulse 13,300-lb Dry Weight Increase per SRM		Low Risk—MOL Configuration & 4 Static Firings Completed
		Longer Forward Closure*	Include Above	None—MOL Design & Test	Low—Data Avail. from Static Tests
		Replace 1/2 Segment with 2 Full Segments*		None—MOL Design & Test	Low—Data Avail. from Static Tests
		UTP-3001B Propellant*		None—MOL Design & Test	Low—Data Avail. from Static Tests
		Larger Throat & Nozzle Assembly*		None—MOL Design & Test	Low—Data Avail. from Static Tests
		Longer Raceway Cables & Ordnance*		None—MOL Design & Test	Low—Minor Change
		Additional Internal Insulation* Cables & Ordnance*		Test Data Available, Changes Based on Analyses & Schedule Test	Low—Will Be Verified by Static Test Firings
Aft Skirt Assembly	Modified Design Based on T34D Skirt Design	Will Use New Materials & Revised Mfg Processes	-2000-lb Dry Weight per SRM	Requires New Design & Structural Analyses	Low—Design Will Be Verified by Structural Test

*Design impacts are secondary effects caused by the primary design modification.

(B) SRM MAJOR COMPONENTS/ASSEMBLIES

5, 5-1/2, & 7 Segment Commonality

- Segment Cases
- Aft Closure Cases
- Nose Cone Assembly
- TVC Injectant Tank
- Injectant Tank Fairing
- Injectant Tank Extension
- Injectant Transfer Tube
- Transfer Tube Support
- Injectant Manifold
- Injectant Valves
- Injectant Valve Housings
- All Fluid Connections
- Igniter
- Destruct Ordnance Material
- Staging Rockets
- Staging Rocket Fairings
- Electrical System Raceways
- All Batteries
- Electrical Distribution Boxes
- Power Transfer Switches
- Power Umbilicals
- Heat Shield
- Propellant Liner
- Insulation Materials
- Nozzle Materials

5-1/2 & 7 Segment Differences

- New Aft Skirt Assembly
- Additional Internal Insulation
- Longer Raceway Cables & Destruct Ordnance
- 1/2 Segment Replaced by 2 Full Segments
- Forward Closure 40 in. Longer
- Nozzle Expansion (9.16:1 vs 8:1)
- Propellant (UTP-3001B vs UTP-3001)

Note:
Shading indicates existing 7-segment configuration

Figure C8-10 TIV SRM Design

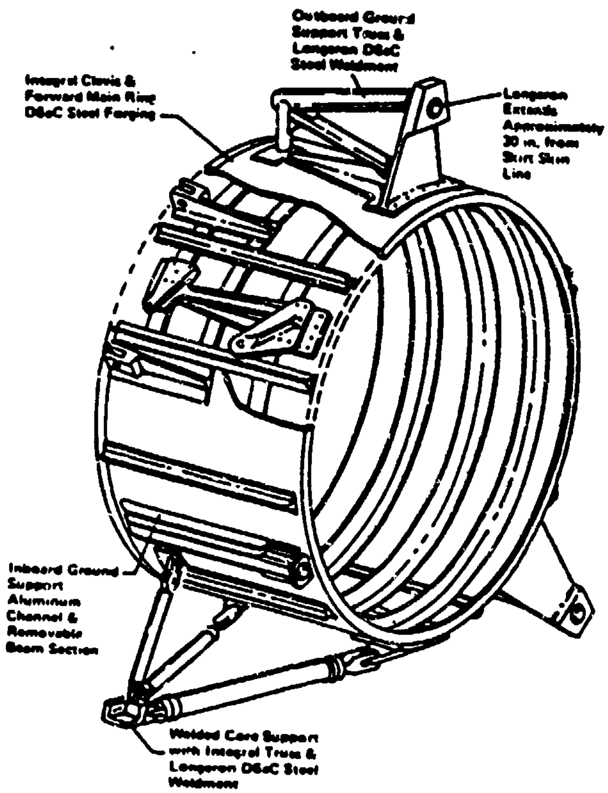
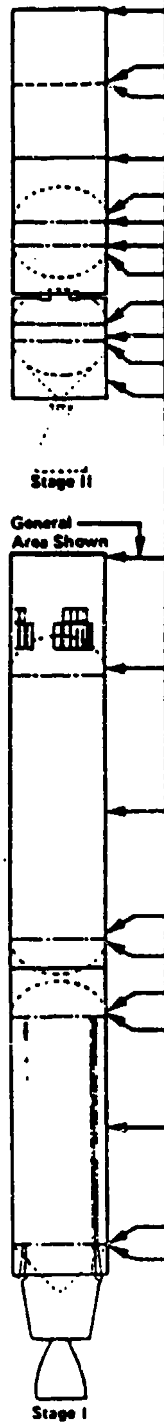


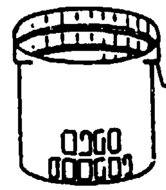
Figure C8-11 TIV SRM Aft Skirt

(A) CORE VEHICLE STRUCTURE SIZING

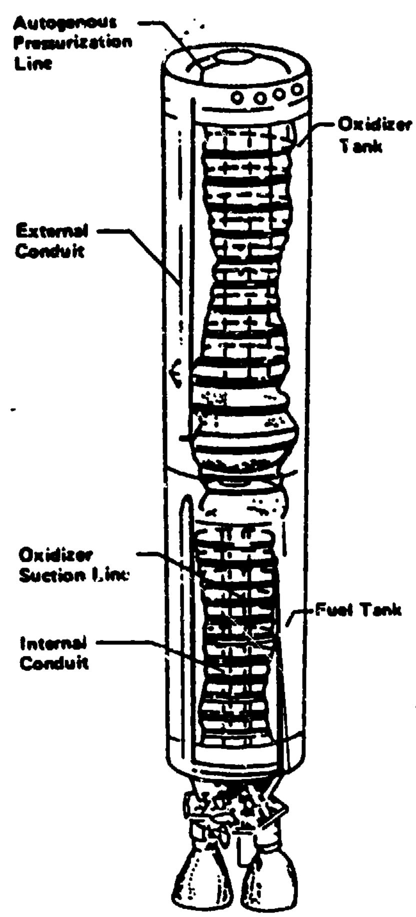


$P_{EQ} \times 10^6$ lb T34D	$P_{EQ} \times 10^6$ lb SLV-X	Design Resizing from T34D, %
1.08	0.81	-25
1.08	0.81	-25
1.08	1.71	+58
1.08	1.77	+64
1.17	1.88	+62
1.17	1.88	+62
1.20	1.92	+62
1.30	2.06	+59
1.37	2.12	+56
1.38	2.10	+54
1.44	2.18	+53
1.50	2.16	+45

$P_{EQ} \times 10^6$ lb T34D	$P_{EQ} \times 10^6$ lb SLV-X	Design Resizing from T34D, %
1.50	2.18	+45
1.37	2.04	+49
1.25	1.94	+55
1.13	1.70	+58
1.66	2.29	+38
1.61	2.09	+30
1.58	2.09	+32
1.86	1.77	0
1.94	1.46	0
2.24	1.77	0



INTERSTAGE STRUCTURE



STAGE I

TITAN 34D CORE VEHICLE

Figure C8-12 Titan 34D Core Vehicle (Sheet 1)

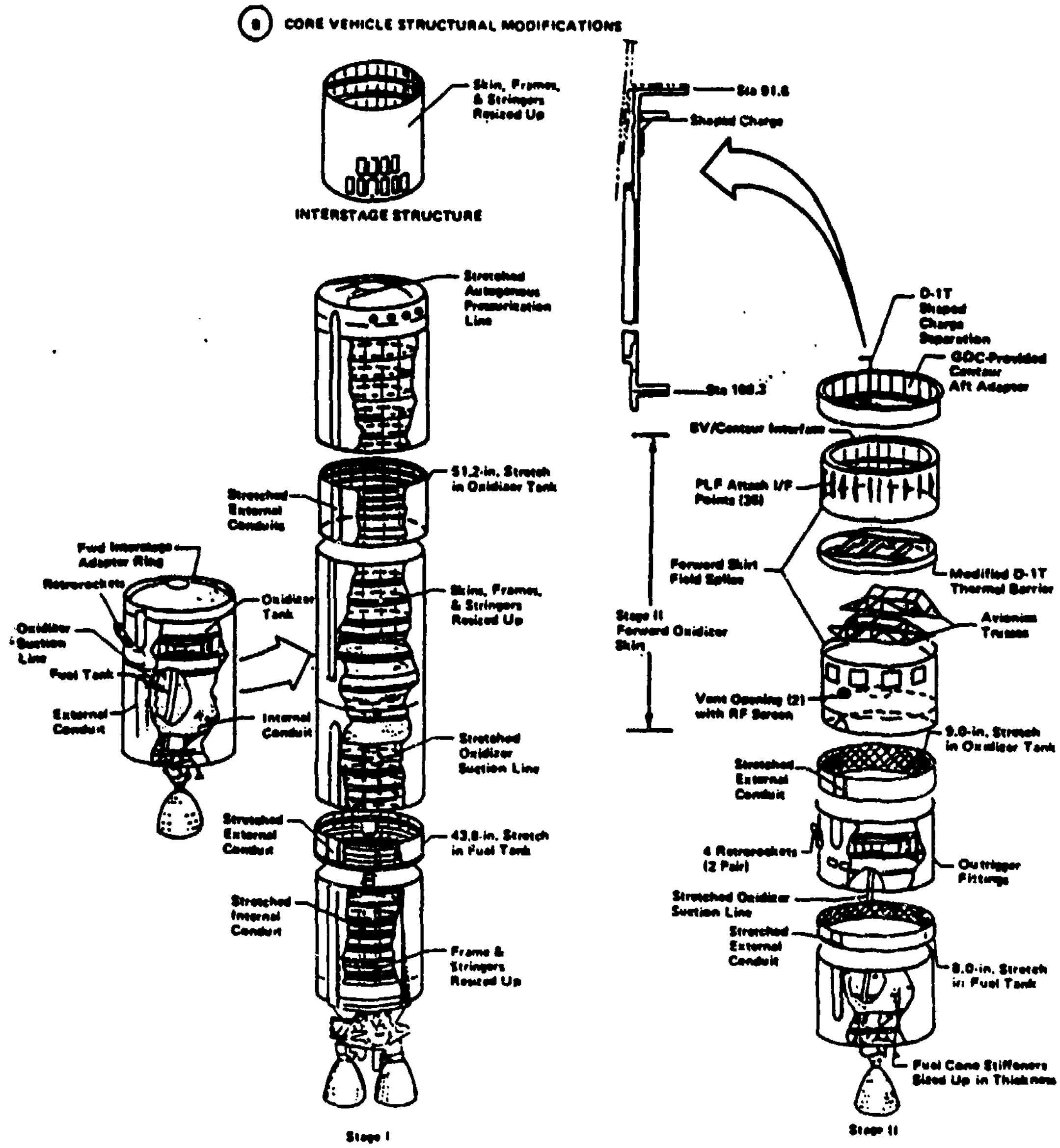
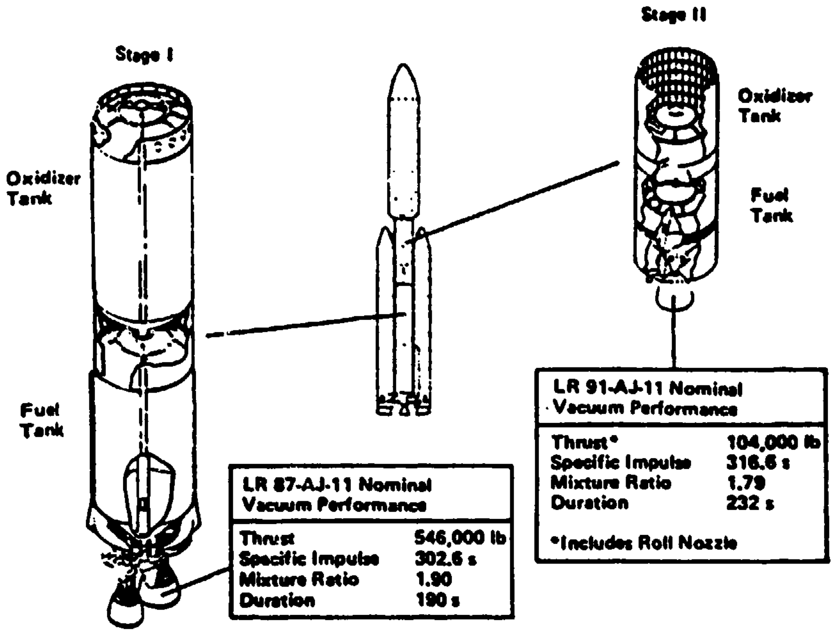


Figure C8-12 Titan 34D Core Vehicle (Sheet 2)



LR 87-AJ-11 Nominal Vacuum Performance	
Thrust	546,000 lb
Specific Impulse	302.6 s
Mixture Ratio	1.90
Duration	190 s

LR 91-AJ-11 Nominal Vacuum Performance	
Thrust*	104,000 lb
Specific Impulse	316.6 s
Mixture Ratio	1.79
Duration	232 s

*Includes Roll Nozzle

(A) REQUIREMENTS MATRIX

SS-ELV-401 3.1.1.1.1.2 Rqmt	Compliance Summary
Core LREs—Pump Fed	Stage I—LR 87-AJ-11 Stage II—LR 91-AJ-11
Storable Propellants	N_2O_4/A-50
Thrust Vector Control	Stage I—2-Gimbaled TCAs Stage II—1-Gimbaled TCA & Roll Control Nozzle
Pressurization	Autogenous—Fuel & Oxidizer
Engine/Core Interfaces	Engine, Frame, Propellant Feed Lines, Pressurization, Electrical, Instrumentation

(B) MAJOR COMPONENTS

Component/Assembly	Description	Flight Quantity	Previous Use	Modifications Required
Stage I				
— Engine Assembly	LR 87-AJ-11	1 Each	T-III B, C, D, E, & T34D	None
— Propellant Tanks	Fuel Tank 2132 ft^3 Oxidizer Tank 2543 ft^3			44 in. Longer 51 in. Longer
— Feed System	Prevalves, Accumulators, & Oxidizer Feedline	4 Prevalves 4 Accumulators 1 Oxid Feedline		Oxidizer Feedline 44 in. Longer
— Pressurization System	Autongenous—Fuel & Oxidizer	1 Each		Fuel Line + 44 in.; Oxid Line + 9[illegible] in.
— Propellant	Fuel—A50 (UDMH/N_2H_4) Oxidizer—N_2O_4	118,000 lb 223,000 lb		None
Stage II				
— Engine Assembly	LR 91-AJ-11	1 Each		5% Higher Thrust Balance
— Propellant Tanks	Fuel Tank 506 ft^3 Oxidizer Tank 564 ft^3			8 in. Longer 9 in. Longer
— Feed System	Prevalves & Oxidizer Feedline	2 Prevalves 1 Oxid Feedline		Oxidizer Feedline +17 in.
— Pressurization System	Autogenous—Fuel & Oxidizer	1 Each		Fuel Line + 8 in.; Oxid Line + 17 in.
— Propellant	Fuel A50 (UDMH/N_2H_4) Oxidizer N_2O_4	27,700 lb 49,300 lb		None

Figure C8-13 TIV Stage I and II Propulsion

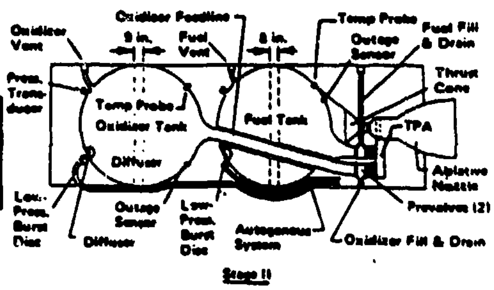
(A) DESIGN SUMMARY

Design Modification	Description	Design Impact	Performance Impact	Design Limitations/ Constraints	Risk
Stage I Longer Tanks	+43.8-in. Fuel Tank +51.2-in. Oxid Tank +95-in. Total	Increased Stage Length & Engine Burn Time	+48000 lb Propellant +25 s Burn Time	None	Low—Have Stretched before
		44 in. Longer Oxidizer Feedline & Fuel Pressurization Line*	Minor Weight Increase & Small Increase in System ΔP	Pressure Drop Analysis Required for Additional Length	Low—Have Stretched Before
		95-in. Longer Oxidizer Pressurization Line*			Low—Have Stretched Before
Stage II Longer Tanks	+8-in. Fuel Tank +9-in. Oxid Tank 17-in. Total	Increased Stage Length & Engine Burn Time	+6300 lb Propellant +13 s Burn Time	None	Low—Have Stretched Before
		8-in. Longer Oxidizer Feedline & Fuel Pressurization Line*	Same As Stage I	Pressure Drop Analysis Required for Additional Length	Low—Have Stretched Before
		17-in. Longer Oxidizer Pressurization Line*			Low—Have Stretched Before
		5% Increase in Engine Ablative Nozzle Liner*	+25 lb Nozzle Dry Weight	Stress Analysis Required on Chamber for Nozzle Mass Addition	Low—Subscale Tests Will Verify Ablative Changes
Increased Thrust Level (Stage II)	5% Higher Thrust Balance	Increased Thrust & Shorter Burn Time	+5% Thrust	Increase Will Be Demonstrated during Engine Acceptance Test	Low—Same Changes Have Been Done on Stage I Engine
		Strengthen Tank Thrust Cone Structure*	+10 lb Dry Wt	None	Low—Minor Change
		Beef Up Pump Discharge Line*	+10 lb Engine Dry Weight	None	Low—Minor Change

* Design impacts are secondary effects caused by the primary design modification.

(B) TANK STRETCH HISTORY

	Program	Fuel Tank	Oxidizer Tank
Stage I	T-III ⟶ 34D T34D ⟶ SLV-X	+31.2 in. (10%) +44 in. (13%)	+36.7 in. (11%) +51 in. (14%)
Stage II	TII ⟶ III/34D 34D ⟶ SLV-X	+7.5 in. (7%) +8 in. (7%)	– +9 in. ([illegible])



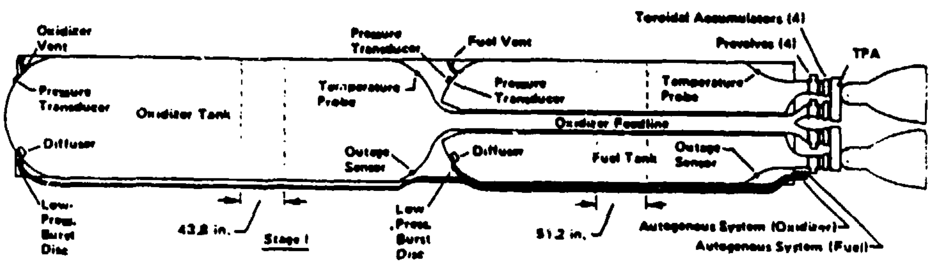
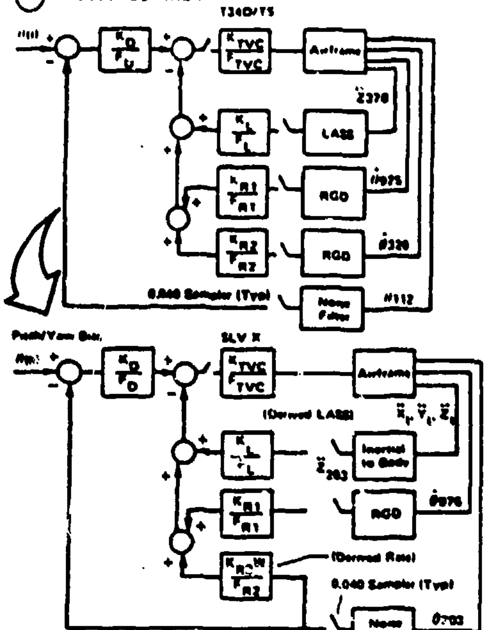


Figure C8-14 TIV Core Propulsion System

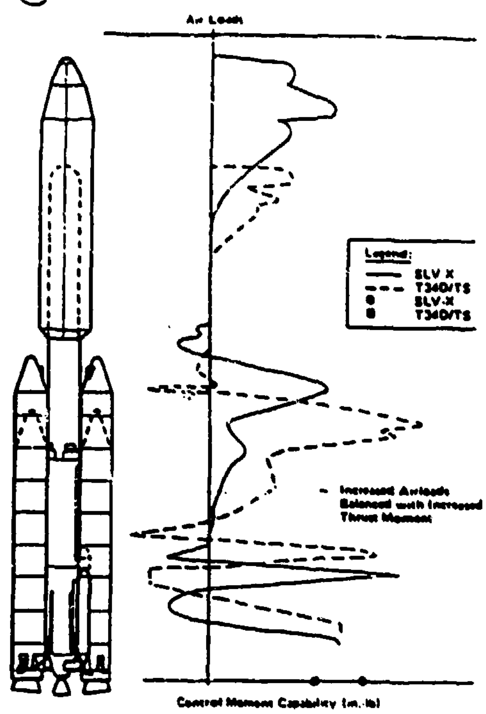
(A) FCS CONFIGURATION



Our flight control system uses the existing T34D/TS configuration with the following modifications:

- Add pitch/yaw bias steering to enhance load relief capability.
- Derive LASS & Rate functions in lieu of hardware components.
- Delete Transtage powered & coast flight software functions.

(B) DISTRIBUTED AIR LOADS



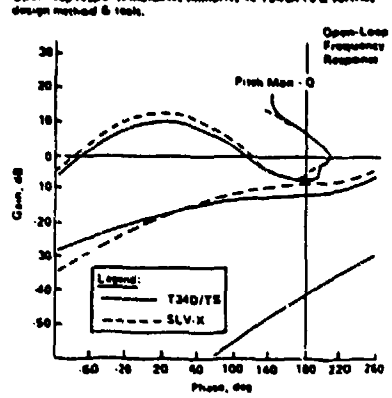
(D) PARAMETER COMPARISON AT MAX Q

Parameter	Units	T34D/TS	SLV-X
– Max Dyn Press.	psf	1000	900
– Thrust	lb	2.17E6	2.69E6
– Thrust Moment Arm	in.	510.2	631.9
– Inertia (Pitch)	in.-lb-s²	2.36E8	4.62E8
– Pitch Moment Coefficient	1/rad	57.11	112.9
Pitch Aero Moment / Inertia	1/s²	2.28	2.07

Comparison of critical pitch channel control system design parameters shows great similarity to T34D/TS.

(E) OPEN-LOOP FREQUENCY RESPONSE

Open-loop response indicates similarity to T34D/TS & verifies design method & tools.



(F) ANALYSIS TOOLS

All current analysis tools are operational & applicable:

- LADCAP–Signal Flow, Transfer Function, Z Transform, & Frequency Response
- QD219–Nonlinear Time Domain Simulation for Limit Cycle Analysis
- DAL–Distributed Airloads Analysis
- QD245–Digital Filter Analysis
- QD259–Flight Parameter Generation

Figure C8-15 Flight Control System with Load Relief

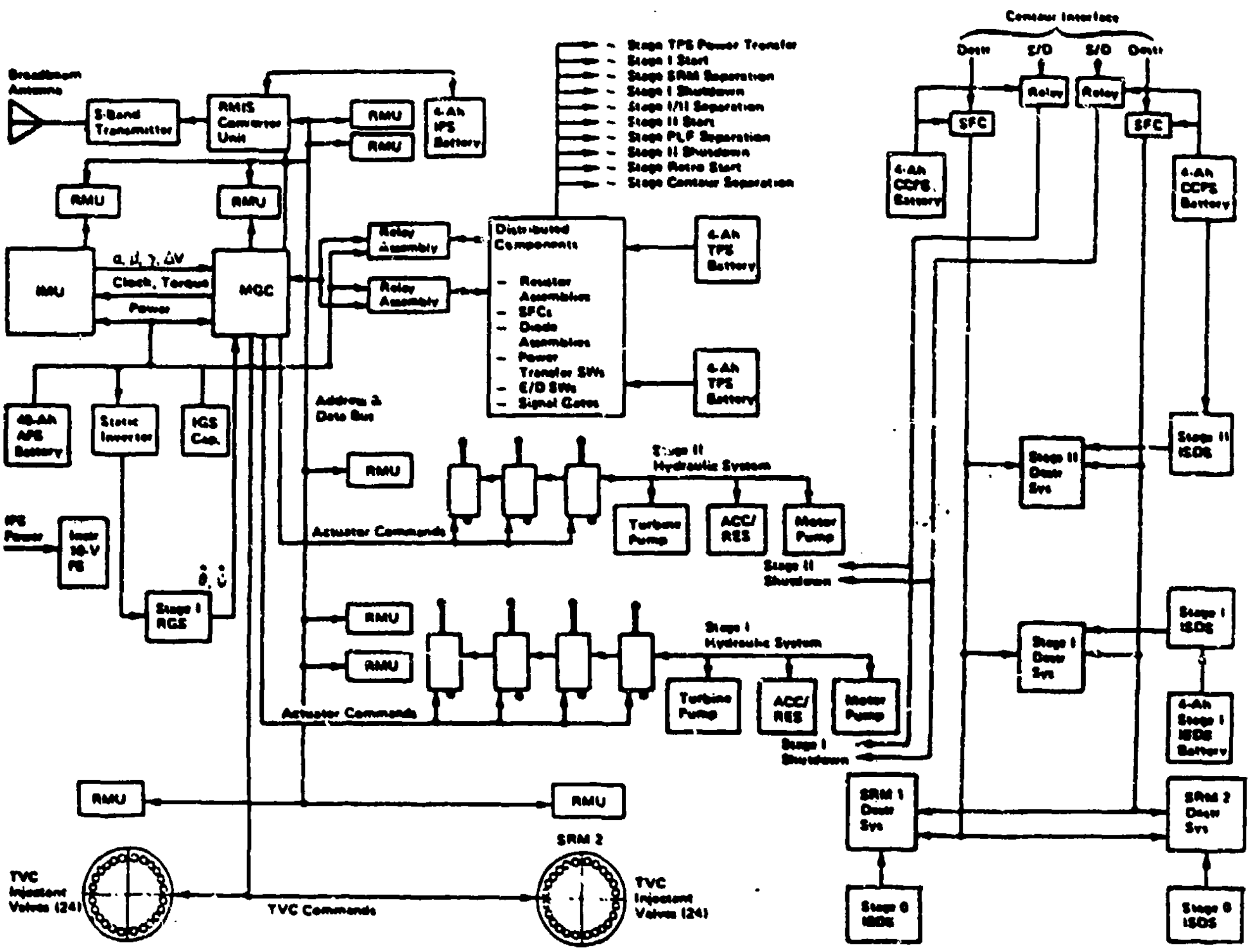


Figure C8-16 Avionics System Design

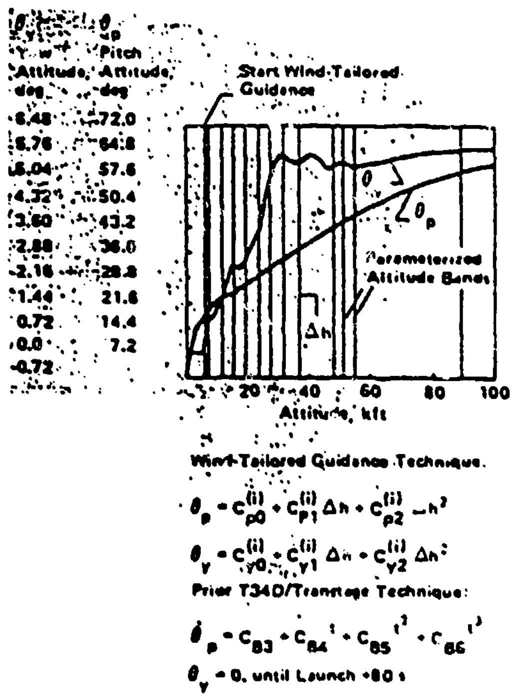


Figure C8-17 Wind-Tailored Guidance Techniques

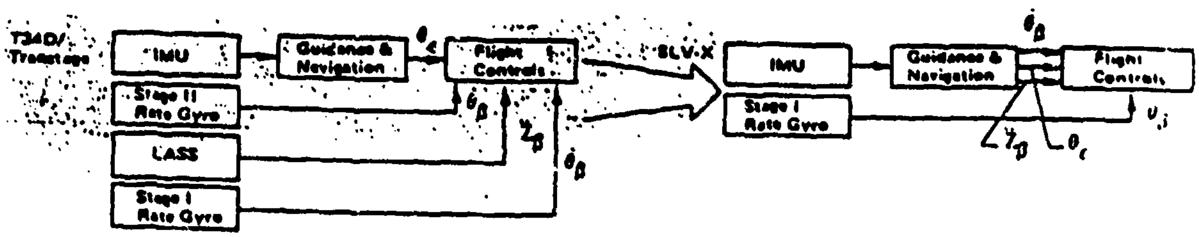


Figure C8-18 Software accommodates removal of excess hardware.

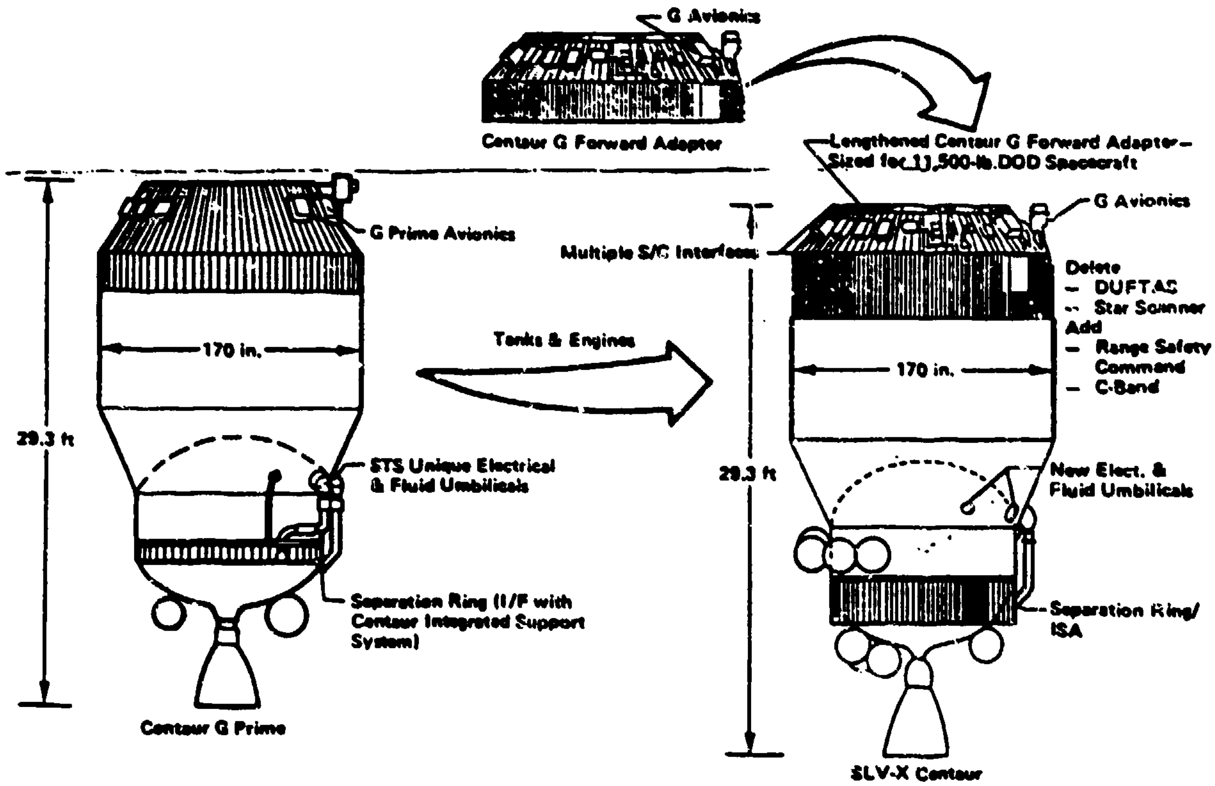


Figure C8-19 Derivation of TIV Centaur

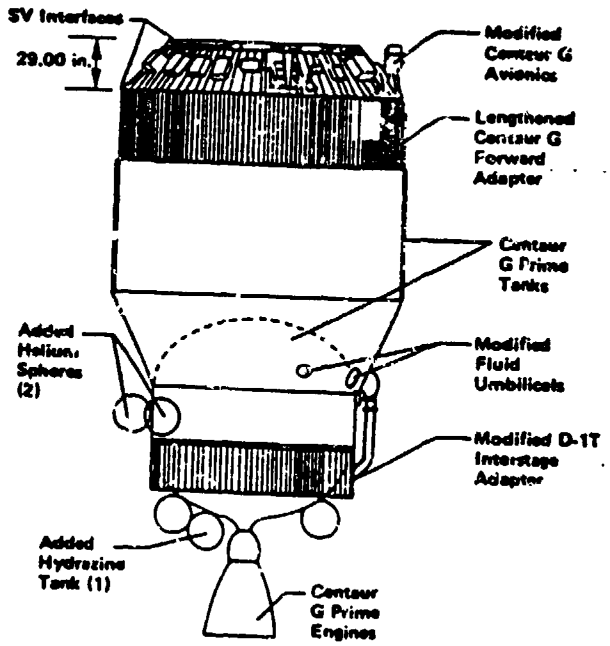
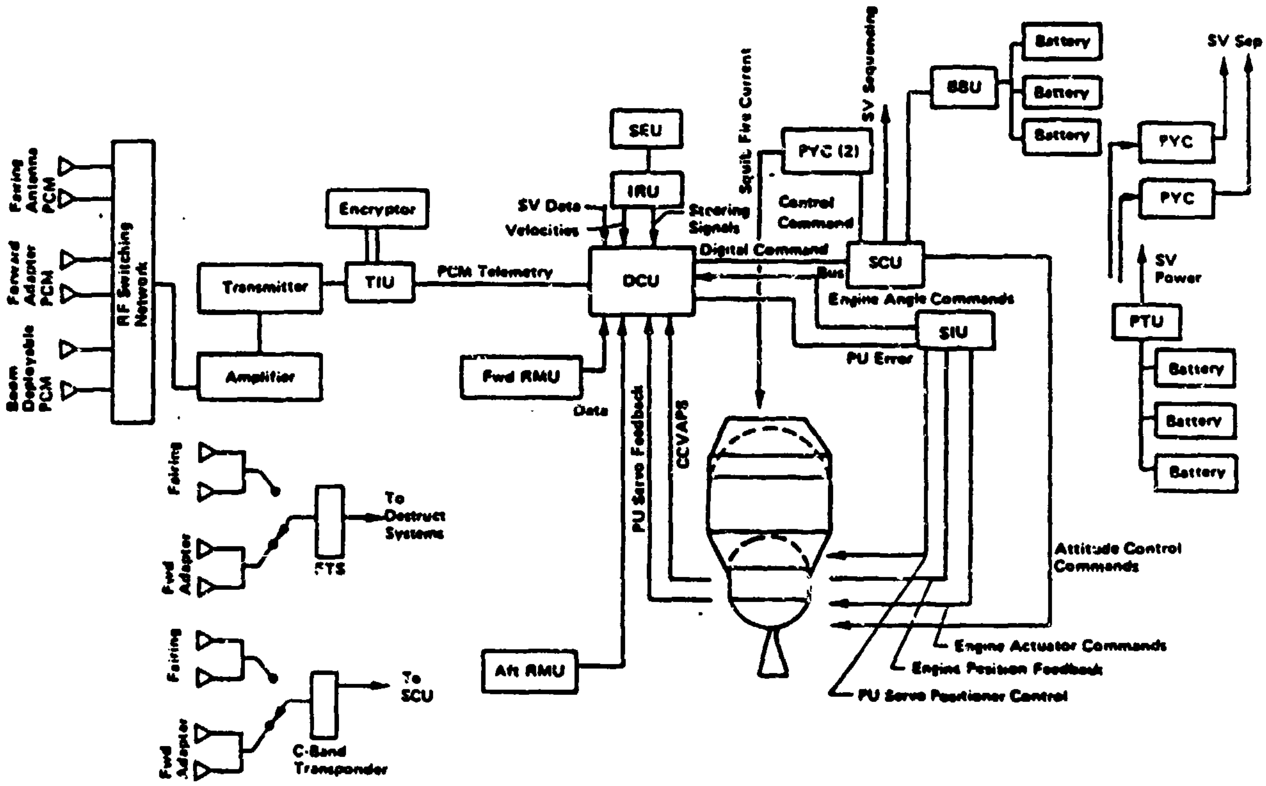


Figure C8-20 TIV Centaur



Acronyms

BBU	Battery Bussing Unit	PU	Propellant Utilization
CCVAPS	Computer Controlled Vent & Pressurization System	PYC	Pyrotechnic Control Unit
		RMU	Remote Multiplex Units
DCU	Digital Computer Unit	SCU	Sequence Control Unit
FTS	Flight Termination System	SEU	System Electronics Unit
IRU	Inertial Reference Unit	SIU	Servo Inverter Unit
PTU	Power Transfer Unit	SV	Satellite Vehicle
		TIU	Telemetry Interface Unit

Figure C8-21 The TIV Centaur Avionics is derived from STS Centaur G Avionics.

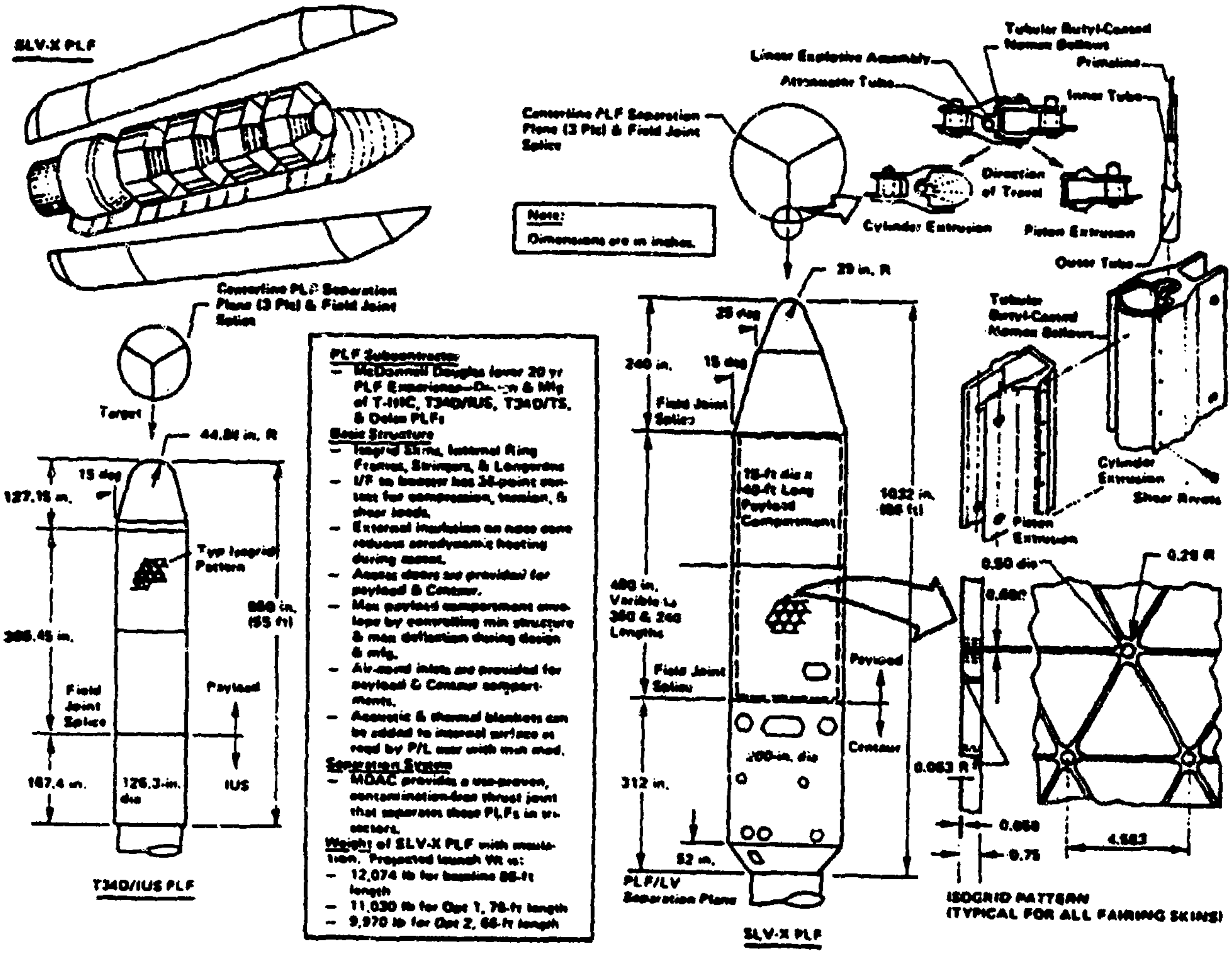
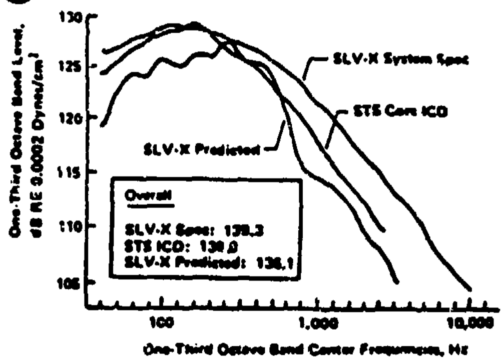


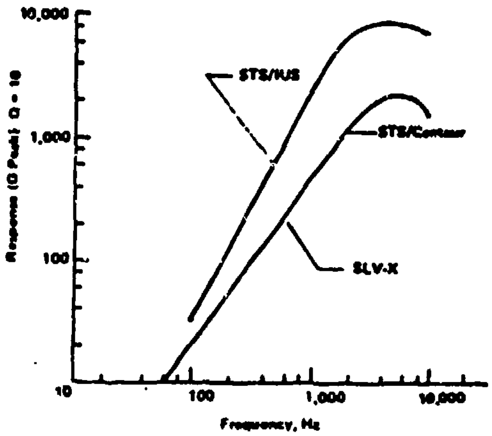
Figure C8-22 TIV Payload Fairing (PLF)

(A) ACOUSTICS

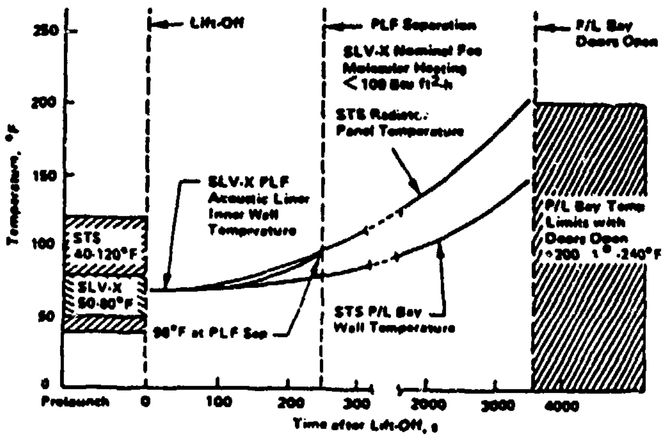


(B) SHOCK

Upper Stage Induced Shock at SV Interface



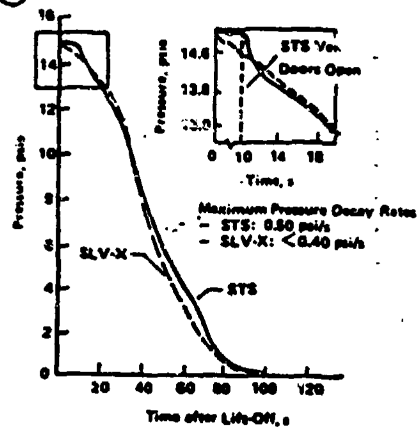
(C) THERMAL



(D) CONTAMINATION

Contamination Sources		
STS/IUS	STS/Contaur	SLV-X
– PCR	– PCR	– LC-41
– Orbiter Bay	– Orbiter Bay	– PLF
– Orbiter RCS	– Orbiter RCS	– Stage II Retro
– IUS Sources	– Contaur Sources	– Contaur Sources
– Payload Mix	– Payload Mix	

(E) PRESSURE DECAY



(F) EMC/EMI

RF Emissions					
STS/Contaur		STS/IUS		SLV-X	
Source	Level, V/m	Source	Level, V/m	Source	Level, V/m
S-band FM	1.2	S-band FM	1.2	Titan S-band	Negligible
S-band PM	18.0	S-band PM	18.0	Contaur S-band	5.0
S-band (SGLS)	2.0	S-band (SGLS)	2.0	Contaur C-band	11.3*
Ku Band	9.0	Ku Band	9.0		
Contaur S-band	5.0	IUS S-band	137.0		

*Value shown is peak; 0.17 V/m is average level.

(G) SLV-X PAYLOAD LOAD FACTORS

Event	Axis	Steady-State Value, Gs*	Variation Value, Gs	Max/Min Value, Gs
Lift-Off	– Axial	1.5C	± 1.5	3.0C/OT
	– Lateral	0	± 2.5	± 2.5
Max A/L	– Axial	2.0C	± 1.0	3.0C/1.0C
	– Lateral	0	± 2.0	± 2.0
Stg I B/O	– Axial	0 to 3.6C	± 2.0	2.0T5.6C
	– Lateral	0	± 1.0	± 1.0
Stg II B/O	– Axial	0 to 2.9C	± 2.0	2.0T/4.0C
	– Lateral	0	± 1.0	± 1.0

*C = Compression, T = Tension

Figure C8-23 TIV Payload Environments

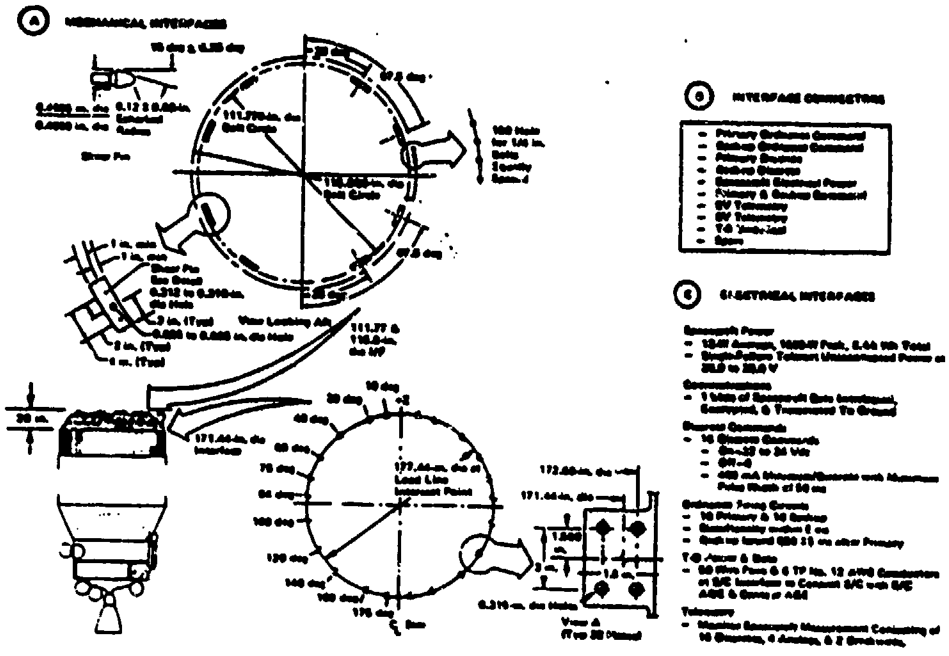


Figure C8-24 TIV Interfaces

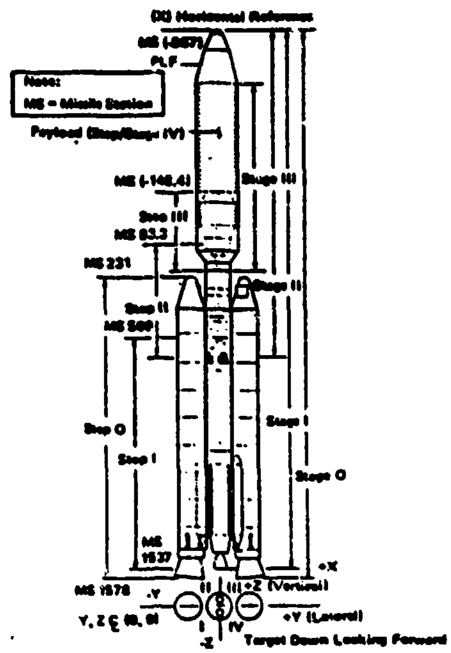
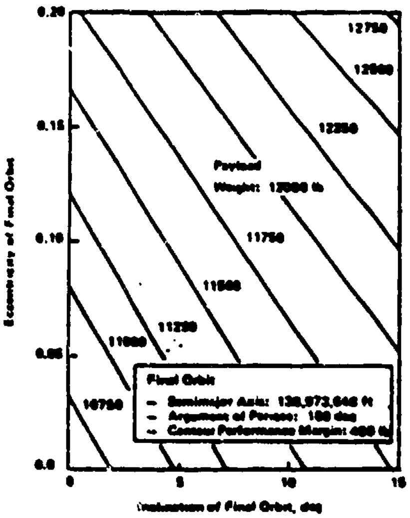


Figure C8-25 TIV Mass Properties Reference Diagram

(A) PAYLOAD WEIGHT

Performance Summary	GSO Spec	12-h Inclined Spec	12-h Inclined Alt (1)	12-h Inclined Alt (2)	24-h Inclined Spec	24-h Inclined Alt
Payload Wt Capability (Includes Performance Margin), lb	10,565	17,819	16,982	12,626	10,918	10,768
Concept-to-Design Contingency, lb*	-310	-347	-347	-347	-318	-318
Minimum Payload Wt Capability (Includes Performance & Contingency Margins), lb	10,255	17,472	16,635	12,279	10,600	10,450
Minimum Reqd Payload Wt, lb	10,000	11,500	None	None	None	None
Mission Definition						
– Launch Azimuth, deg	93	37.9	49	49	37.8	49
– Park Orbit Inclination, deg	28.5	56	46.5	46.5	56	46.5
– Final Orbit Inclination, deg	0.0	63.4	63.4	63.4	65	65
– Argument of Perigee, deg	–	–	–	270	–	–

(B) SLV-X PAYLOAD CAPABILITY FOR MISSION RANGE



(C) BOOSTER VEHICLE FLIGHT PERFORMANCE RESERVE

Booster Vehicle Parameter	Dispersed Value	Contour Propellant Margin, lb GSO	12-h Inclined	24-h Inclined
Stage 0 I_{sp}	-0.7%	78	90	85
Stage 0 Web Action Time	+2.99%	170	202	184
Stage I Thrust	-3.27%	163	200	177
Stage I I_{sp}	-0.76%	106	120	114
Stage I Outage	+1672 lb	139	160	151
Stage I Propellant	-5135 lb	36	31	39
Stage II Thrust	-3.8%	61	82	66
Stage II I_{sp}	-1.11%	105	116	114

(C) (cont)

Booster Vehicle Parameter	Dispersed Value	Contour Propellant Margin, lb GSO	12-h Inclined	24-h Inclined
Stage II Outage	+516 lb	100	110	100
Axial Force Coefficient	+10%	93	118	101
Payload Fairing Weight	+635 lb	46	52	49
Unsimulated Dispersions		60	93	87
Booster Vehicle 3-Sigma Flight Performance Reserve (rss)		360	436	390

(D) SLV-X CONTINGENCY

SLV-X Contingency Items	Contingency Wt, lb	Equivalent Contour Propellant Margin, lb GSO	12-h Inclined	24-h Inclined
Booster Vehicle Growth				
– Solid Rocket Motor (SRM)	3380	71	80	73
– Stage I	151	21	24	22
– Stage II	223	47	53	48
– Payload Fairing	956	133	151	137
Total		272	308	280
Booster Vehicle Analysis				
– SRM Perf Predictions		37	42	38
– Liquid Rocket Motor Perf Predictions		73	83	75
Total		110	125	113
Contour Growth Analysis – Total Contingency		100	100	100
SLV-X – Total Concept-to-Design Contingency, Root-Sum-Square of Booster Vehicle Growth, Analysis, & Contour		310	347	318

*For conservatism, the partial δ payload/δ propellant margin = 1 was used.

(E) PROPELLANT MARGIN

SLV-X Items	Stage II Propellant Reqmt, lb	Equivalent Contour Propellant Margin, lb GSO	12-h Inclined	24-h Inclined
Booster Vehicle				
– Target-to-Launch	120	46	52	47
– 3D-to-6D Simulation	35	13	15	14
– Close-to-TAG	38	11	13	12
– 3-Sigma Flight Performance Reserve		360	436	390
– Booster Vehicle Reqmt		437	510	472
Contour – 3-Sigma Flight Performance Reserve		220	220	220
SLV-X Performance – Total Performance Margin, (Root-Sum-Square of Booster Vehicle & Contour)		489	563	521
SLV-X Contingency – Total Concept-to-Design Contingency Margin		310	347	318
SLV-X Total – Total Contour Propellant Margin (Sum of Performance & Contingency)		799	910	839

Figure C8-26 TIV performance capability exceeds requirements.

(A) GROUNDRULES FOR SLV-X GSO BASELINE PERFORMANCE TRAJECTORY

Vehicle Definition

- Baseline Vehicle Dry Weight and Propellant Loading Data
- Stage 0 SRM Temp = 71.5°F; Web Action Time at 71.5° = 108.8 s
- SRM Nozzle Exit Area = 12,483 in.^{2} per SRM
- Stage I Propellant Temp = 72.5°F
- Stage II Propellant Temp = 70°F
- Reduced Ullage with Optimum Fuel Bias and Propellant Depletion Shutdowns Apply to Both Stage I and Stage II
- Avg Stage I Nozzle Centerline Thrust = 543,498 lb
- Avg Stage I I_{sp} = 302.77 s
- Avg Stage II Vacuum Thrust = 106,654 lb (Includes Roll Nozzle; Uprate 5% over T34D)
- Avg Stage II Vacuum I_{sp} = 319.56 s
- Payload Fairing = 86 ft long by 200 in. dia; Jettison Wt = 12,074 lb
- Avg Centaur Vacuum Thrust = 33,000 lb
- Avg Centaur Vacuum I_{sp} = 446.3 s
- Aerodynamic Data - Preliminary SLV-X Configuration Data

Mission Profile

- Flight Azimuth = 93 deg
- PLF Separation = 50 s before Stage I/II Separation
- Stage 0 Separation Sequence - Stage 0 separation occurs 0.45 s after sensed longitudinal accel declines to 1.3 g during SRM thrust tailoff.
- Steering Techniques:

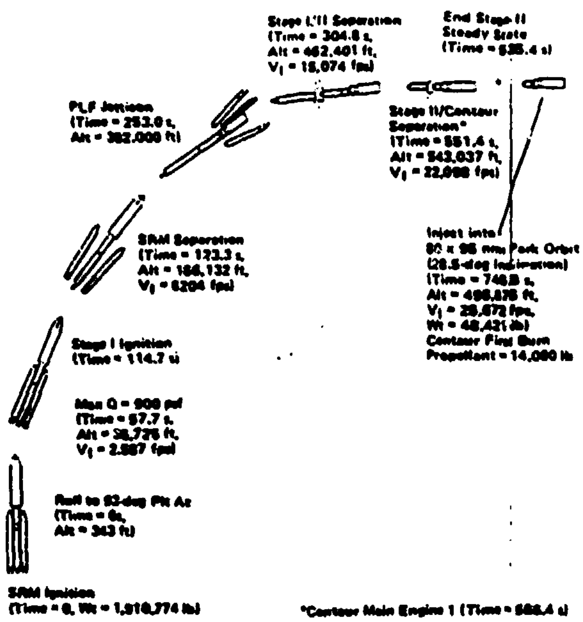
Time from SRM Ign, s	Steering
0 to 10	Vertical Rise, Roll to Flight Azimuth
10 to 20	Initiate Pitch Program
20 to 25	Reorient to Zero Lift Attitude
25 to 90	Zero Lift Attitude
90 to Park Orbit Inject	Optimized Pitch Rates for Maximum Performance
Centaur Orbit Burns	Optimized Pitch Rates to Maximize Final Orbit Weight

- Park Orbit - 80 by 95 nmi, 28.5-deg Inclination
- Transfer Orbit Burn at First Equatorial Crossing
- Final Orbit 19,323 nmi Circular, 0-deg Inclination

(B) BOOST TRAJECTORY DESIGN GUIDELINE CRITERIA

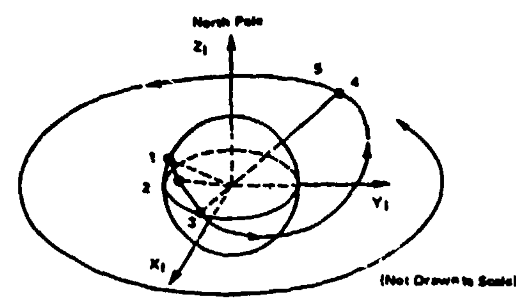
Parameter	Guideline Value	GSO Mission Simulation Data
Max Dynamic Pressure	≤900 psf	900 psf
Aerodynamic Heating Indicator (AHI)$_I$	≤100 x 10^6 ft-lb/ft^2	83.3 x 10^6 ft-lb/ft^2
Angle of Attack at Stage 0/I Separation	≤4.0 deg	2.3 deg
Dynamic Pressure at Stage 0/I Separation	≤60 psf	24.5 psf
Payload Fairing Separation	Will Not Occur within 10 s of Stage I/II Sep	50 s before Stage I/II Sep
Nominal FMH at PLF Separation	≤100 Btu/(ft^2-h)	100 Btu/(ft^2-h)

(C) GSO MISSION ASCENT TO PARK ORBIT



*Centaur Main Engine 1 (Time = 565.4 s)

(D) TYPICAL GSO MISSION PROFILE



Note:

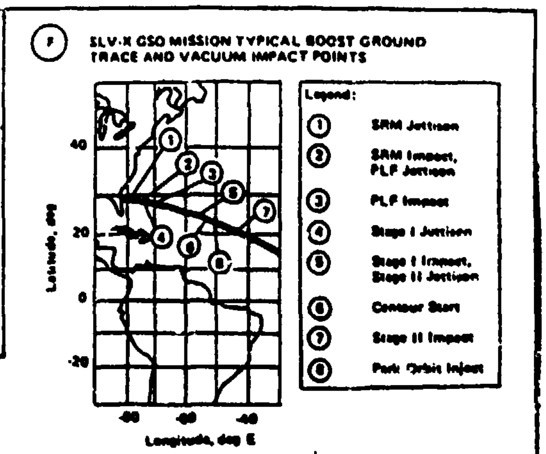
1. Launch from ESMC (Time = 0 s)
2. Park Orbit Injection (Time = 747 s, Alt = 82 nmi, Inclin = 28.6 deg)
3. First Equatorial Crossing, Initiate Centaur Second Burn (Time = 1393 s, Alt = 81.6 nmi, Inclin = 28.6 deg) Second Burn Duration = 288 s, Centaur Propellant Used = 20,707 lb
4. Initiate Centaur Third Burn, Final Orbit Injection (Time = 20,370 s, Alt = 19,323 nmi, Inclin = 28.6 deg) Third Burn Duration = 123.2 s, Centaur Propellant Used = 8,110 lb
5. Payload Separation (Time = 20,493 s, Alt = 19,323 nmi, Inclin = 0 deg, Eccentricity = 0)

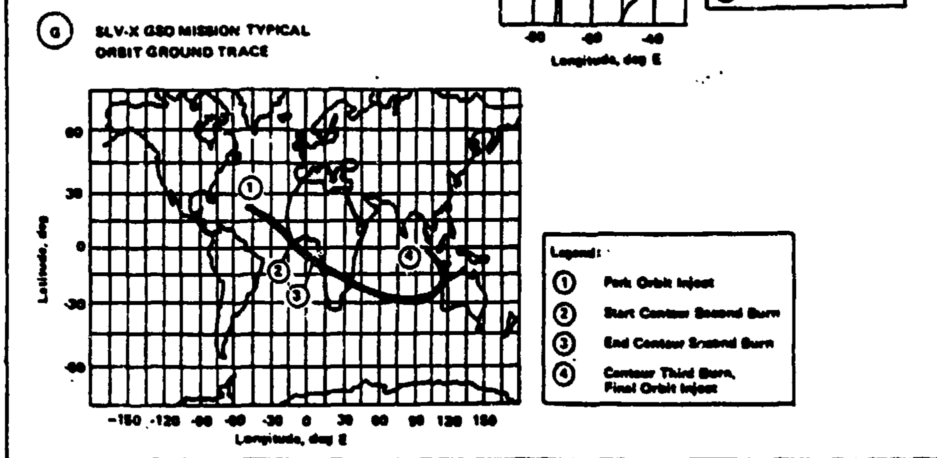
Figure C8-27 The GSO Design Trajectory Meets Performance, Vehicle Design, and Range Safety Objectives (Sheet 1)

(E) GSO MISSION INJECTION ACCURACIES

Component	Injection Accuracies			
	Position, nmi		Velocity, fps	
	Reqd	Actual	Reqd	Actual
Radial	50	32.1	75	52.1
Normal	40	21.2	24	23.1
Tangential	66	38.8	16	11.3

Note:
Required injections accuracies are 3 sigma limits.





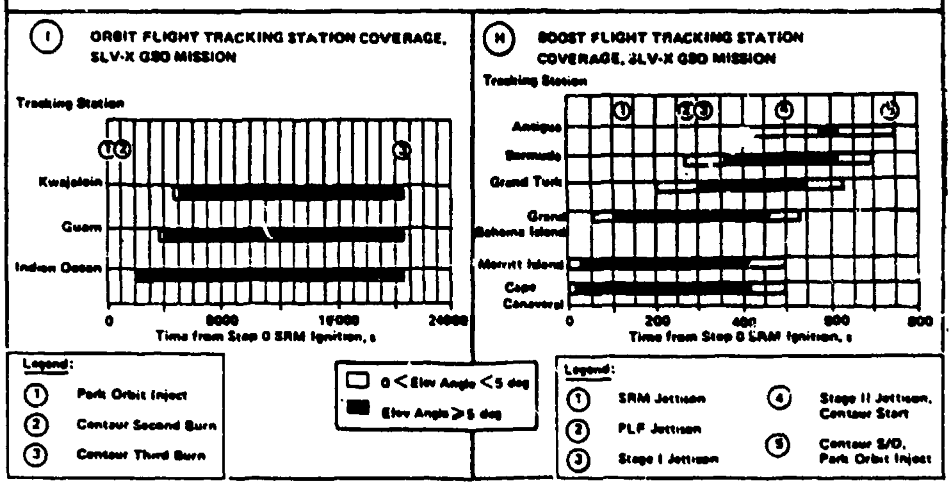


Figure C8-27 (Sheet 2)

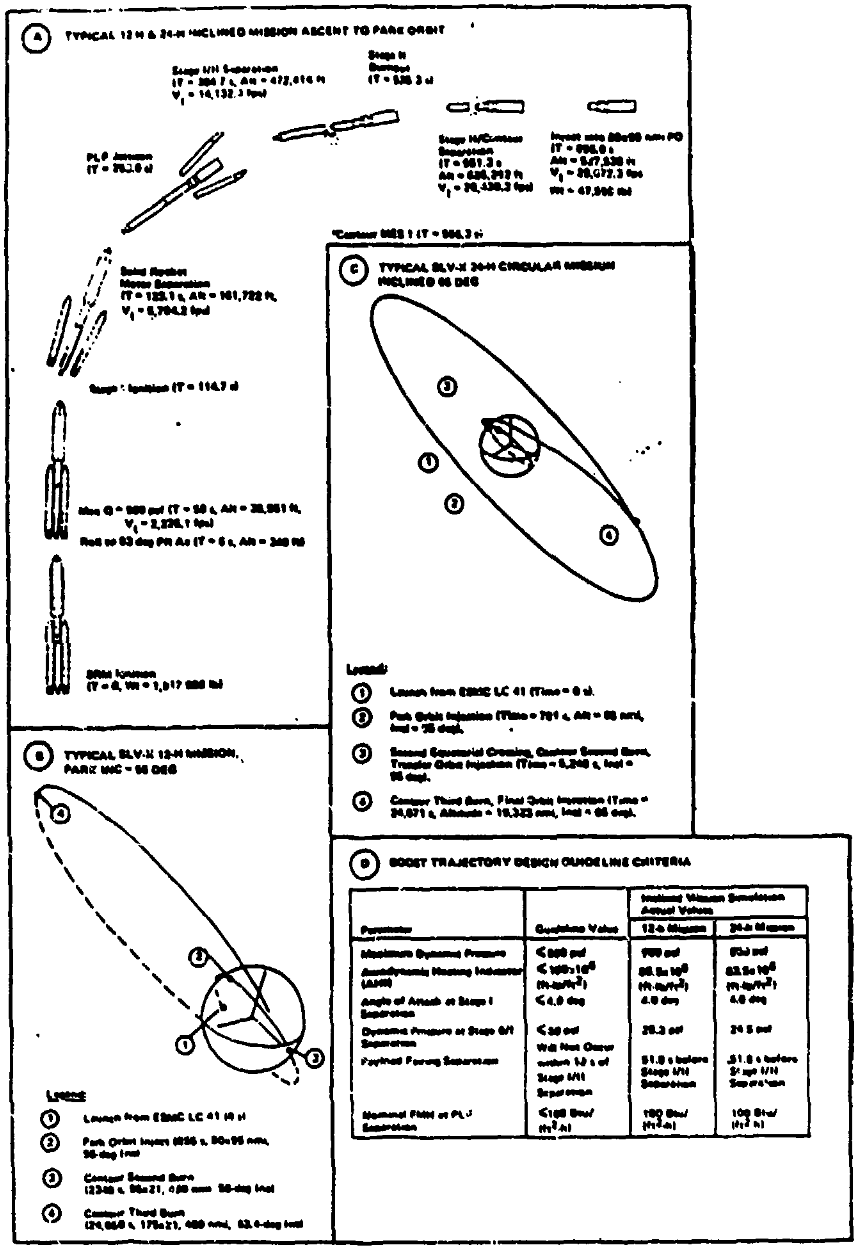


Figure C8-28 Design Trajectories for the Inclined Mission Safety Performance, Vehicle Design, and Range Safety Objectives (Sheet 1)

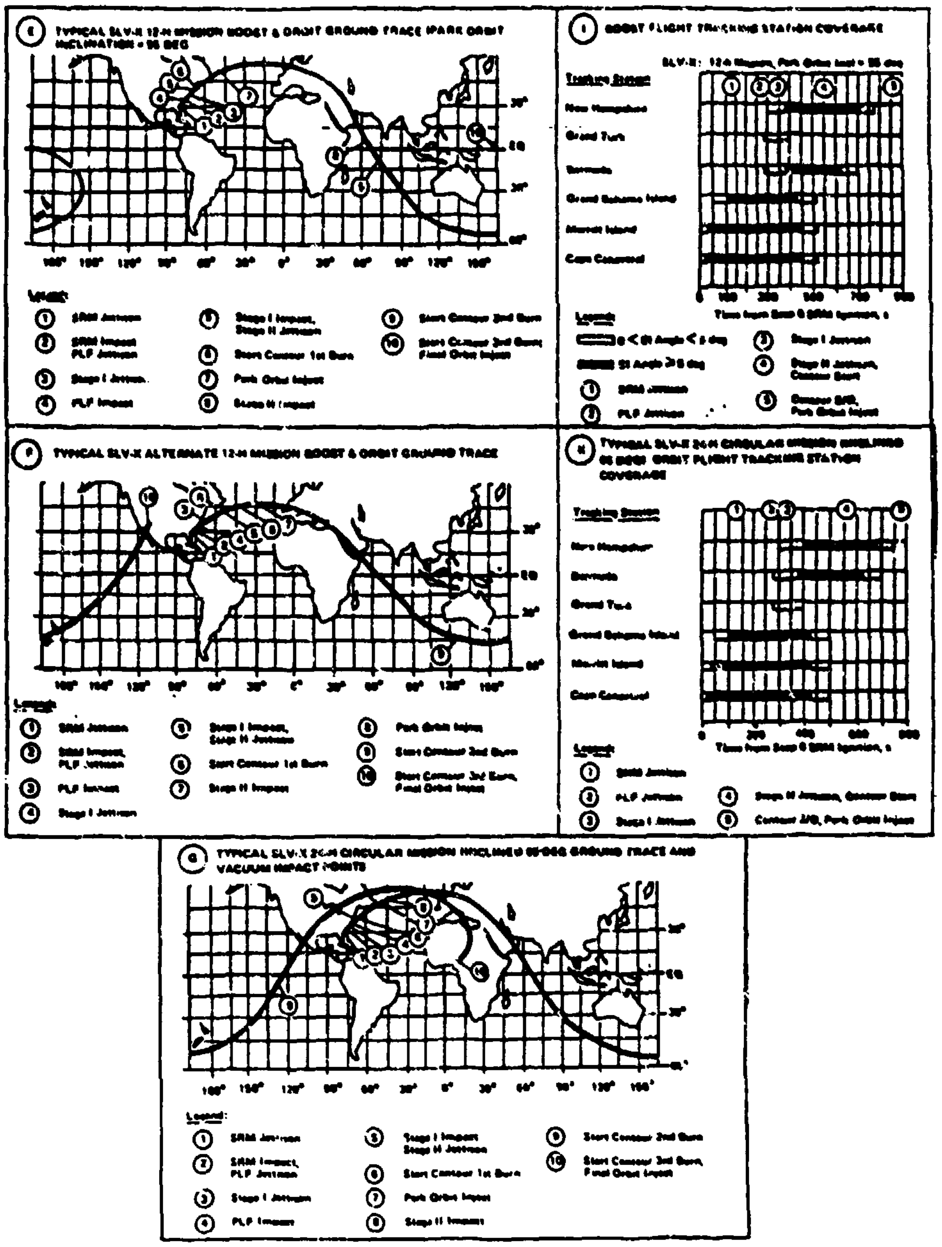


Figure C8-28 (Sheet 2)

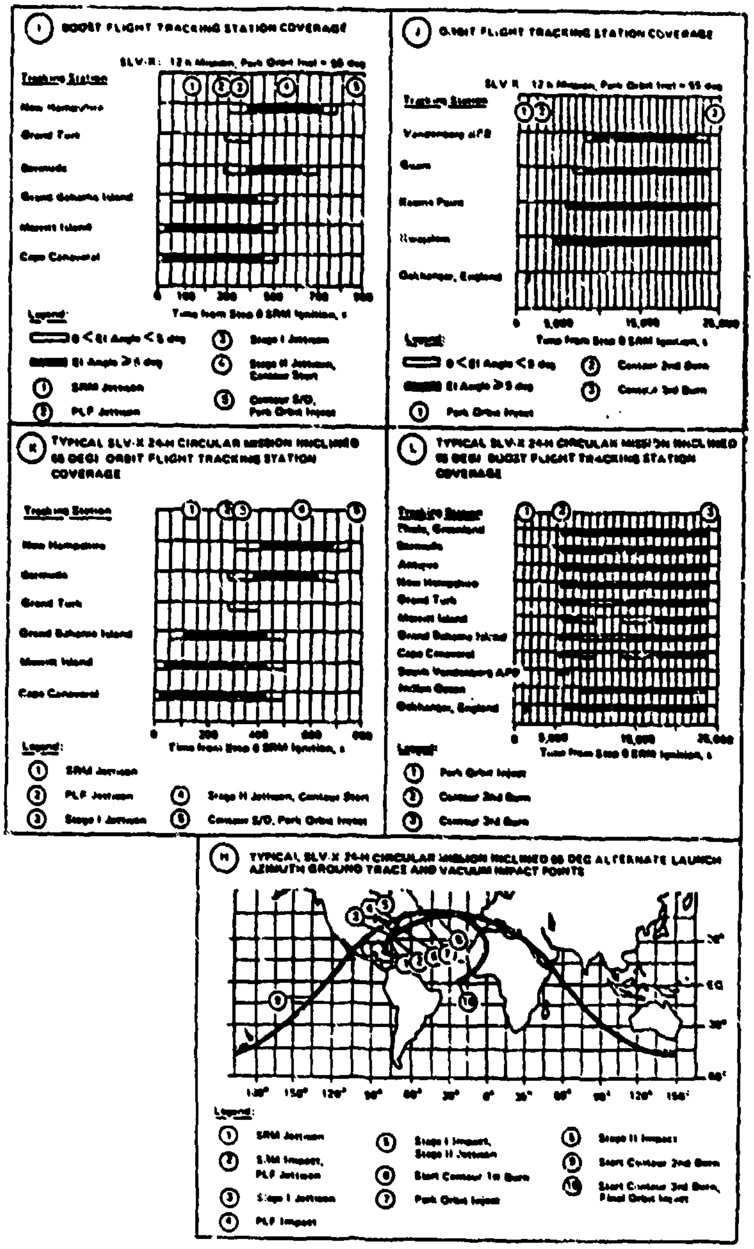


Figure C8-28 (Sheet 3)

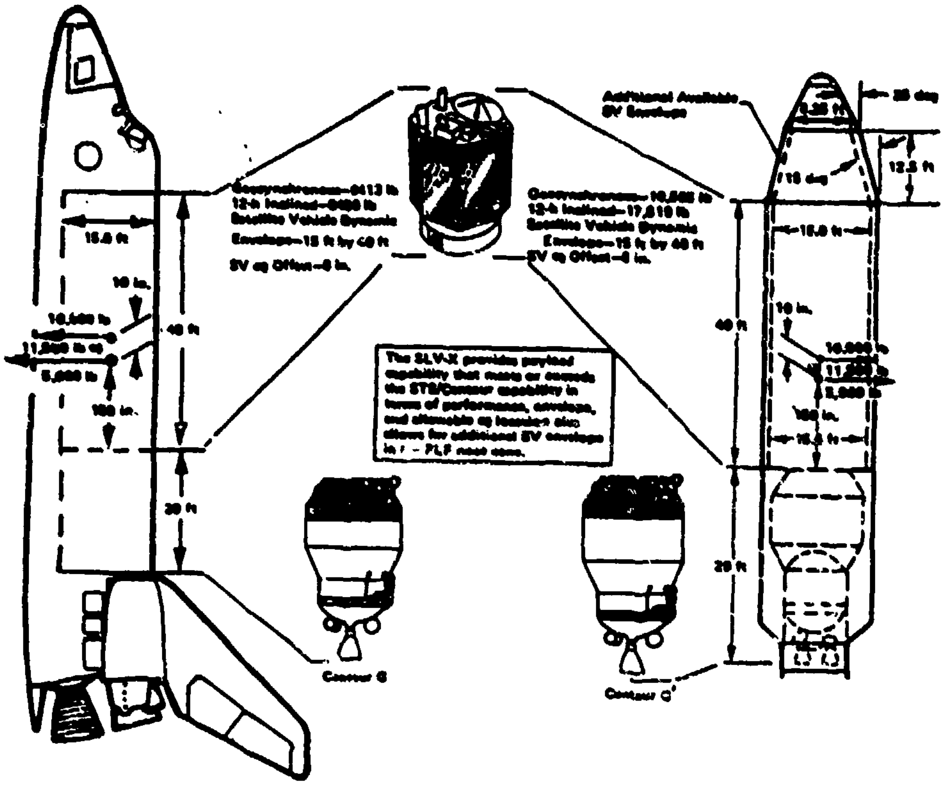


Figure C8-29 TIV provides Shuttle equivalent payload capability.

Appendix C9
Centaur G-Prime

APPENDIX C9
CENTAUR G PRIME

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APPENDIX C9
CENTAUR G PRIME

C9.0 INTRODUCTION

The following data were extracted from the National STS Program Document "Space Shuttle Data for Planetary Mission Radioisotope Thermoelectric Generator (RTG) Safety Analysis," JSC 08116, Feb 15, 1985, NASA, L.B. Johnson Space Center, Houston, Texas 77058.

This wide-body version of the standard Centaur Vehicle is being developed by General Dynamics/Convair Division for NASA as a high-performance STS upper stage for such missions as the Galileo mission to Jupiter and the Ulysses (formerly the International Solar Polar) Mission (ULS). The NASA-unique Centaur G-Prime Vehicle is the "long" derivative of the DOD-unique Centaur G Vehicle; both versions are designed as components of the STS.

C9.1 GENERAL DESCRIPTION

The Centaur Vehicle, a high-performance STS upper stage, (Figs. C9-1 and C9-2) consists of a 10-foot-diameter liquid oxygen tank that transitions to a 14-2-in.-diameter hydrogen fuel tank. The overall vehicle is 29.1 feet long. The cryogenic tanks are insulated with combinations of helium-purged foam blankets and insulation shields. The forward end of the vehicle consists of a bolted-on forward adapter comprised of a cylindrical section and a conical section. This adapter provides mounts for all electronics packages. The aft end of the vehicle consists of a cylindrical aft adapter and a pyrotechnic separation ring.

The Centaur avionics system performs the functions necessary for autonomous control of the Centaur Vehicle from a predefined safe separation distance following Centaur deployment from the Orbiter through post-separation maneuvers.

The Centaur is provided structural, fluid, and avionics support in the Orbiter prior to deployment by means of airborne support equipment designated as the Centaur Integrated Support System (CISS). The CISS consists of Centaur Support Structure (CSS), a deployment adapter, and associated electronics and fluid lines.

C9.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C9.2.1 Centaur Structural System

The components of the Centaur structural system are the propellant tanks, adapters, and the Centaur Support Structure (CSS). The propellant tanks consist of a liquid-oxygen oxidizer tank and a liquid-hydrogen fuel tank, as illustrated in Figure C9-3. The tank volumes are designed to provide total propellants of 46,000 lb at a burn mixture ratio of 5.0 to 1.0 oxygen to hydrogen. Besides containing the main engine propellants, the tanks establish vehicle primary structural integrity and support vehicle systems and components. The tanks are of monocoque construction formed by a series of short stainless steel cylinders weldnd together. The basic tank material is 301 CRES. Ends of the tanks are formed by stainless steel bulkheads. The fuel and oxidizer tanks are separated by a double-walled intermediate bulkhead. The tank skin is stabilized at all times by internal pressure or by the application of mechanical stretch. During assembly and certain checkout operations, support is provided by applying stretch to ground handling adapters attached to the forward and aft cylindrical tank rings. After erection, structural integrity is provided by pressurization gas supplied to the tanks from external sources. With propellant in the tanks, pressure is maintained by propellant boil-off.

The liquid oxygen tank, Figure C9-4, consists of a 120-inch-diameter cylindrical section closed at each end by an ellipsoidal adapter. The forward end of the cylindrical section terminates at the junction of the intermediate bulkhead and aft end of the liquid hydrogen tank cone section. The aft end of the liquid oxygen tank cylindrical section terminates at the junction of the aft bulkhead. The aft bulkhead has welded brackets and fixtures that support propellant lines, helium and hydrazine bottles, main engines, electrical boxes, and radiation shields.

The liquid hydrogen tank, Figure C9-4, consists of a 170-inch-diameter cylindrical section closed by an ellipsoidal forward bulkhead and conical aft bulkhead. The aft bulkhead attaches to the forward buldhead cylindrical section joint of the liquid oxygen tank.

The two propellant tanks are insulated to prevent excessive propellant boil-off. Figure C9-5 shows the insulation system. The liquid hydrogen tank insulation consists of the forward bulkhead insulation and tank sidewall insulation. The liquid hydrogen tank forward bulkhead insulation consists of a three-layer thermal radiation shield and a two-layer foam blanket. The tank sidewall insulation is made up of a thermal radiation shield and two layers of foam insulation extending along the tank. The liquid oxygen tank has an aft bulkhead and sidewall thermal radiation shield but no foam insulation blankets.

Two adapters, a forward and aft adapter (Fig. C9-6), are mounted to the Centaur tanks and serve as structural interface support and transition members. The forward adapter is installed on the forward liquid-hydrogen tank ring. The cylindrical section is a graphite/epoxy structure 170 inches in diameter and 25 inches long. The conical section of a forward adapter is a conical aluminum skin stringer alloy structure with a 170-inch-diameter base. It is 47 inches long and 108 inches in diameter at the forward end. The forward adapter serves as a mount primarily for the avionics boxes and as a structural interface with the spacecraft mission-peculiar adapters.

The aft adapter is a 10-ft-diameter, 11.2-inch cylindrical graphite/epoxy structure with attachment rings at each end. The adapter distributes Centaur Integrated Support System (CISS) support loads into the Centaur tank and provides an interface for attaching the separation system. The forward ring bolts to the liquid oxygen tank aft ring and the aft ring, attaches to the separation ring.

Support structure is mounted on the aft adapter for the vehicle separation springs, field and electrical disconnect panels, radiation shields, and wiring.

C9.2.2 Centaur Support Structure

The Centaur Vehicle and spacecraft are supported within the Orbiter by the CSS. The support structure remains within the Orbiter after deployment. As shown in Figure C9-7, the support structure consists of semicircular aluminum beams connected by longitudinal side beams and a keel beam. This construction supports the deployment adapter to which is bolted the aft end of the Centaur. Loads are transferred from the CSS to the Orbiter by means of steel trunnion pins. Additional CSS trunnions extend from the Centaur equipment module to latch the forward portion of the Centaur to the Orbiter. The CSS also provides support for helium storage bottles and the CISS avionics system.

The deployment adapter is a cylindrical structure, 10 feet in diameter and 44 inches high, which supports the aft end of the Centaur. During the Centaur deployment sequence, the deployment adapter rotates around the CSS trunnion support pins to bring the Centaur to the desired separation attitude. The deployment adapter includes the separation ring and also supports the fluid system and electrical disconnect panels. The CISS electronics hardware is also supported on the deployment adapter.

The separation ring portion of the deployment adapter contains the pyrotechnic Centaur separation mechanism designated as the Super*Zig system. The separation ring is a 10-foot-diameter, 5-inch-long aluminum cylinder. The rings bolt to the aft adapter (Fig. C9-8) of the Centaur and the CISS deployment adapter. When the pyrotechnic Super*Zip explosive cord is fired, the separation ring is fractured, and a spring system pushes the Centaur away from the Orbiter. Should both the primary and backup Super*Zig pyrotechnic devices fail to operate, the Centaur would be lowered back into the payload bay.

C9.2.3 Centaur Propulsion and Hydraulic Systems

Primary thrust is provided to the Centaur by two Pratt & Whitney RL10A-3-3A engines developing 33,000 lb of total thrust. These engines are regeneratively cooled and turbopump fed. Each propellant (liquid hydrogen and oxygen) is injected into the throat chamber where combustion is initiated by an electric spark igniter. Heat of combustion vaporizes the fuel as it passes through the thrust chamber tubes. This expansion of gas is the energy source that operates the turbine. A propellant utilization system controls the liquid oxygen flow rate to ensure that both propellant tanks will be emptied simultaneously. Sensor probes to detect the propellant liquid levels are mounted within the tanks. The engines operate at a 5:1 mixture ratio of oxidizer to fuel and have a specific impulse of 446.4 seconds. Tank pressurization maintains an adequate flow of propellants to the turbopumps. Control pressure is isolated from the propellant prevalves and engine inlet valves by redundant pyro valves that also provide two-failure tolerance against inadvertent engine operation. The pyro valves will be fired open after Centaur is deployed a safe distance from the Orbiter.

Two identical and separate hydraulic power supply systems provide the force to gimbal each of the main engines. A power package assembly and two actuators are the main components of the system. Each power package contains an engine-driven pump that supplies pressure to the actuators. This pump is coupled to the engine turbine drive and operates during the burn phase when the engines are firing. During the coast phase, a recirculation pump circulates hydraulic fluid through the system. A diagram of the hydraulic system is shown in Figure C9-9.

C9.2.4 Reaction Control System

The Reaction Control System (RCS) is a system of small rocket engines mounted on the periphery of the Centaur liquid oxygen tank. This engine system provides thrust for attitude control, settling of propellants, and for making separation and orientation maneuvers. The system consists of 12 6-lb thrust units, a positive expulsion tank with 170-lb hydrazine capacity, two parallel sets of pyro valves, one fill/drain valve, two pneumatic checkout valves, and heated feedlines. The pyro valves are fired open, pressurizing the system and allowing hydrazine flow to the thrusters after Centaur is deployed a safe distance from the Orbiter. The arming mechanism is provided by the dual-failure-tolerant arm/safe sequence (DUFTAS)(See Fig. C9-16) and is two-failure tolerant against inadvertent operation.

C9.2.5 Fluid Systems

Figure C9-10 is a stylized schematic of the Centaur Vehicle pneumatic system, which consists of the propellant tank pressurization system, pneumatically actuated valve control system purge systems, helium supply, and the intermediate bulkhead relief system. All of these individual pneumatic systems with the exception of the bulkhead relief system are interconnected and function as a single system.

The pressurization system, Figure C9-11, consists of valves, tubing, and components for pressurizing the propellant tanks. The Digital Computer Unit (DCU) provides helium pressurant control before engine starts and liquid oxygen tank pressurization during engine burns. The liquid hydrogen tank is pressurized during engine burns by gaseous hydrogen diverted from the main engines. Tank pressures are monitored by the DCU from outputs of redundant transducers in each tank. Preprogrammed logic defines the desired pressure level in the tanks and sequences for pressure changes throughout the mission. Before the Centaur is deployed, pressurization of the two propellant tanks is controlled by the five computer control units located on the CISS.

The pneumatic-actuated valve control system consists of tubing and valves that provide actuation pressure for the tank fill/dump valves, backup tank vent valves, and the four liquid hydrogen tank zero-g vent system valves.

The purge system, Figure C9-12, consists of solenoid valves, flow control orifices, tubing and components to direct helium purges to various vehicle systems at various times during the mission. The capability exists, for example, to supply a helium purge to the hydrogen tank insulation blanket during a post-abort landing. A helium purge is provided to the blanket during prelaunch operations. The LO_2 tank sensing line is purged with helium gas to remove impurities. Two helium bottles mounted on the Centaur aft bulkhead contain the gaseous helium required for purging and pressurization.

Vent systems on the Centaur provide the redundant ground, ascent, and on-orbit venting of the propellant tanks to maintain safe pressure levels. The system consists of valves mounted on ducts that extend from the tanks to disconnects located on the Centaur/CISS propellant tanks as shown in Figure C9-13.

The fill/dump system (Figure C9-14) is designed to ensure Centaur compatibility with all Shuttle abort modes that occur before vehicle deployment. The system has been sized to provide single-failure-tolerant propellant dump capability within 250 seconds. With this system a simultaneous dump of liquid hydrogen and oxygen can be accomplished while the Orbiter is above 100,000 feet altitude. The fill/dump system is a foam-insulated duct system with a redundant set of normally closed dump valves. The ducts interconnect the propellant tanks to self-sealing disconnects in the Centaur/CISS umbilical panels. Propellant loading and dumping are accomplished through the same system. The Centaur fluid interfaces with Orbiter are shown in Figure C9-15.

Corresponding to the Centaur fluid systems, interconnecting systems are provided in the CISS. The CISS fluid systems provide interconnecting ducting from the Centaur to Orbiter interfaces and monitor/control the Centaur fluid systems prior to deployment. The CISS propellant tank vent system, for example, consists of valves and ducting that connect the Centaur propellant vent lines with the associated Orbiter overboard ports. The vent lines join at the umbilical panels located on the CISS. At deployment, separation of the CISS vent lines from the Centaur lines occurs along the umbilical panel plane. The CISS fill/dump systems and pneumatic systems are similar in design and operation to the vent system.

C9.2.6 Centaur Vehicle Avionics

The Centaur avionics system integrates many hardware functions into the airborne computer software. These functions include attitude control, sequencing, telemetry formatting, propellant management, steering, navigation, and guidance. Most of the Centaur airborne avionics components are located on the forward adapter as shown in Figure C9-16.

The principal element of the avionics system is the Teledyne Systems Company Digital Computer Unit (DCU). The DCU is stored program random-access core machine. The task of the DCU is to receive data measurements, process this data in accordance with a prestored program, and output response information and commands. Functions performed by the DCU include navigation, guidance, vehicle control, Orbiter axes transfer alignment update (if used), platform rotation to remove accelerometer biases, star-scanner operations (if used), sequencing, propellant utilization, propellant tank pressurization, telemetry formatting, and communications to and from the Orbiter.

The control system provides vehicle attitude stabilization and points the vehicle in response to guidance steering commands. These control functions are performed on the basis of analog attitude errors, which are the differences between the actual and commanded attitudes of the vehicle. In response to these errors, thrust vector actuator control signals are generated by the digital autopilot in the DCU. The analog engine actuator commands are sent from the DCU to the servo inverter unit (SIU). The SIU is a mechanical engine-control device that positions the engines based on the DCU actuator commands. Signals from the SIU to a hydraulic servovalve causes hydraulic fluid flow and engine gimbaling during the burn. During the coast phases, attitude control signals are also generated by the DCU based on the input attitude. Commands are digitized on-off commands sent from the DCU to the Sequence Control Unit (SCU). The SCU switch commands activate the reaction control thrusters mounted on the perimeter of the Centaur. The thrusters provide rotational torques that balance the vehicle during coast to hold it in a rate-displacement-limit cycle.

The SCU is an interface device that converts signals from the DCU into switched and/or timed commands that can be used by the requisite vehicle systems.

Propellant utilization management function of the DCU controls the propellant mass flow rate to ensure that the oxygen and hydrogen tanks will be depleted simultaneously. Sensors mounted inside the tanks provide indication to the DCU of the propellant residuals. Flow rates are adjusted based on the ratio of the residual liquid masses.

Pressurization control of the propellant tanks is provided by the computer controlled vent and pressurization system (CCVAPS) functions of the DCU. This function also optimizes helium use and provides failure detection and corrective action for the redundant tank pressurization components. The DCU selects valid tank pressure level measurements, compares actual tank pressures to predetermined values, and commands actuation by appropriate valves to achieve the desired pressure levels.

The electrical power system provides electrical power and distribution for the operation of the avionics and other systems of the vehicle. The system consists of a primary battery, a power changeover switch to select between internal and external power, electrical rise of umbilicals located on the CISS and wiring harness comprising the distribution system.

C9.2.7 CISS Avionics System

The CISS includes avionics support and structural support of the Centaur within the Orbiter. Figure C9-17 is a diagram of the Shuttle/Centaur CISS avionics system. The CISS avionics system provides all Centaur-to-Orbiter electrical interfaces, computer control for all safety functions prior to deployment, electrical power and power control, instrumentation and telemetry, pyrotechnic control for Centaur separation, and monitoring of propellant tanking levels.

The computer control avionics subsystem is made up of five microprocessor Control Units (CUs), two control distribution units, and a digital computer unit. The function of the subsystem is to actively control all safety-related Centaur vehicle operations up through deployment (such as sequencing, pressurization and vent control, and deployment operations). Each of the safety functions (such as tank pressurization) is monitored independently by each control unit. Commands for a function by the control units are executed only when the control units are in majority agreement. A Centaur DCU located on the CISS provides PCM data from all control units to the ground and to the Orbiter. The DCU is also used in the instrumentation and telemetry subsystem.

Electrical power for the CISS and Centaur prior to deployment is supplied by the Orbiter. In the event of failure of Orbiter power, power could still be supplied from the two silver-zinc batteries located on the CISS. The electrical distribution unit provides power control for both Centaur and CISS loads.

The CISS instrumentation and telemetry systems record vehicle tank pressurization and avionics data and transmit these data to ground stations.

C9.2.8 Centaur Integrated Support System Mechanisms

The rotation system and umbilical detachment mechanism are two mechanical system installations on the CISS consisting of identical primary and backup deployment adapter rotation systems. These systems meet the intent of the two-failure-tolerant requirements. Both the primary and backup systems are tolerant to single failures of the drive motors and clutches, and either can rotate the deployment adapter under maximum expected loading conditions. The rotation system rotates the Centaur from the stowed position in the cargo bay to 45 degrees for deployment.

If deployment is aborted, the Centaur is rotated back to the stored position by reversing the drive unit direction. The forward latches of the Centaur are then relatched, and the Centaur cryogenic propellants are dumped prior to reentry and landing.

In the deployment sequence, the Super*Zip is fired, and deployment springs accelerate the vehicle away from the deployment adapter at a separation velocity of one foot per second. This motion causes separation of the umbilical panel disconnects for the fill, drain, dump, and vent lines mounted on the panels. Following Centaur deployment, the payload bay doors may be closed with the deployment adapter in any position between 0 and 45 degrees.

C9.2.9 Pyrotechnic System

Pyrotechnic devices are employed in conjunction with certain CISS and Centaur avionics functions. These devices are used for fluid isolation control and pyrotechnic deployment sequences. The CISS pyrotechnic functions include firing of the Super*Zip detonation cord for Centaur separation and activation of helium purge pyro valves. The helium pyro valves provide capability to supply tank insulation purge during an abort landing.

Pyrotechnic functions are also used during the Centaur flight phase. Reaction control system and pneumatic pyro isolation valves provide two-failure-tolerant protection against inadvertent Centaur operation following deployment. These valves are fired open prior to Centaur main engine Centaur startup. Pyro devices are used to preclude premature RF antenna deployment. After Centaur separation, the RF antennas, which are spring loaded and located on the equipment module, are released by the firing of redundant pyrotechnic pin pullers. Spacecraft separation involves a pyrotechnic deployment sequence, including firing of a Super*Zip detonation cord and actuation of springs to accelerate the spacecraft away from the Centaur. Pyrotechnic valves installed in the balanced thrust vent paths assure no venting into the or biter cargo bay prior to deployment. These valves are fired open after deployment.

C9.2.10 Centaur/Spacecraft Weight Summary

Weight estimates for the Centaur/spacecraft combination are tabulated in this subsection. These estimates are for the Ulysses and Galileo missions because these are the first two missions which will use the Centaur G-Prime upper-stage vehicle.

An initial weight estimate for Centaur with the Galileo spacecraft appears below:

Spacecraft Separated Weight	5,302 lb
Orbiter Mounted Items	2,900 lb
Centaur Integrated Support System	6,900 lb
Centaur Jettison Weight	6,818 lb
Centaur Propellants	43,029 lb
Hydrazine	49 lb
Helium	2 lb
Total Loaded Weight	65,000 lb

An initial weight estimated for Centaur with the Ulysses spacecraft appears below:

Spacecraft Separated Weight	805
Orbiter Mounted Items	2,900
Centaur Integrated Support System	6,896
Centaur Jettison Weight	6,639
Centaur Propellants	45,209
Hydrazine	49
Helium	2
Total Loaded Weight	62,500

Weights of individual major components on the Centaur integrated support system are as follows for Galileo and Ulysses:

	Galileo	Ulysses
Centaur Support Structure	2,417 lb	2,417 lb
Deployment Adapter	637 lb	637 lb
Fluid System Dry	1,451 lb	1,451 lb
Avionics	1,513 lb	1,509 lb
Rotation System	461 lb	461 lb
Separation System	184 lb	184 lb
CISS LVMP Hardware	97 lb	97 lb
Helium	120 lb	120 lb
Trapped Propellant	20 lb	20 lb
Centaur Airborne Support Weight (lbs)	6,900 lb	6,896 lb
Orbiter Payload Chargeable Items	2,900 lb	2,900 lb
Total	9,800 lb	9,796 lb

C9.3 MISSION SCENARIO

C9.3.1 Prelaunch Timeline

As indicated in Figure C9-18, loading of the Centaur propellants begins shortly after initiation of STS External Tank (ET) loading. Ground chill for the Centaur LOX tank generally starts as the ET LOX tank starts fast fill. (The numbers in parentheses after each Centaur operational phase in Figure C9-18 indicate the approximate time in minutes for each phase.) Tanking of the Centaur's LH_2 begins when the Centaur LOX tank is approximately 75% filled. This is performed to assure that the common bulkhead between the LOX and LH_2 tanks does not reverse because of adverse pressures.

Before any tanking operations begin, the payload bay (PLB), Lower Midbody Compartment (LWR MID), and Aft Fuselage (AFT FUS) of the Orbiter are purged of air by flowing GN_2 into the compartments. The exhausting gases leave the Orbiter via any of the vent doors on both left and right sides, which have been commanded open. For the Centaur missions, all vent doors will be closed at the start of tanking operations.

At the initiation of tanking of the Centaur LH_2, Vent 6 is closed so that all the GN_2 flow is forced into the aft fuselage through relief ports in the aft bulkhead at Station 1307. This vent configuration accommodates close scrutiny of potential GH_2 leakage by the Hazardous Gas Detection System (HGDS), which has sensors located at several of the aft bulkhead relief ports.

As can be seen in Figure C9-18, the HGDS is used to monitor only the payload bay and lower midbody during a check of the Centaur system integrity against GH_2 leakage at four hours before launch. The check is performed with the Centaur tanks temporarily pressurized to the higher lift-off pressures. After a successful check, the flight crew will enter the Orbiter (approximately three hours before launch). The ground control will end 31 seconds before launch when automatic sequencing of events from the Orbiter will begin.

C9.3.2 Typical Mission Event Timeline

The orbit part of the operation cycle may differ considerably from one mission to another. For some missions, the crew will deploy or retrieve a satellite and return to Earth in one revolution. For other missions, the Orbiter will stay in space as long as 30 days. For the Galileo and Ulysses missions, the orbital part will include releasing the spacecraft and the Centaur upper stage, the orbiter moving some distance away, staying in orbit for several hours while the crew eats and sleeps, and then preparing to return to Earth. The typical time profile for the Centaur planetary missions from lift-off to cargo deployment is shown in Table C9-1.

C9.3.3 Abort Modes

Four basic abort modes have been developed to provide continuous intact abort capability for Centaur missions during the ascent phase: Return-To-Launch-Site (RTLS), Transatlantic Abort Landing (TAL), Abort-Once-Around (AOA), and Abort-To-Orbit (ATO). The four modes are available during different segments of the ascent flight to provide intact abort capability. Two modes, AOA and ATO, are available after MRVO in the event of an OMS engine failure. Figure C9-19 is an altitude/range profile showing the relationship between RTLS, TAL, and AOA aborts. Figure C9-20 shows the overlapping abort regions for a typical Shuttle/Centaur mission with an SSME failure. Figures C9-21 and C9-22 provide a more detailed timeline for Centaur LH_2 residuals versus abort scenarios. It should be noted that for purposes of RTG concerns on the gralileo and ulysses missions, both the LOX and LH_2 dump systems (valves, lines, etc) must fail to dump during any abort before an explosive environment can exist with the Centaur. In other words, a failure which prevents the successful dumping of only LOX or LH_2 is in itself not a threat to the RTGs.

Table C9-1 Typical Centaur/Planetary Mission Event Timeline

EVENT	TIME (HR:MIN:SEC)
SSME Ignition	-0:00:06.6
SRB Ignition Command	0:00:00
First Motion/Lift-Off	0:00:00.24
Begin Roll Program (V=125 fps)	0:00:07
End Roll Program (V=300 fps)	0:00:15
Begin First Throttle Down (V=428 fps)	0:00:20
Begin Second Throttle Down (V=716 fps) (Optional)	0:00:31
Begin Throttle Up (V=1599 fps)	0:01:08
SRB Separation	0:02:05
3G Throttling Begins	0:07:39
MECO Command	0:08:34
Zero Thrust	0:08:40
ET Separation	0:08:52
OMS-1 TIG	0:10:34
OMS-1 Cutoff	0:13:22
OMS-2 TIG	0:46:10
OMS-2 Cutoff	0:48:29
Open Payload Bay Doors	1:15:00
IMU Realignment	3:00:00
Meal	4:00:00
Centaur/Spacecraft Checkout	5:00:00
Centaur Deploy	6:40:00
Centaur Engine Ignition	7:20:00

The basic TAL abort flight mode may be stated as follows: After selection of the TAL abort mode (TAL abort mode is acquired pre-MECO by selecting an AOA on the abort select switch), the vehicle will accelerate downrange to the TAL MECO targets. At abort initiation the OMS propellant and Centaur propellant dumps are initiated and run to completion, if time permits, to achieve the correct landing weight and center-of-gravity for entry. At 90 seconds before MECO the Orbiter rolls to the head-up ET separation attitude. After ET separation and after a 30-second MPS dump interval, the onboard computers are loaded with the entry operational flight software. The Orbiter then glides to the primary landing site or the weather alternate landing site. Landing sites change as a function of intended orbit inclination. For example, for orbit inclinations near 28 degrees, Dakar, Senegal (on the west coast of Africa) is the primary landing site. For inclinations near 57 degrees, Zaragoza, Spain, is currently the primary landing site. The weather alternate landing site for both of these inclinations is Moron, Spain. A typical TAL abort sequence of events is defined in Table C9-2.

A successful Centaur mission can be achieved from a Press-to-MECO (PTM) situation if the achieved orbit is at least 110 nm (Centaur propellant has not been dumped). An acceptable orbit is achieved with a PTM point at or after approximately 100 seconds prior to nominal MECO and is identified as the Centaur mission completion boundary on Figure C9-20.

Specific flight procedures have been developed to maximize the crew survival changes for multiple SSME failures from lift-off to MECO or single SSME completion boundaries. However, a contingency abort during a mission with a Centaur stage in the cargo bay will probably rupture the Centaur tanks at ditching or crash landing and therefore can cause an explosion.

Centaur deployment is assumed to occur during the fifth orbital revolution, at lift-off plus 24,084 seconds (or 6 hours, 41 minutes, 24 seconds) for the Galileo mission.

The Shuttle flight sequence up to Centaur deployment is divided into two phases to facilitate the overpressure analysis: (1) the ascent phase, including OMS-1 burn and the early coast phase with the payload bay doors closed, and (2) the portion of orbital coast phase prior to Centaur deployment when the payload bay doors are open. The payload bay doors are open from 4,980 seconds to 24,084 seconds (with Centaur onboard).

Table C9-2 TAL Abort Sequence of Events for an Earliest and Latest "Systems TAL" Abort

Mission Elapsed Time (sec)		
Earliest	Latest*	Event
0	0	Lift-off of the Space Shuttle Vehicle
125	125	SRB Separation
240	355	"Systems TAL" Abort Selected
240	355	OMS Propellant Dump Initiated
240	355	Centaur Propellant Dump Initiated
480	415	Vehicle Roll-To-Heads Up Attitude
435	510	OMS Propellant Dump Completed
490	510	Centaur Propellant Dump Completed
570	520	MECO
588	538	ET Separation
600	550	+X ARCS Thruster Firing Initiated (Manual)
605	555	+X ARCS Stop, MPS Dump Initiated (Manual)
635	585	MPS Dump Stop (Auto)
685	610	Entry Flight Software Memory Load
1650	1600	Mach 3.5
2100	2050	Landing

*The latest TAL (MET=355 seconds) is defined as one which permits Centaur residuals of 10,000 lbs at MECO (LH_2 = 2800 lbs, LOX = 7220 lbs). There are also 2500 lb of OMS propellant remaining. The last TAL abort opportunity is at 425 seconds. However, excessive Centaur and OMS residuals remain at MECO (LH_2 = 5980 lb, LOX = 27,650 lb; OMS = 13,000 lb). A post-MECO Centaur and OMS propellant dump is mandatory to eliminate the hydrogen (LH_2) residuals prior to landing. Figures C9-19 and C9-10 are altitude/time and altitude/range respectively for a typical TAL abort.

C9.3.4 Centaur Failure

For Centaur Category 1 failures occurring with the payload bay doors open, ambient conditions on orbit cannot support a LO_2/LH_2 explosion. Tank conditions would be:

LH_2 tank: 21.5 psia, 40°R, saturated vapor
LO_2 tank: 30 psia, 180°R, saturated vapor

while the triple point values are:

hydrogen: 1.02 psia, 25°R
oxygen: 0.02 psia, 98°R

With the bay doors open, the ambient pressure drops to less than 0.1 psia. Freezing of the hydrogen can reasonably be expected to occur at pressures below the triple point pressure of 1.02 psia. Should rupturing of the tanks occur, vapor would blow out of either tank at supersonic velocities. Immediately after exiting the tank, solid (slush) H_2 or

O_2 would form due to isentropic expansion. This solidification acts to prevent or reduce the mixing and interaction of the propellants necessary for explosion. Some solid formation would occur even in the event of a CBM accident. By the time tank pressures are reduced, in just a matter of seconds, solids would also form inside the propellant tanks, and the degree of hydrogen/oxygen mixing would be greatly reduced.

C9.3.5 Centaur Secondary Failure

A Centaur secondary failure is defined as the response of the Centaur vehicle to Shuttle failures. These are also termed as sympathetic or interactive Centaur failures. In certain ET failures, the explosive yield from a primary Shuttle accident is so large that it completely overwhelms the explosive contribution of the Centaur. That is, since a shock wave emanating from an ET explosion would encounter the Centaur tank axially rather than frontally, the Centaur propellants would more likely disperse and burn rather than mix and explode. Intervening Orbiter structures are here assumed to offer no blast wave blockage.

In instances of a low equivalent yield from a Shuttle accident, a high-yield Centaur explosion with significantly higher shock wave overpressures can be produced at the RTGs owing to their proximity to the Centaur.

A timeline representation of an RTLS maneuver is shown on Table C9-30 beginning with the start of Centaur propellant dump and terminating when the residual liquid hydrogen has boiled off. At the end of Centaur dump, it is estimated that 50 lb of LH_2 and 600 lb of LO_2 residual fluid propellants remain. The LH_2 should be boiled off within 50 minutes after touchdown.

Table C9-3 RTLS Timeline

Mission Event	Event Time, sec	Delta, sec	Interval, sec
Centaur Start Dump	T + 255	-	-
Centaur End Dump	T + 505	250	1005
Orbiter Touch-down	T + 1260	755	
Centaur LH_2 Boil-Off Complete	T + 4440	3180	3180

After the 250-sec Centaur propellant dump is completed, about one percent of the total propellants remain during the RTLS maneuver. Given an intermediate bulkhead rupture, the remaining LH_2 propellants could be expected to be driven into the LO_2 tank where an explosion can occur. The predicted overpressure, based on PYRO reference curves, is less than 100 psi. The center of explosion distance is estimated to be about 25 feet. After touchdown, with the Orbiter in a horizontal attitude, the residual liquid propellant is no longer in direct contact even with intermediate bulkhead rupture. Moreover, within 50 minutes, the remaining LH_2 fluids will have vaporized. Modification to the dumping procedures are planned to further reduce the quantity of Centaur residual propellants.

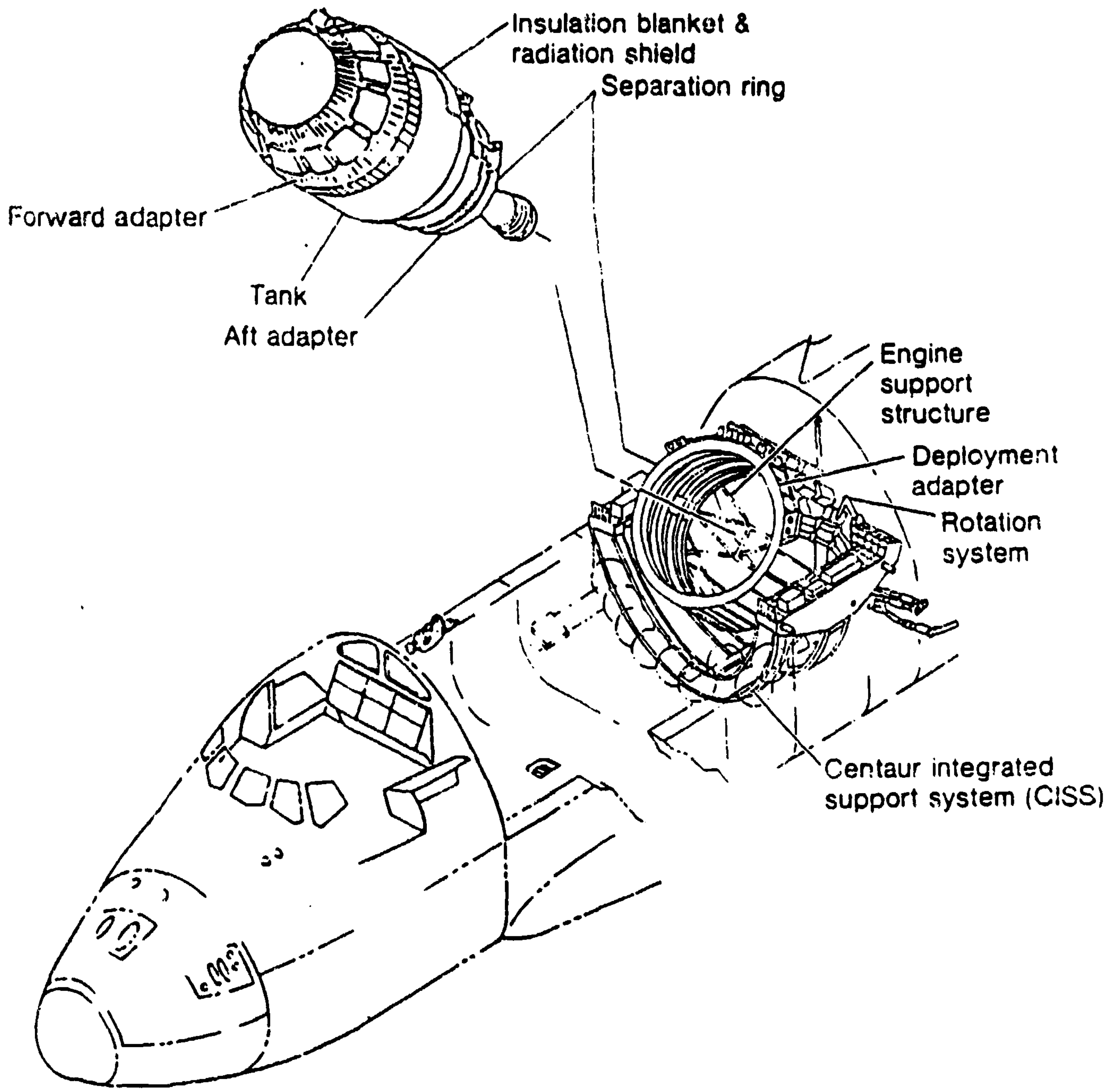


Figure C9-1 Shuttle/Centaur System Summary

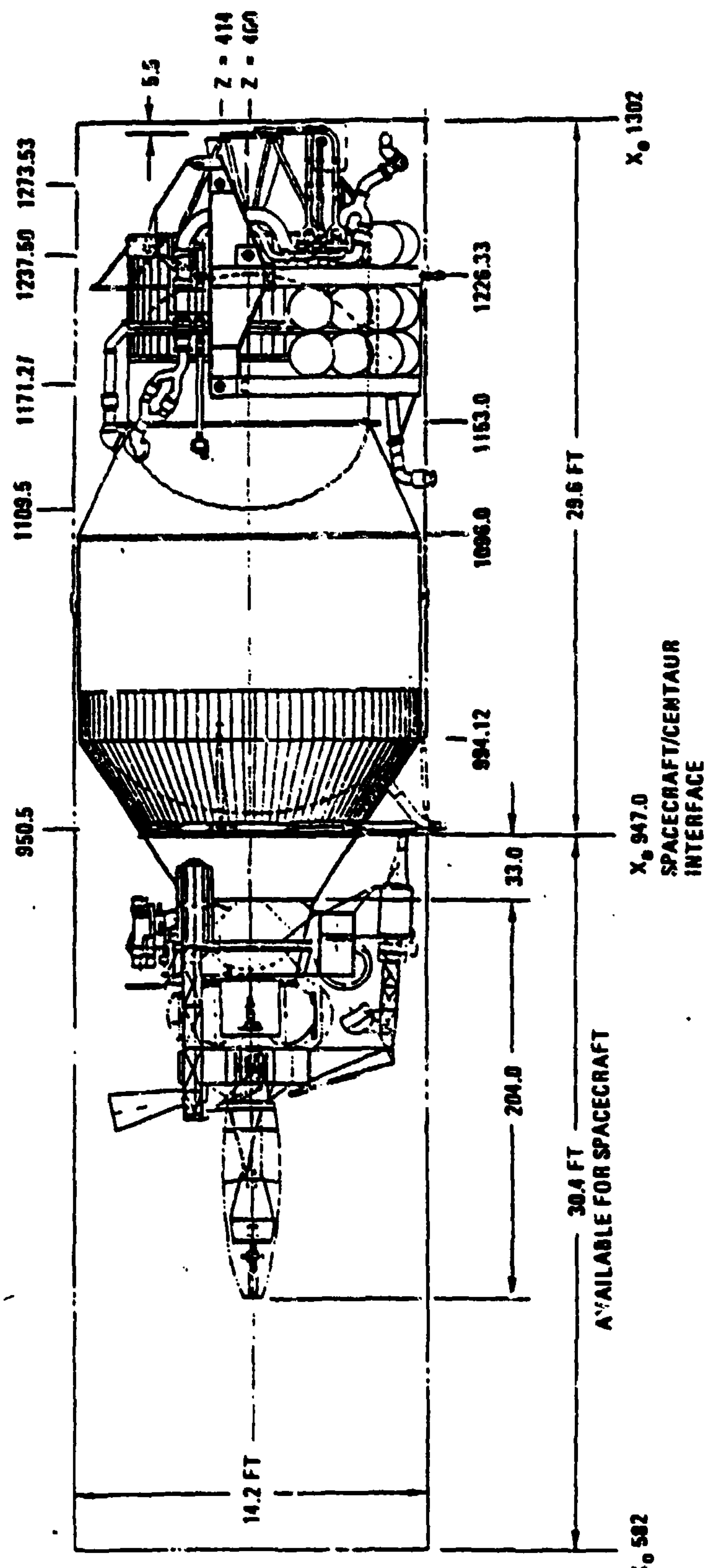


Figure C9-2 Centaur G-Prime for Galileo and Ulysses (Galileo Shown)

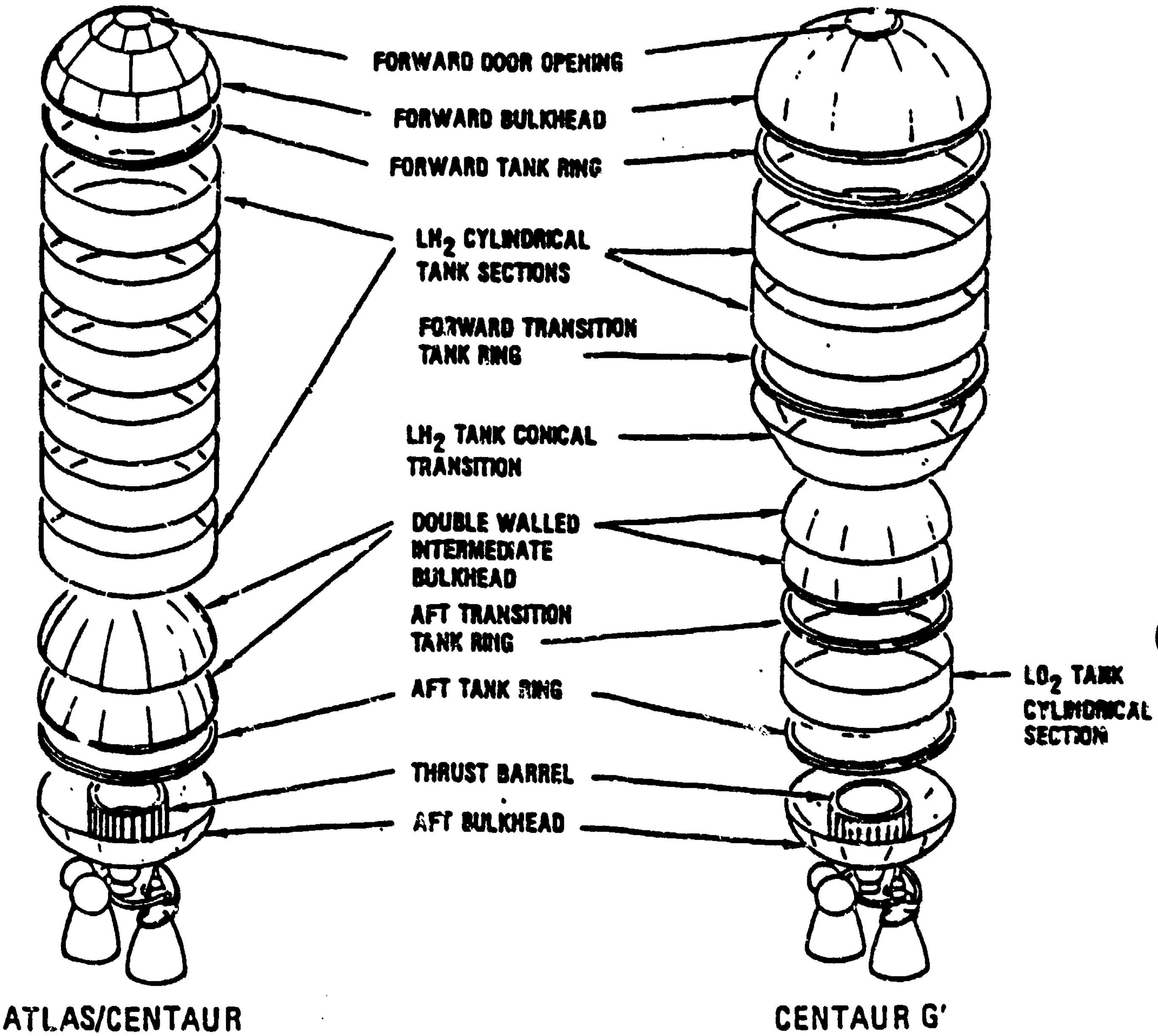


Figure C9-3 Centaur Structural System

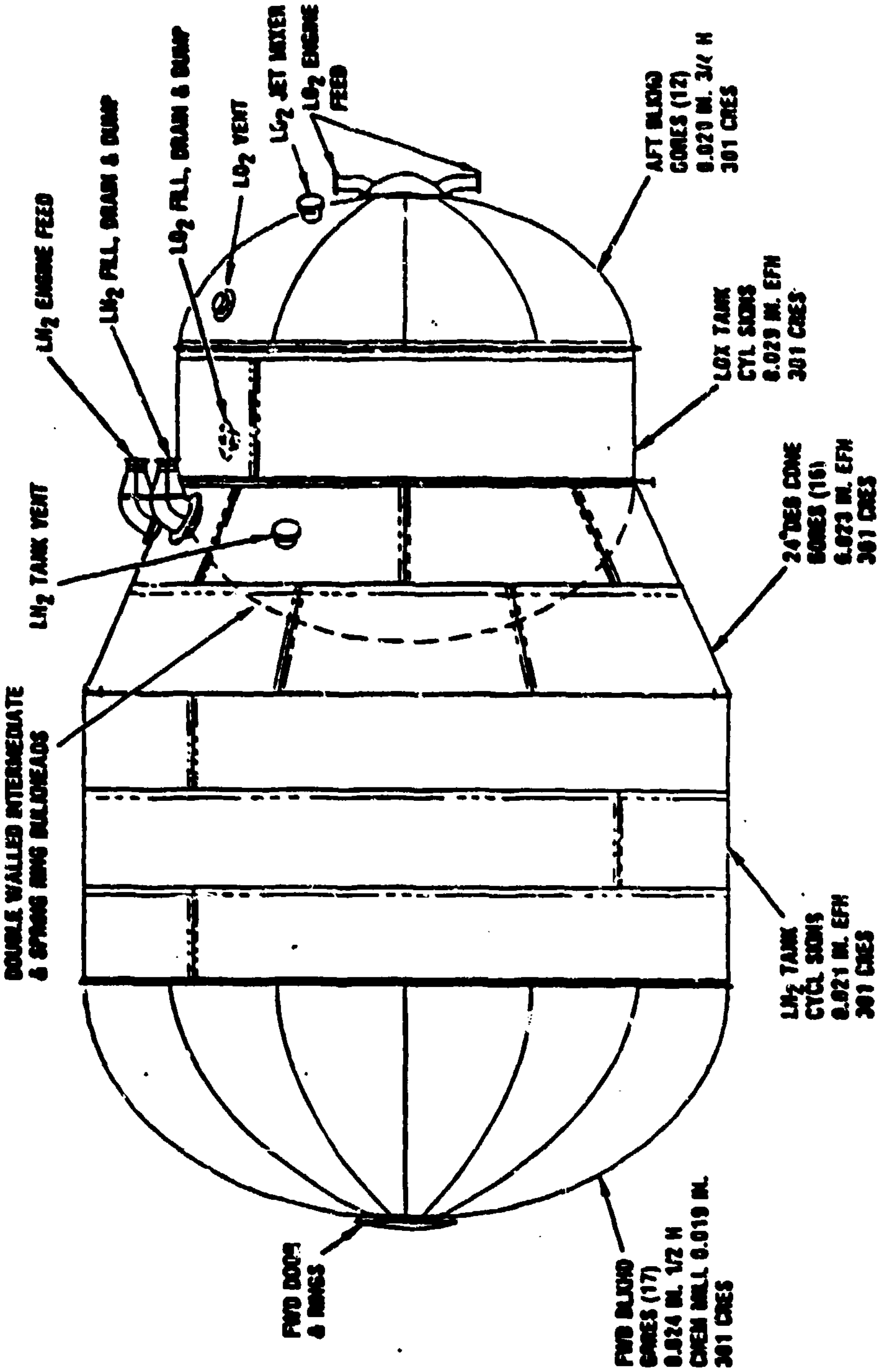


Figure C9-4 Propellant Tank Configuration

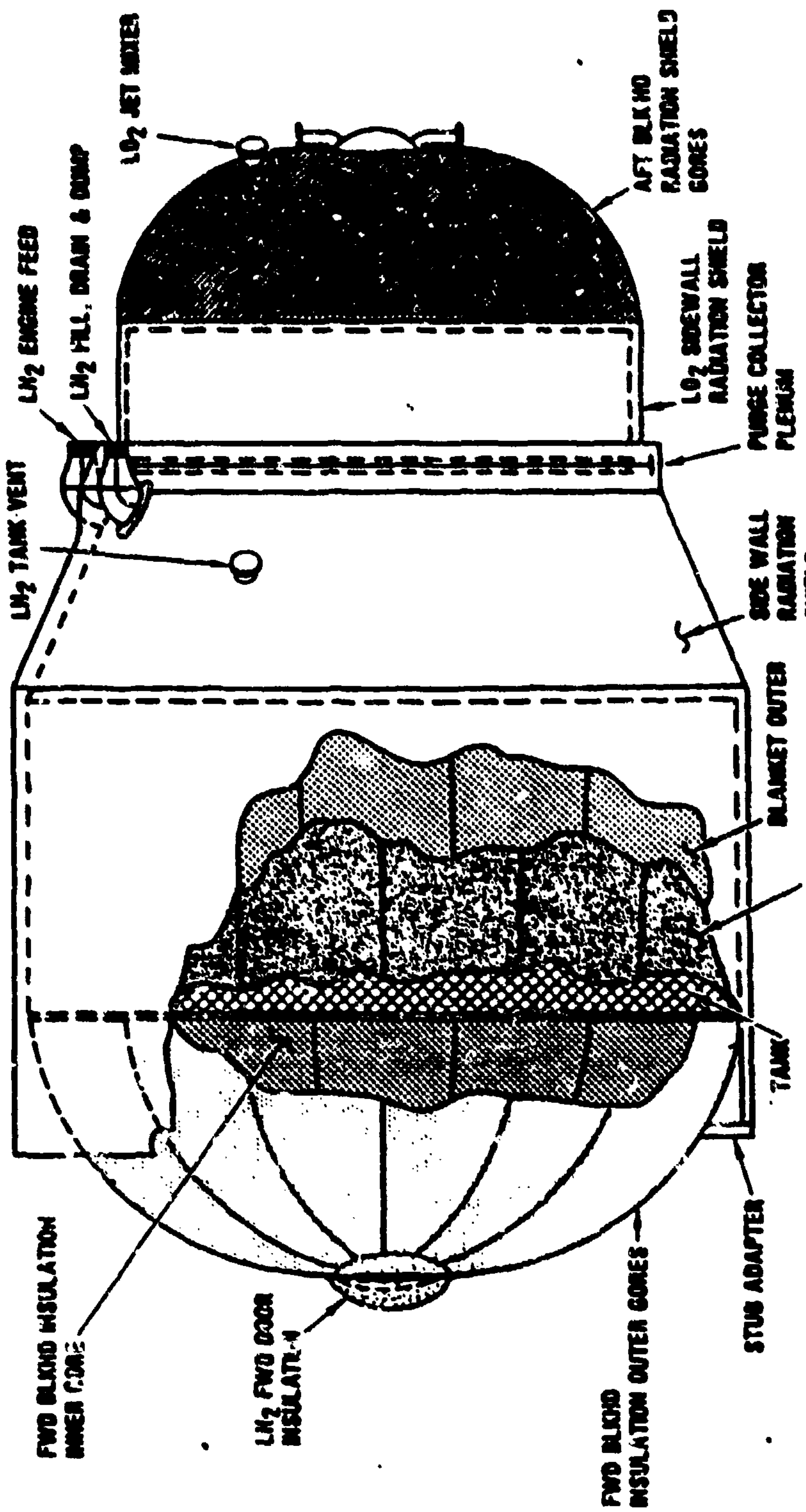


Figure C9-5 Tank Insulation System

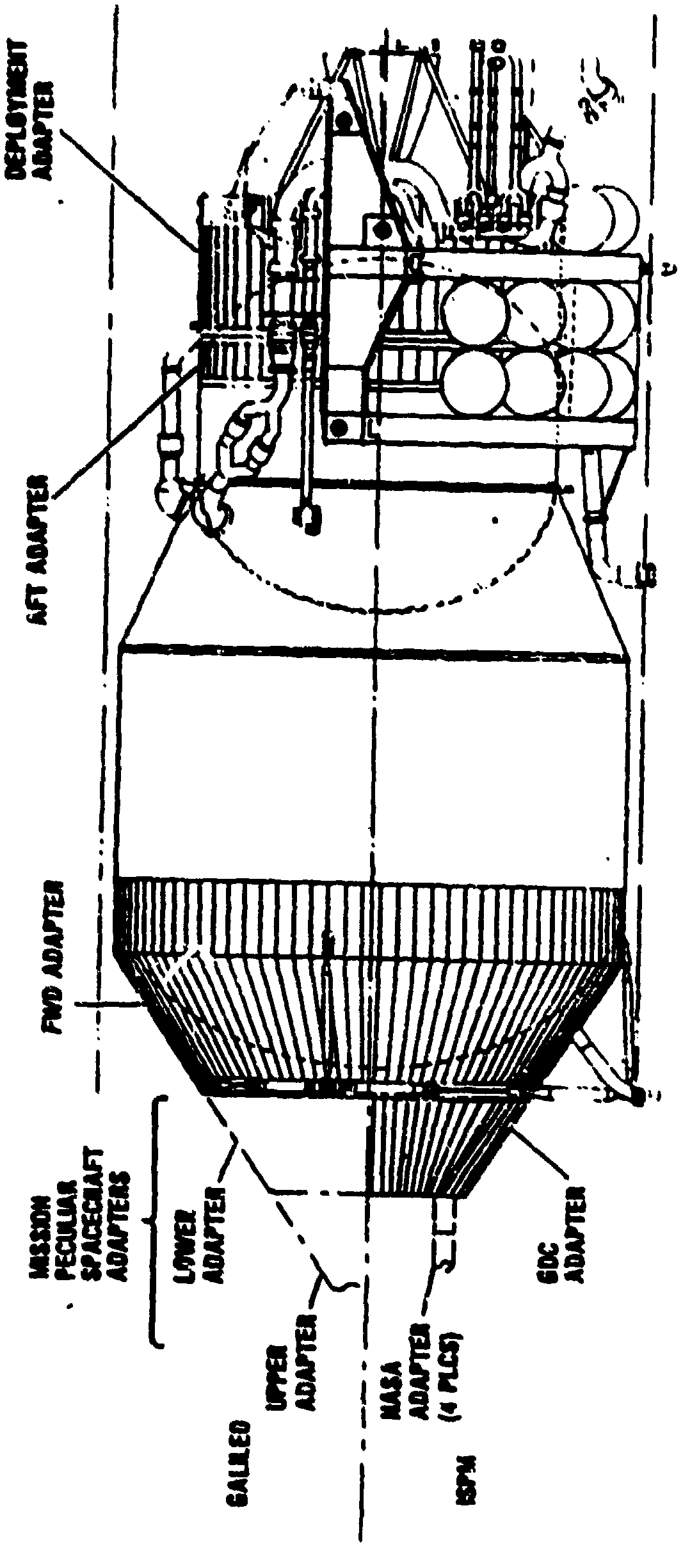


Figure C9-6 Shuttle/Centaur G-Prime Adapters and CISS

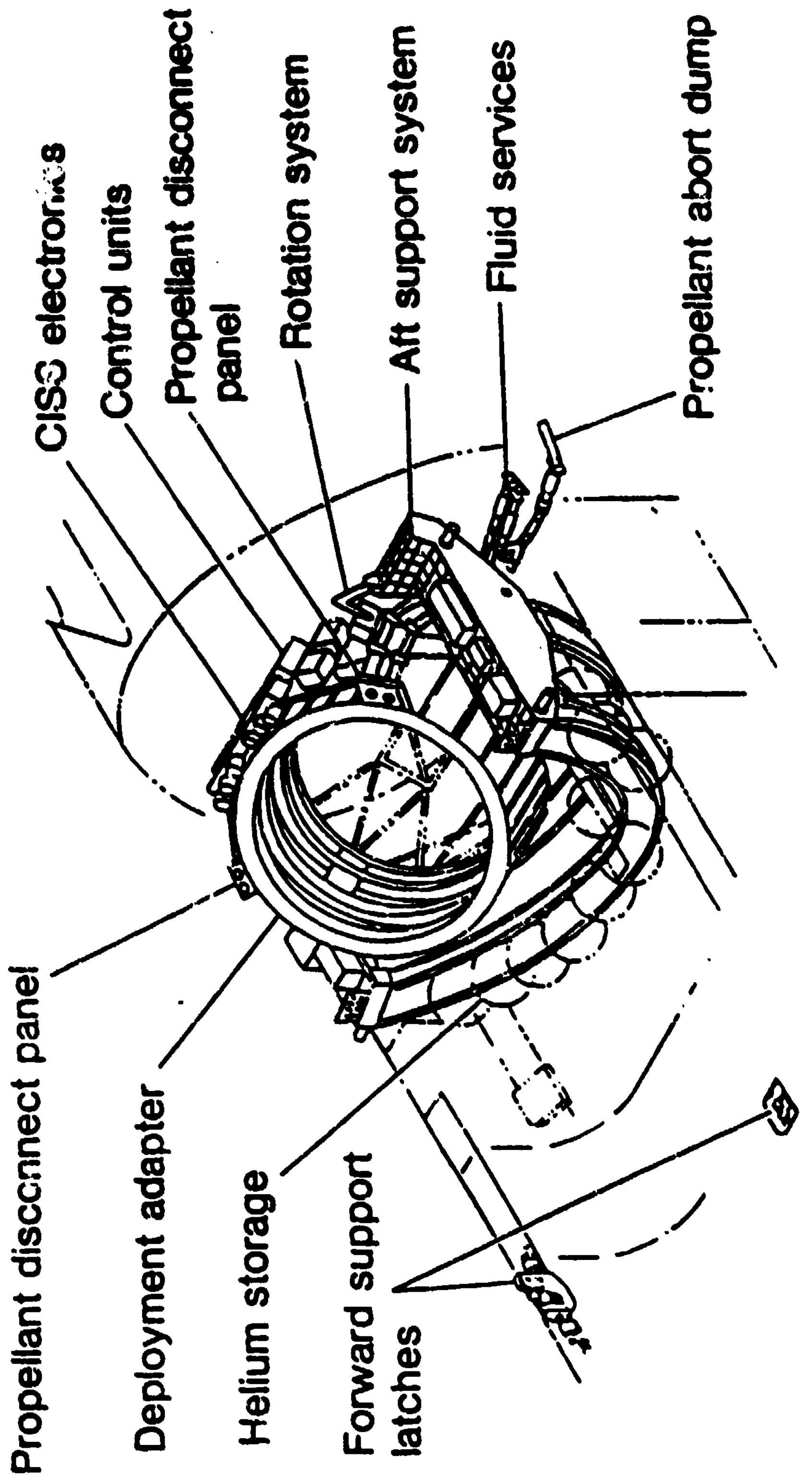


Figure C9-7 Centaur Integrated Support System (CISS)

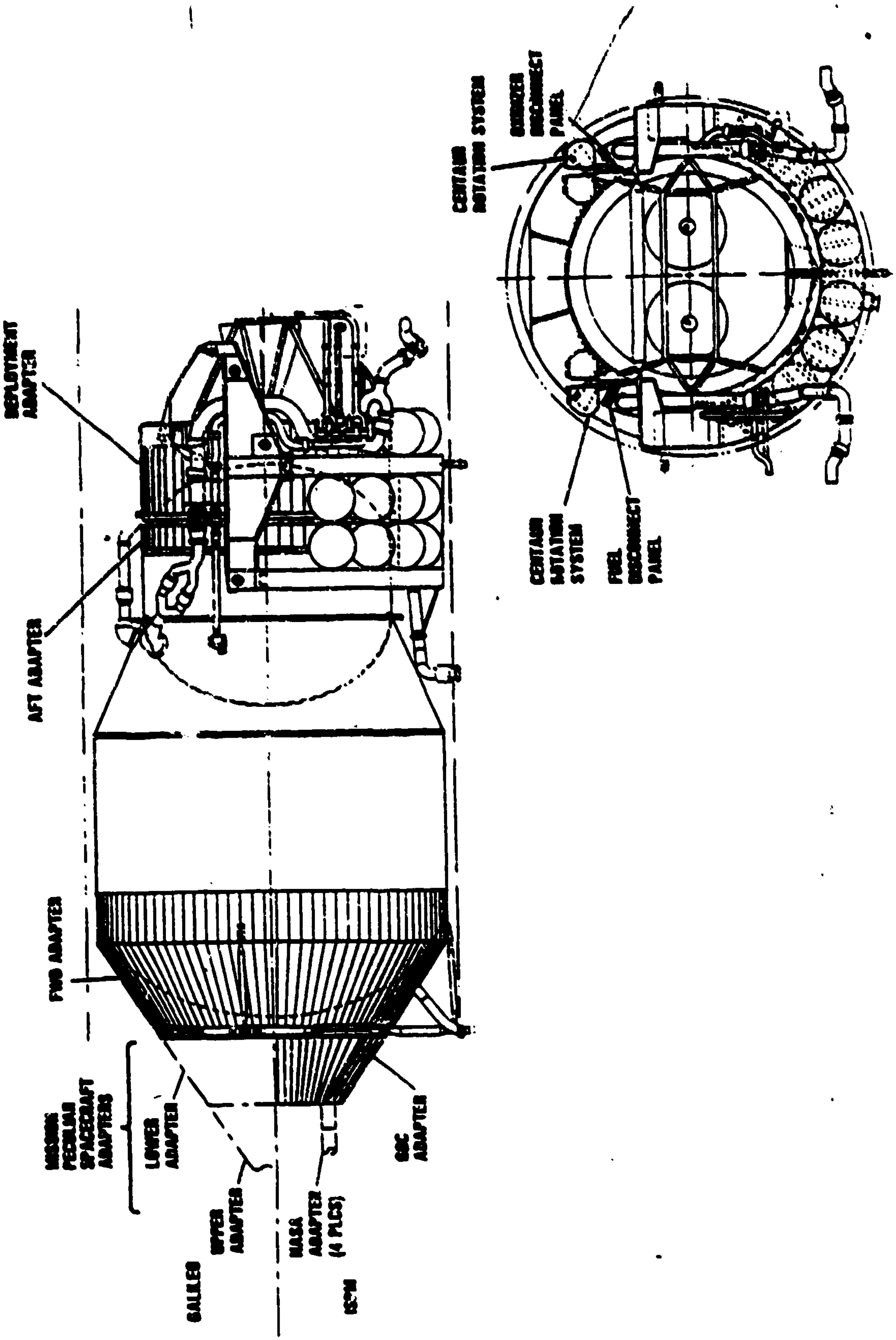


Figure C9-8 Deployment and Separation Mechanisms

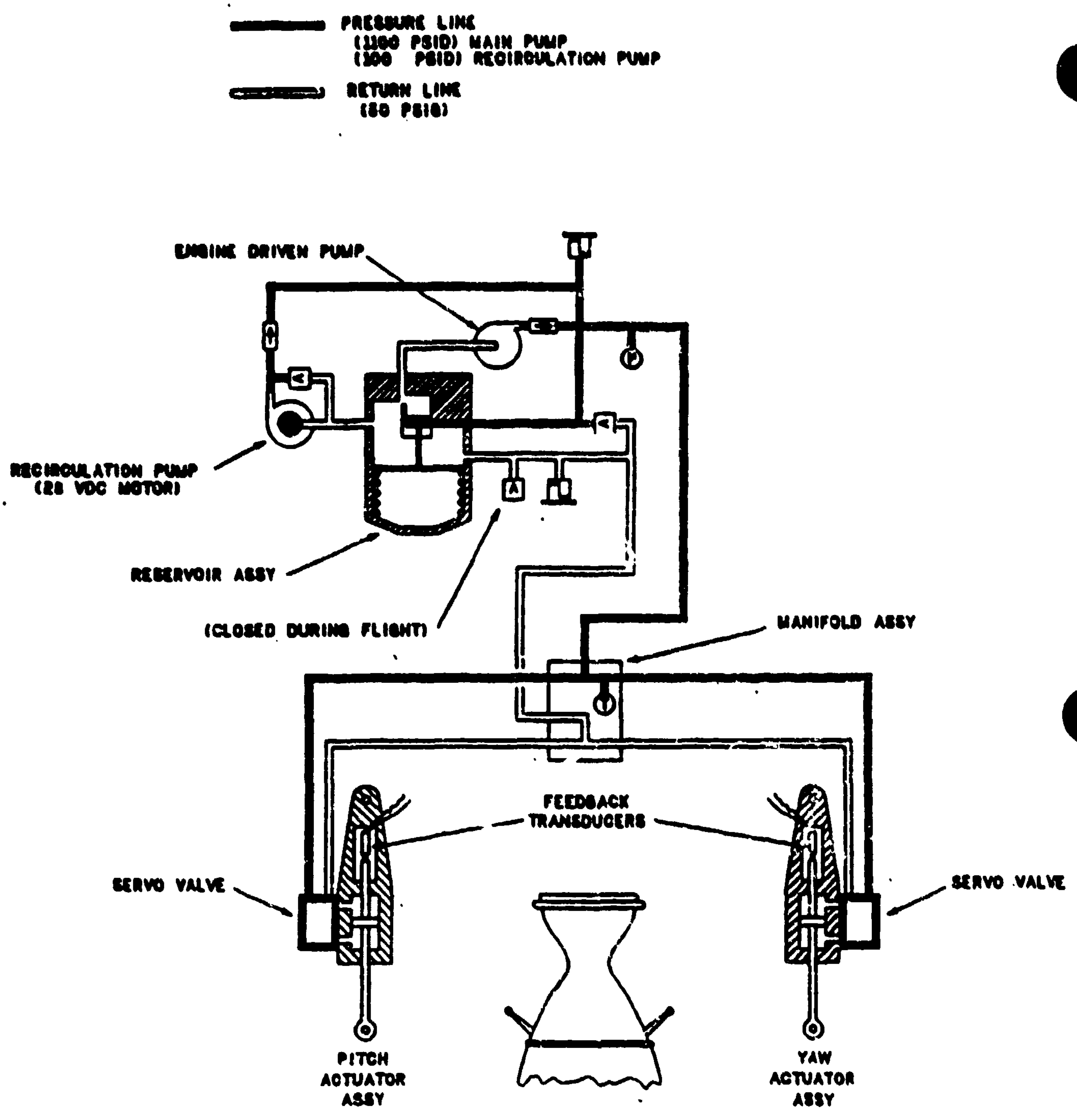
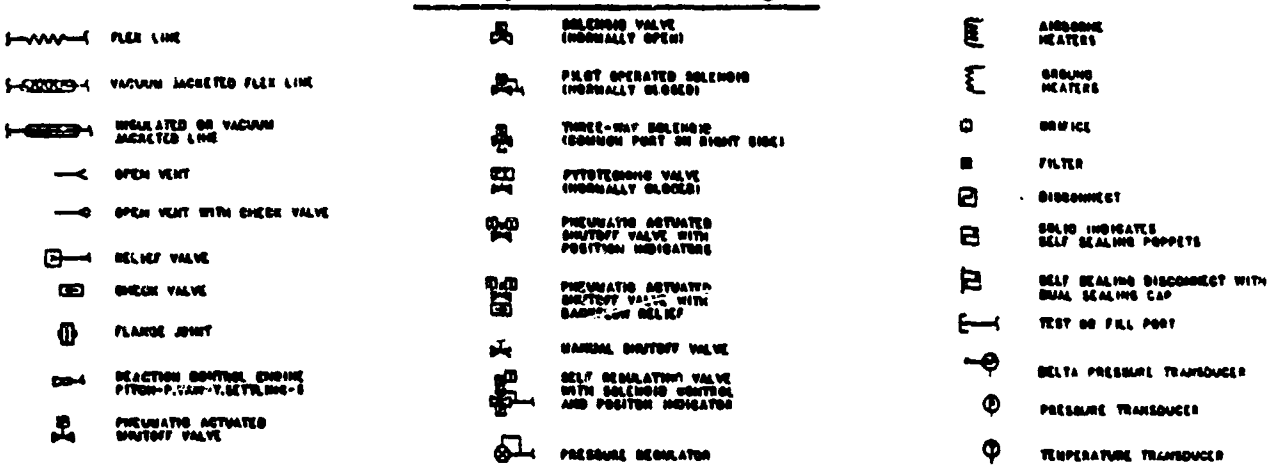


Figure C9-9 Hydraulic System Diagram

Fluid Systems Schematic Legend



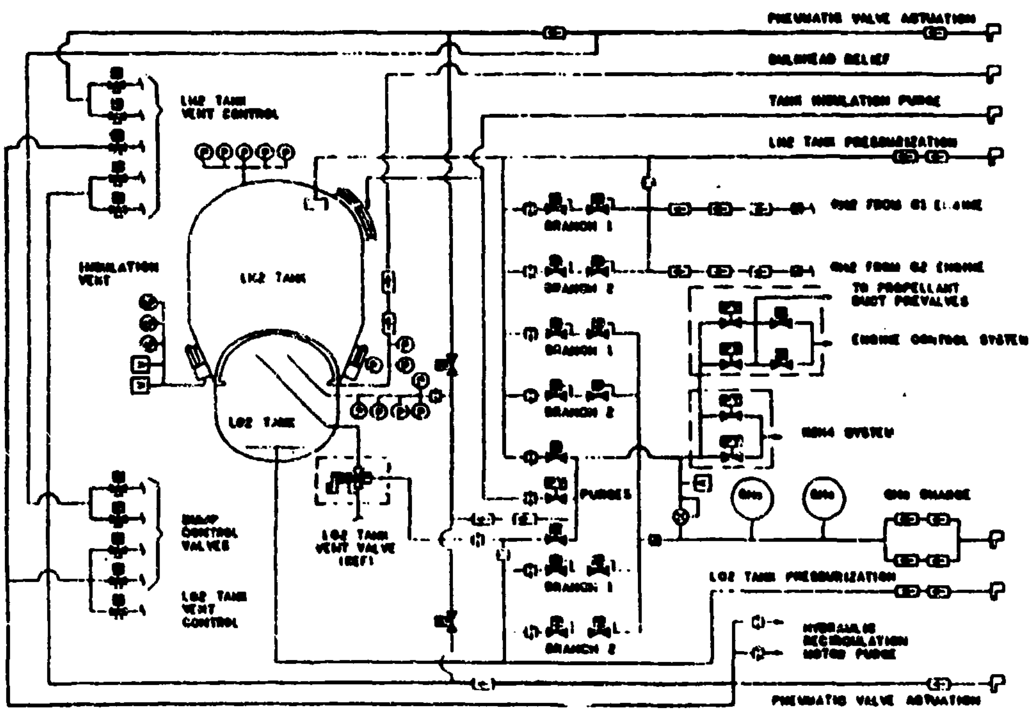


Figure C9-10 Centaur Pneumatic Systems

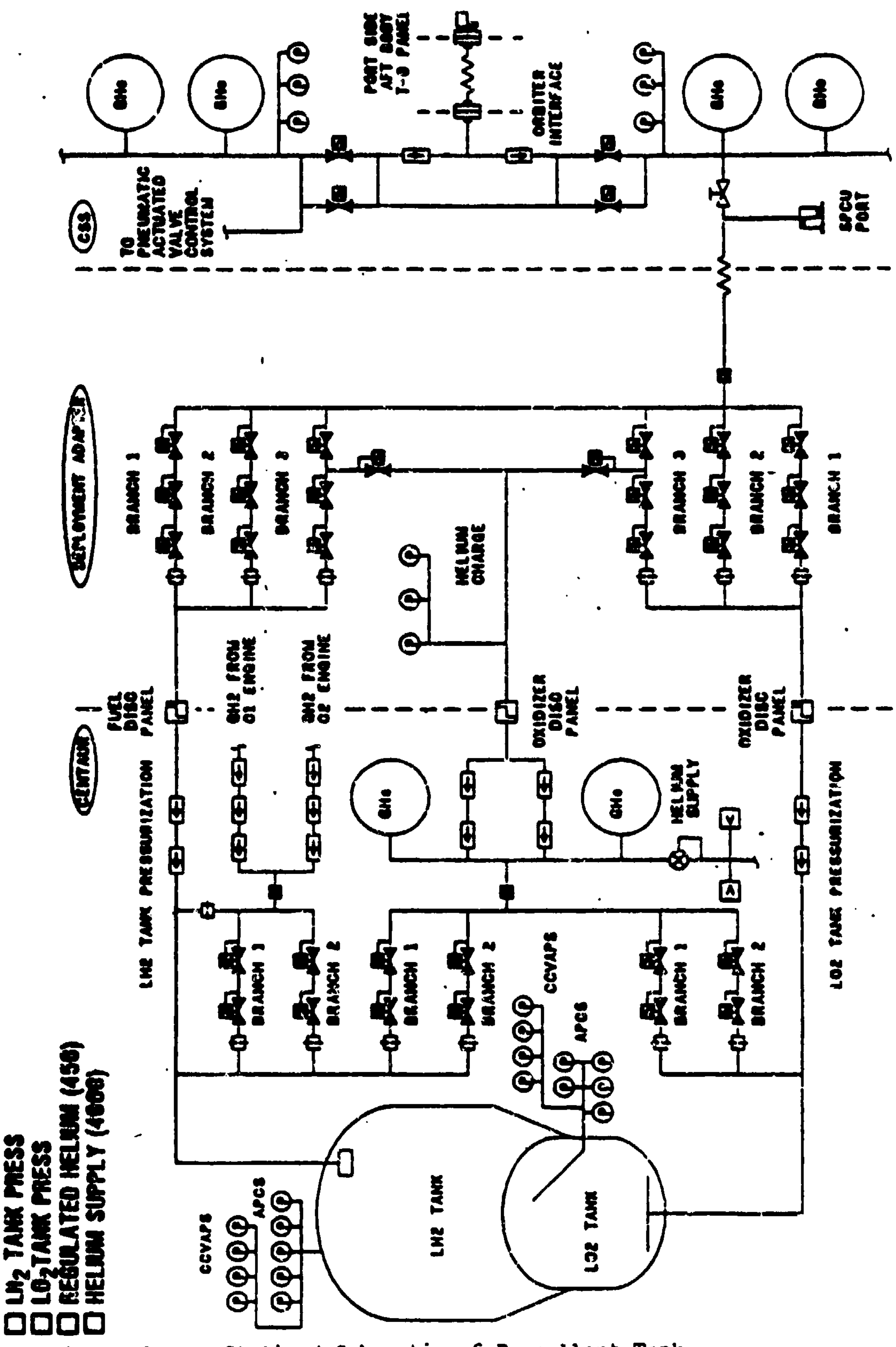


Figure C9-11 Stylized Schematic of Propellant Tank Pressurization System

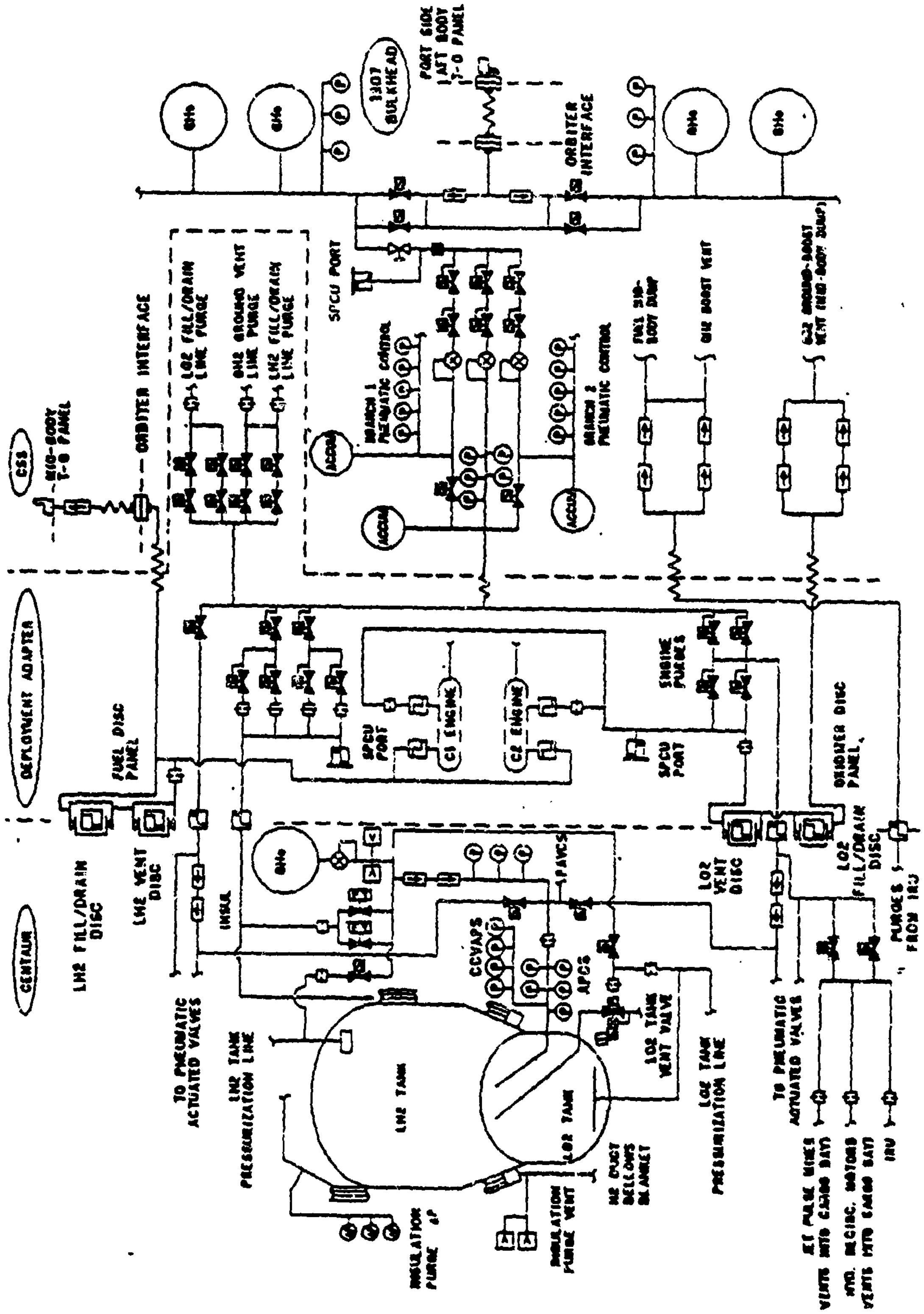


Figure C9-12 Stylized Schematic of Purge System

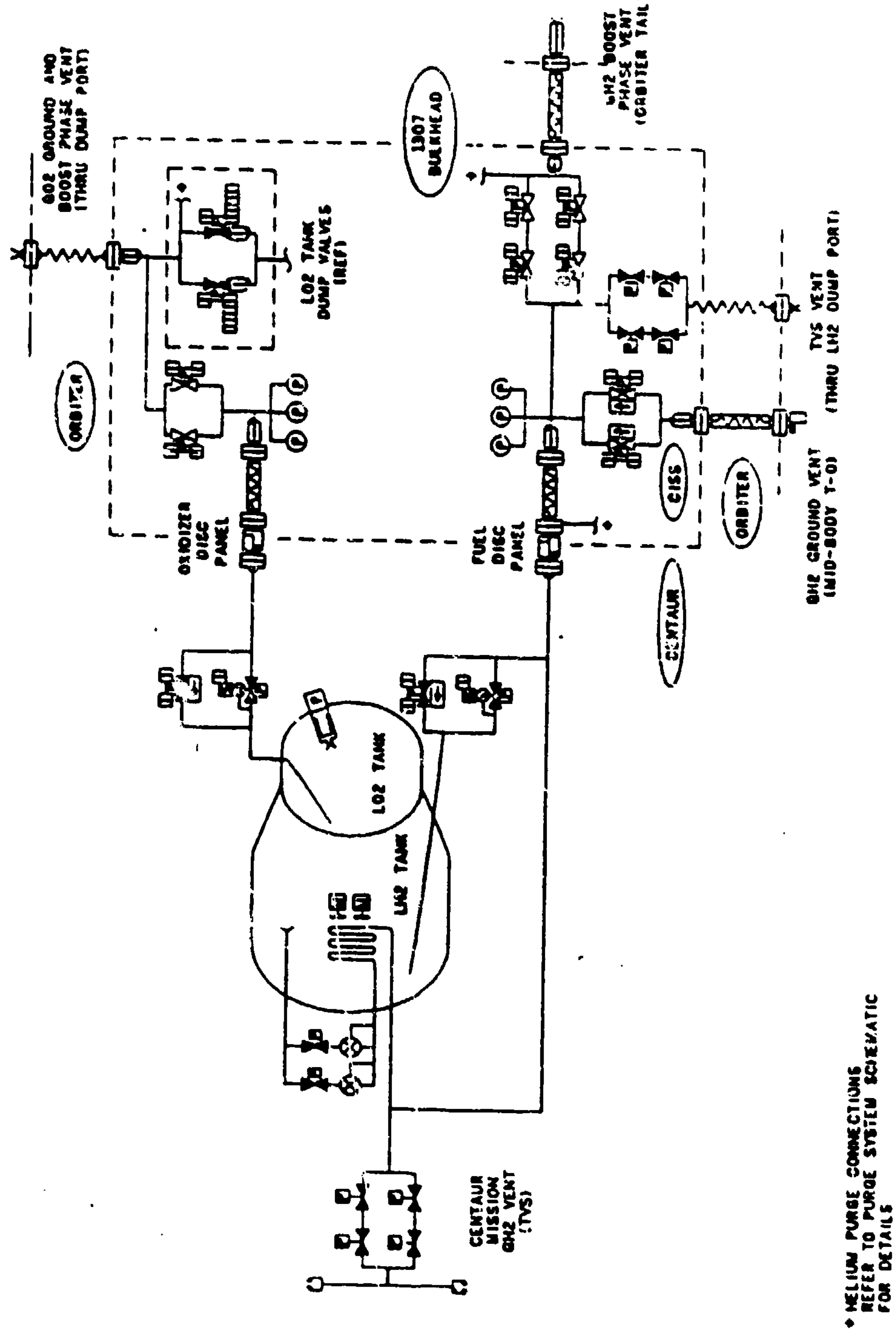


Figure C9-13 Stylized Schematic of Propellant Tank Vent System

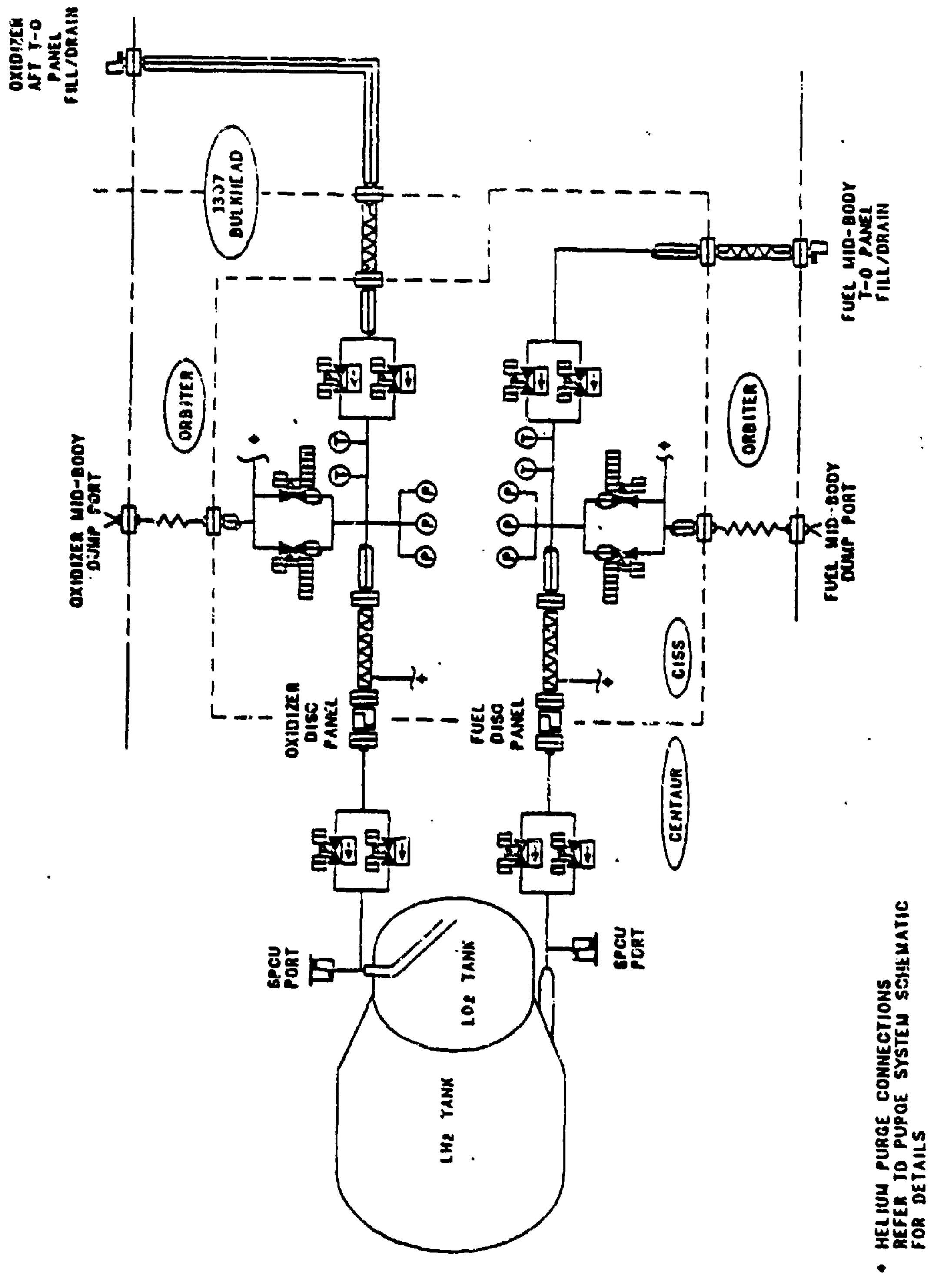


Figure C9-14 Stylized Schematic of Propellant Tank Fill/Drain and Dump Systems

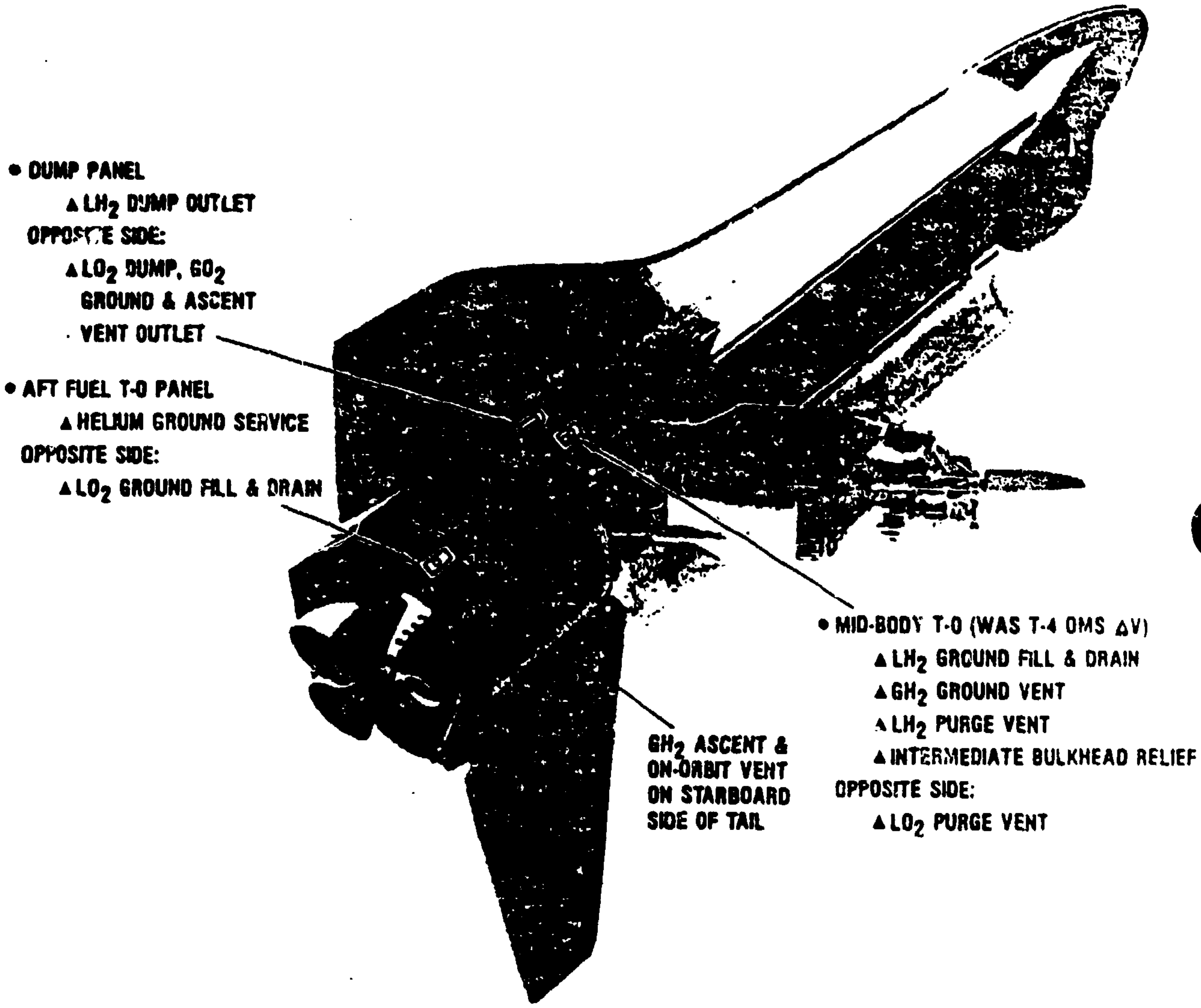


Figure C9-15 Centaur G-Prime Fluid Interfaces with Orbiter

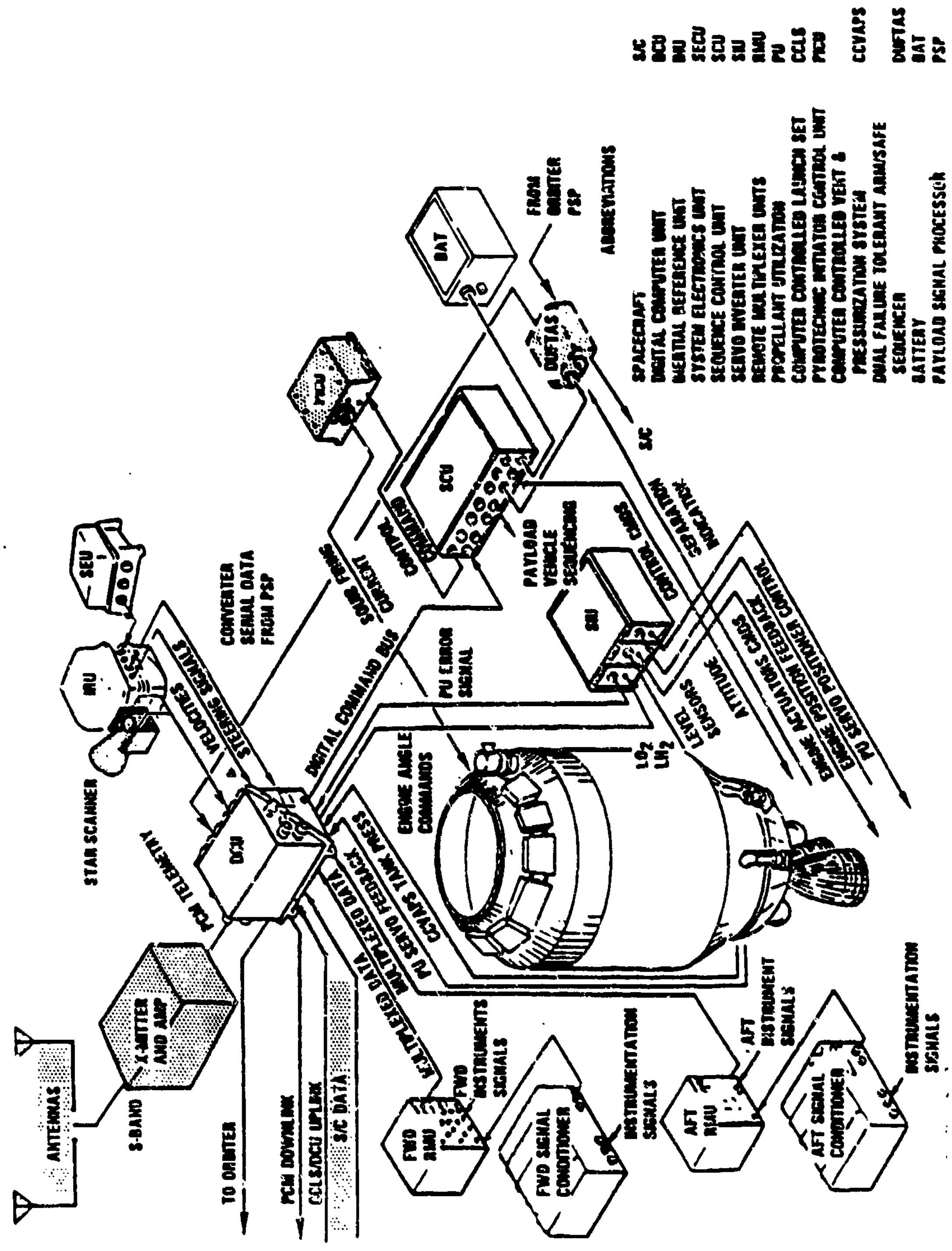


Figure C9-16 Shuttle/Centaur Avionics Systems

ABBREVIATIONS

DIGITAL COMPUTER UNIT	DCU
CONTROL UNIT	CU
CONTROL DISTRIBUTION UNIT	CDU
REMOTE MULTIPLEXER UNIT	RMU
PROPELLANT LEVEL INDICATING UNIT	PLIU
ELECTRICAL DISTRIBUTION UNIT	EDU
BATTERY	BAT
PYROTECHNIC INITIATOR CONTROL UNIT	PICU

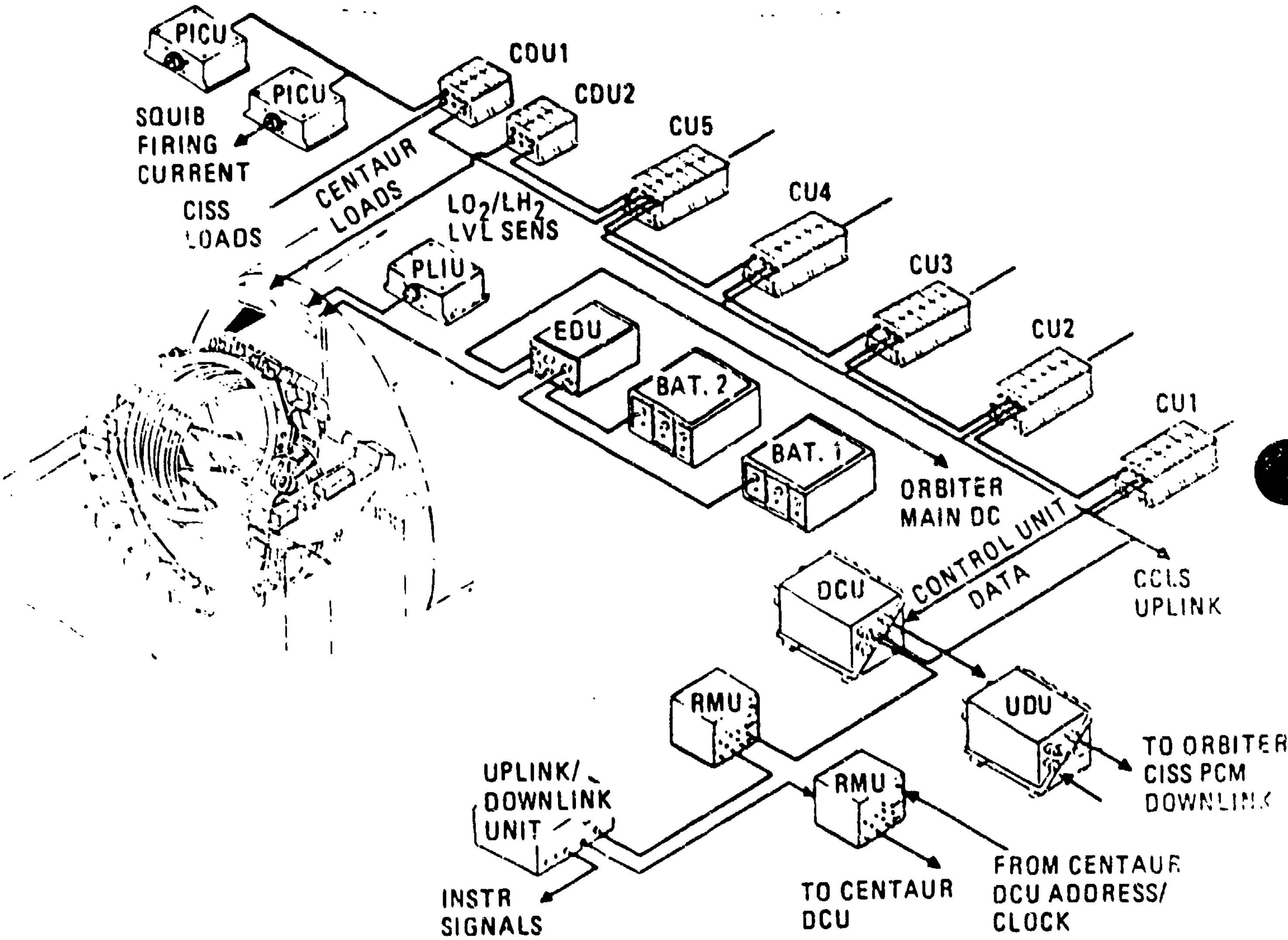


Figure C9-17 Shuttle/Centaur CISS Avionics System

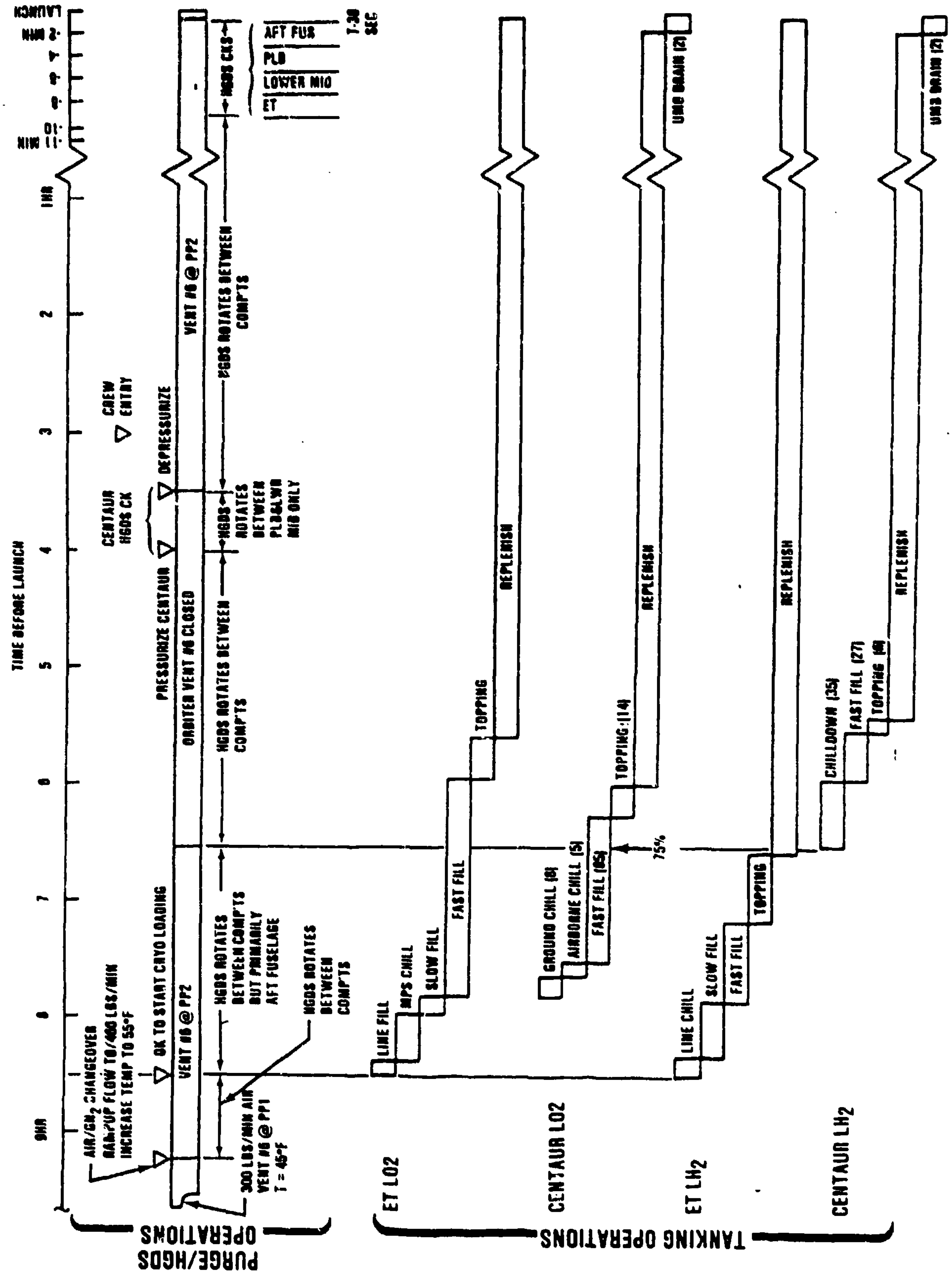


Figure C9-18 Prelaunch Controlled Environment and Tanking Timeline

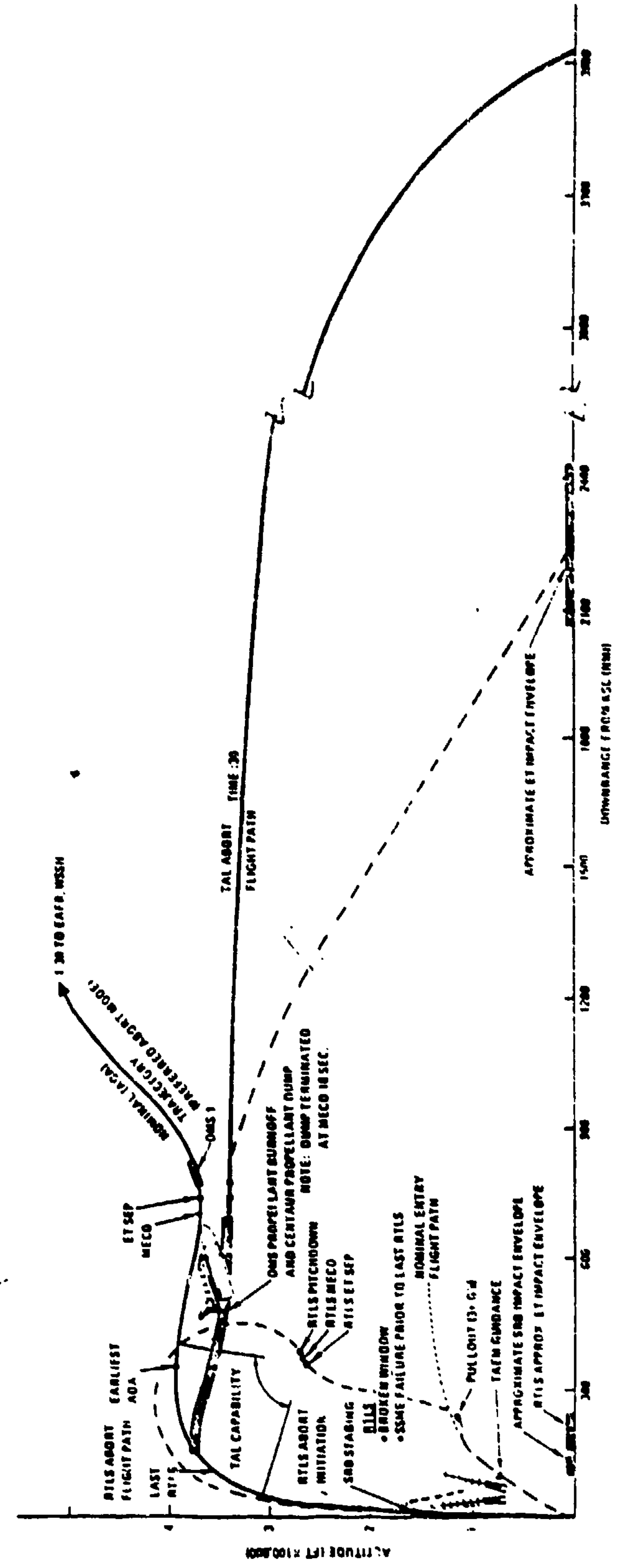


Figure C9-19 RTLS and TAL Aborts Altitude vs Range

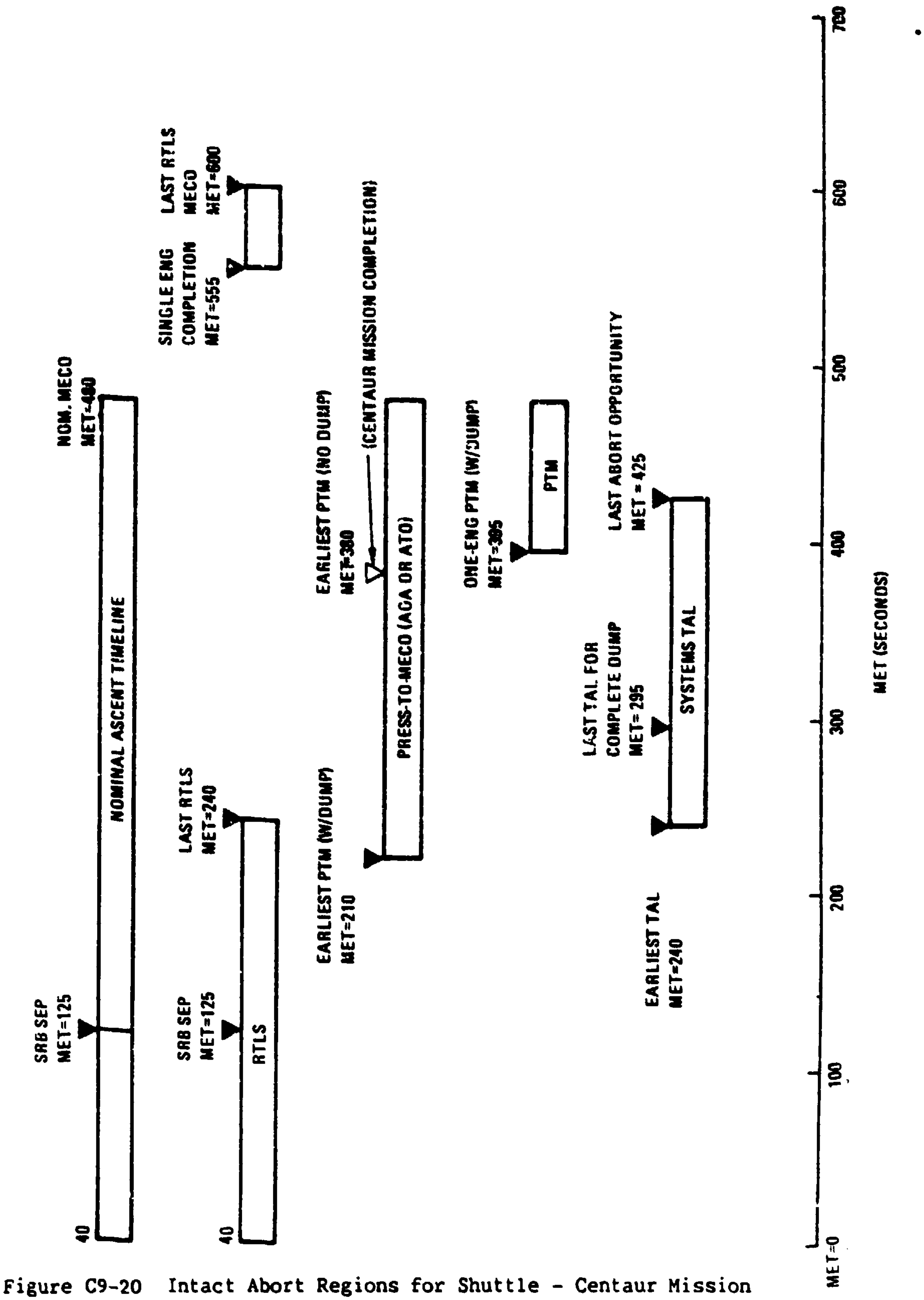


Figure C9-20 Intact Abort Regions for Shuttle - Centaur Mission

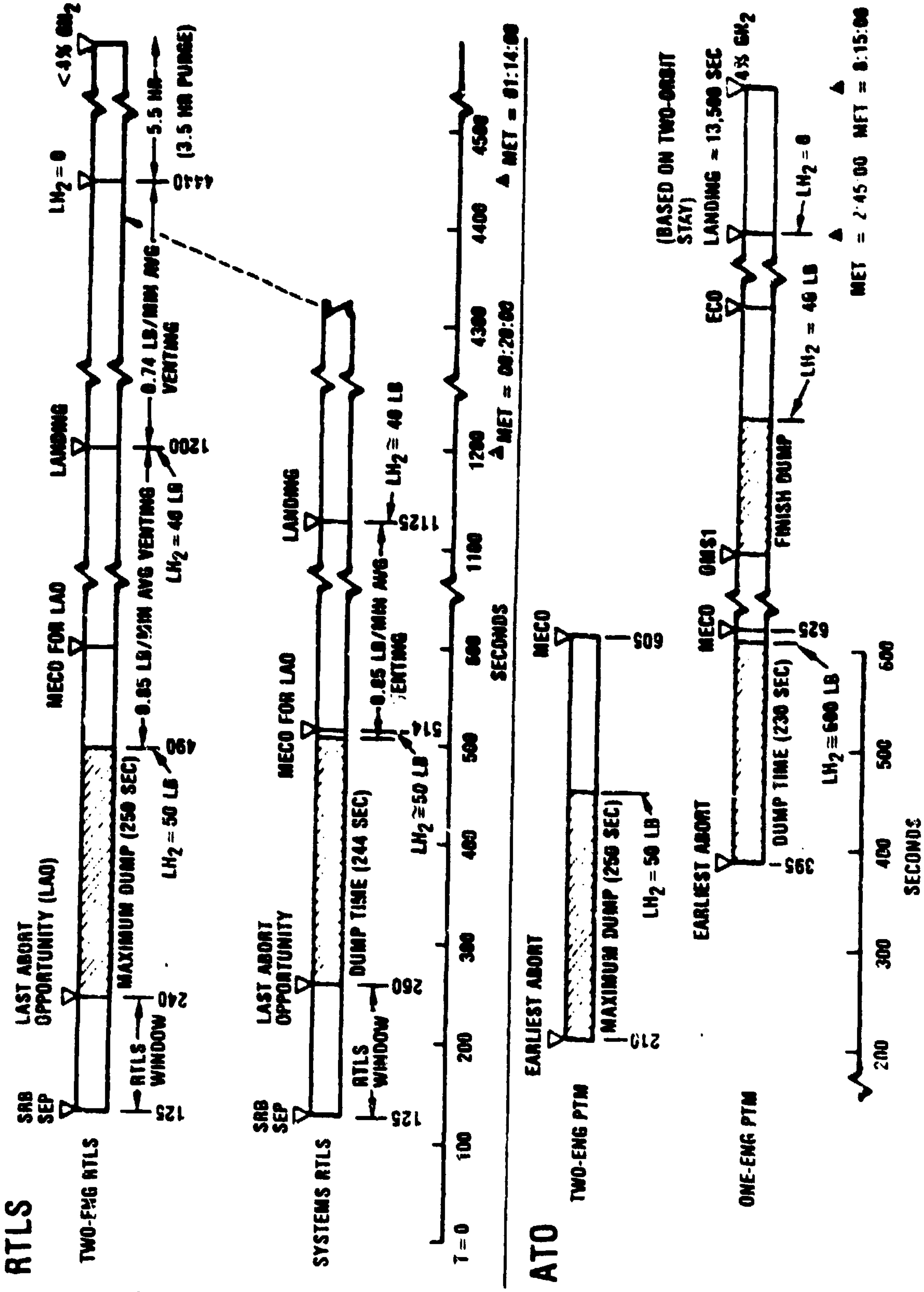


Figure C9-21 Centaur LH_2 Residuals vs Abort Scenarios (Preliminary)

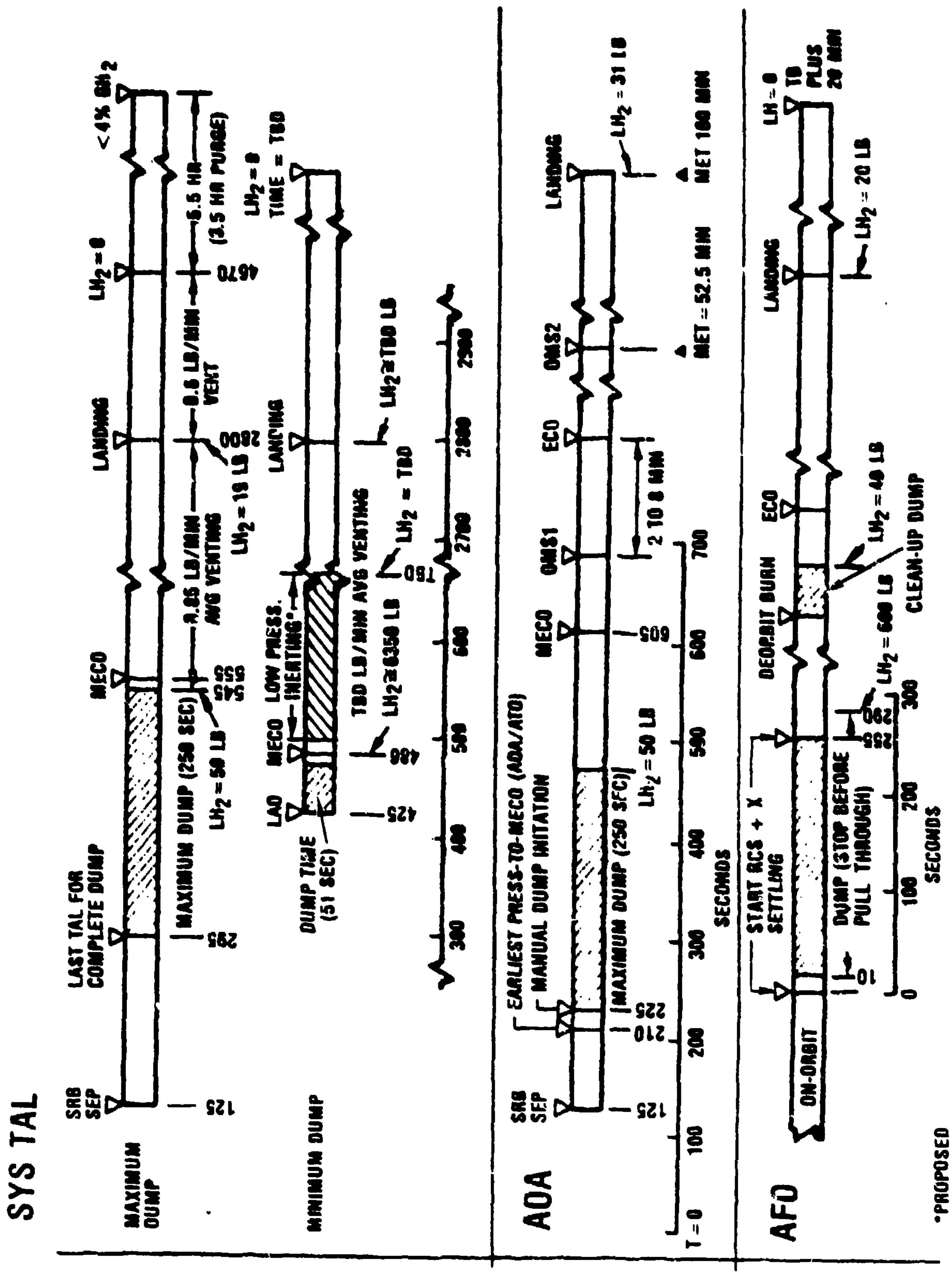


Figure C9-22 Centaur LH_2 Residuals Vs Abort Scenarios (Preliminary) - Continued

Appendix C10
Inertial Upper Stage

APPENDIX C10
INERTIAL UPPER STAGE

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APPENDIX C10
INERTIAL UPPER STAGE (IUS)

C10.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref No 016, (Attachment A, IUS ARAR, BAC Doc D290-10017-1).

C10.1 GENERAL DESCRIPTION

The Inertial Upper Stage (IUS) vehicle is to be an integral part of the Space Transportation System (STS), which is to serve the operational space requirements of the Department of Defense (DOD) and the National Aeronautics and Space Administration (NASA) in the 1980 decade and beyond. The IUS system will also be used in conjunction with the Air Force Titan 34D expendable launch vehicle.

The IUS baseline vehicle is a two-stage solid-propellant rocket upper stage. It consists of a propulsion module, an avionics module, airborne support equipment, and associated structures and mechanisms. Figure C10-1 shows the IUS Vehicle and airborne support equipment. Basic dimensions for the two-stage vehicle are shown in Figure C10-2.

C10.2 SYSTEMS DESCRIPTION, HAZARDOUS MATERIALS, SCHEMATICS

C10.2.1 Propulsion Module

The propulsion module consists of two solid rocket motors (SRMs), a thrust vector control (TVC) subsystem, and a reaction control subsystem (RCS). The SRMs contain a maximum of 27,400 lb of Class II propellant in the two-stage configuration. The two SRM combinations currently planned are shown in Figure C10-2. The baseline IUS vehicle for STS incorporates an extendible exit cone (EEC) on the second stage. The EEC is not included on the baseline vehicle for Titan, but it may be added.

C10.2.1.1 Propulsion System - A vectoring nozzle provides for directional control during motor burn. For greater adaptability to various missions and payload weights, the motor is designed for a variable propellant loading range of 50-100% of maximum capacity. The motor propellant is cast hydroxy terminated polybutadiene (14%), aluminum (18%) and ammounium perchlorate (68%). The case material is a filament-wound composite of epoxy-bonded Kevlar (aramid) fibers. The flame-side of the nozzle is fabricated from carbon/carbon-composite materials of an axisymmetric three-dimensional structure for the throat and entrance sections and a two-dimensional rosette structure for the exit cone.

The functional interfaces and overall dimensions of the rocket motors are shown in Figures C10-3 and C10-4. Coaxial cylindrical skirts are provided forward and aft for bolt attachment to IUS structure provisions are made for attachment of Thrust Vector Control (TVC) actuators at two places, 90° apart.

The Rocket Motor consists of the following: (a) loaded motor case, (b) nozzle and seal assembly, (c) TVC actuator bracket (2 each), (d) igniter (e) safe and arm device (2 each), (f) ignition ordnance train (2 each), (g) Extendible Exit Cone (EEC) on SRM-2 only.

Maximum and target weight values are given below for large motors with the two nozzle configurations and the full propellant loading. Weight given includes the actuator brackets and the ignition system (with safe and arm). The allowable weight for offloaded motors is reduced by the change in igniter and/or nozzle weights: (a) with the two-stage nozzle, the target weight is 22,607 lb and the maximum allowable is 22,757 lb (generic configuration is the 60-inch first-stage nozzle); (b) with the short nozzle, the target weight is 22,565 lb and the maximum allowable is 22,715 lb. Fully loaded, the small motor carries 6002 lb of propellant and the large motor 21,404 lb.

The propellant has a Class II explosive classification and will not ignite when conditioned to 275°F for a period of eight hours. Propellant physical properties and bond strength at the propellant/liner/insulation/case is adequate to withstand all stresses induced by motor firing and/or by specified environmental conditions at any time during the useful life.

The rocket motor is designed to sustain design limit loads without experiencing detrimental deformations or stresses in excess of the yield strength of the structural material. Design limit load is defined as the maximum load resulting from exposure to the acceleration, shock, and vibration environments and self-induced thermal and pressure loads. The unit is designed to sustain ultimate loads without failure. Ultimate load is defined as the limit load times the design ultimate factor of safety. For the attachments and associated structure, the ultimate factor of safety is 1.25 for free-flight load conditions remote from the orbiter, 1.4 for captive flight loads and when near the orbiter, and 1.5 for ground handling and transportation load. For thermal conditions, the minimum design margin above and below the expected operating temperatures is 11°C. In addition to the above factors of safety, a fitting factor of 1.15 is also applied.

All pressurized components except the nozzle throat and exit cone are subjected to a proof pressure equal to 1.05 times the maximum expected operating pressure of the component without experiencing detrimental deformation or stresses in excess of the yield strength of the structural material. Minimum burst (ultimate) pressure of 1.25 times the maximum expected operating pressure will be withstood without the failure or rupture. The nozzle throat and exit cone is designed for an ultimate factor of safety of 1.25 for the worst-case combination of pressure loads, thermal stresses, and TVC actuator loads, considering the reduction in shell strength to be expected from temperature and erosion effects. When packaged for shipment, the rocket motor and the ignition system will be undamaged by sawtooth shock pulses of 20 g terminal amplitude and 11 milliseconds duration, applied to the shipping container in any lateral direction or in an upward direction, with respect to the intended shipping attitude.

The motor (cylindrical shell) structure is designed to withstand the ultimate net section design loads shown below for bending moment, torsion, and transverse shear loads applied concurrently with either the compression or tension axial load.

1.	Bending moment	12.8×10^6 in-lb
2.	Torsion	6.1×10^6 in-lb
3.	Transverse shear	174.0×10^3 in-lb
4.	Axial compression	270.0×10^3 lb
5.	Axial tension	140.0×10^3 lb

The rocket motor nozzle is capable of angular deflection in any direction to an angle of four degrees for SRM-1 and seven degrees for SRM-2, over the full range of motor chamber pressure and with the least favorable combination of manufacturing tolerances. The nozzle will withstand, without damage, the worst-case combination of actuator force, gravitational load (one g environment, motor axis horizontal), and impact loading during ground checkout and predeployment tests of the thrust vector control system with no chamber pressure in the motor. Stall force developed by each actuator may be as high as 750 lb and actuator driving velocity as high as 7.9 inches per second. In the normal operation life of the motor, nozzle vectoring is expected to be equivalent to ten ground checkout duty cycles and one mission duty cycle.

A closure is fitted to the nozzle to seal the motor interior until ignition occurs. The closure will fail at ignition and leave the nozzle contour unrestricted. It is designed for a normal (internal minus external) pressure differential of 25 psi and a reverse (external minus internal) differential of 5 psi (limit load conditions).

The motor is fired by application of electrical firing impulses to redundant safe and arm devices after electrical arming of the devices has been accomplished. Propagation of energy via explosive trains to the igniter results in initiation of propellant combustion.

The ignition system is illustrated schematically in Figure C10-5. The ordnance trains provide redundant energy transfer channels from the electro-explosive initiators in the safe and arm to the igniter. For each SRM, they consist of: (a) through-bulkhead initiators (two required), (b) shielded mild detonating cord assemblies (two required), (c) explosive transfer manifolds (two required), and (d) confined detonating fuse assemblies (4 required). Ordnance train components shall not autoignite when subjected to a temperature of +250°F minimum for two hours.

The through-bulkhead initiator (TBI) design provides duration transfer through a bulkhead that is integral with the TBI body. Explosive material is packed intimately against both sides of this bulkhead in a manner that will ensure detonation propagation through the bulkhead but not adversely affect the integrity of the bulkhead. The TBIs are designed for installation in ports in the base of the igniter with provisions for sealing against the chamber pressure. The output section of the TBI is also hermetically sealed. When tested, the leakage will not exceed 1 x 10^{-6} cc/sec of H_e at 1 atm.

Each shielded mild detonating cord (SMDC) assembly includes a detonating cord completely enclosed in stainless steel tubing, with end fittings for attachment to the mating components. The confined detonating fuse (CDF) is similar to SMDC, but it is flexible. Tip booster charges will ensure reliable propagation across component interfaces.

The Explosive Transfer Manifold (ETM) has a Y or T junction where the dual SMDCs from each Safe and Arm converge into a single SMDC, which continues to one TBI. It has physical interfaces for attachment of the SMDC end fittings and a means for mounting the ETM to vehicle structure.

All small solid rocket motors used on Orbiter launched missions employ an extendible exit cone (EEC) to increase the effective nozzle length, thereby improving thrust. The EEC is deployed just before motor ignition by the EEC deployment mechanism. The preliminary concept of the EEC deployment mechanism (Fig. C10-6) uses four furlable thin-foil extension booms which extend the telescoping nozzle exit cones approximately 40 inches aft to the latched-up position, then rotate away to provide clearance for nozzle vectoring. The deployment mechanism is comprised of four boom extension units equally spaced around the motor nozzle, a motor drive unit, and a cable system. Each extension boom is fabricated by welding two preformed thin-foil sections together to form a somewhat circular cross-section with flanges on opposite sides as shown in Fig. C10-6. After heat treatment and flange preparation, the boom is flattened and coiled on a storage hub in the boom extension unit. Recessed in the boom end is an adapter receptacle that engages the aft exit cone interface fitting. At EEC deployment, the coiled boom is drawn from the storage drum and driven out by two sprocket wheels. As the boom straightens, it assumes a circular shape and becomes rigid. Sprocket wheel torque is provided by a cable drum keyed to the sprocket shaft. The drum in each extension unit is prewound with approximately 90 inches of aircraft cable with the free ends attached to cable take-up sheaves in the motor drive unit. Rotation of the drive unit applies equal tension to all four sprocket drum cables, thereby providing synchronous boom extension. The drive unit is comprised of dual motors on a common shaft, drive reduction gears, and the drive sheaves. The EEC dual drive motors are powered by squib-activated thermal batteries.

C10.2.1.2 Reaction Control System (RCS) - The RCS is a monopropellant blowdown system that uses hydrazine as the propellant. It is housed in the avionics bay of the IUS, surrounding the upper dome of the second-stage SRM. The RCS contains multiple positive expulsion near-spherical propellant tanks and multiple thrusters with dual series inlet valves and 80° nozzles. The feed system is manifolded with sufficient filtering to provide the required propellant to each thruster inlet. The thrusters are arranged in pairs as Reaction Engine Modules (REM). The RCS consists of one to three titanium tanks and associated control valves and piping to the thrusters. Each tank contains 122 lb (nominal) of hydrazine, pressurized by 380 psig GN_2 over an expulsion bladder. Each tank is isolated by an ordnance-actuated valve from the feed system manifold piping. RCS isolation valves are activated after deployment from the orbiter or after separation from the T-34D. RCS thruster operation is controlled by the IUS computer.

The Reaction Control System operational requirements include orienting the vehicle before solid rocket motor firings, roll control during the motor firings, vernier corrections for motor impulse variations, attitude control and maneuvering during transfer orbit coast period, payload spinup if required, and attitude control and maneuvering after payload separation.

The RCS is shown schematically in Figure C10-7. A modular tankage approach was selected to maximize flexibility for altering propellant capacity. Each tank is self-contained and pre-serviced before IUS installation in the Orbiter and requires no Orbiter interfaces. To minimize leakage potential, propellant is isolated in each tank until IUS deployment and system activation. Redundant pitch-and-yaw thrusters face aft to preclude spacecraft plume impingement. Roll thrusters operate in pure-couple pairs and are located at the same tangential location as the yaw thrusters. The tanks are initially pressurized to 380 psig with an 85% nitrogen/15% helium gas mixture. Isothermal propellant explusion results in 100 psig at depletion. The system is designed for a burst pressure of 1720 psig and provides a design factor of four to permit normal personnel operations arcund the preserviced tanks.

The RCS Power Distribution Schematic is shown in Figure C10-8. The REM valves and isolation valve squibs receive power from the utility batteries, and the REM, tank, and line heaters are powered by the avionics battery. The power levels will be 32 Vdc initially, decaying to approximately 24 Vdc at the end of the mission.

A three-view drawing of the Hamilton Standard IUS REM shown in Figure C10-9 depicts the two thrusters mounted side-by-side, each with 80° nozzles and series redundant Valcor solenoid valves. A single Dynatube inlet fitting provides propellant to both thrusters. The key features of the IUS thruster designs are simplicity and low-cost fabrication compatible with the IUS life and duty cycle requirements. Typical of the approach being taken is the configuration of the REM. Six REMs, each with a pair of thrusters, are used on each IUS. A single design was made possible by specifying a near-right-angle nozzle. Figure C10-9 illustrates how a pitch module can be converted to a roll/yaw module by merely clocking one of the thrusters 90°. No changes in either mechanical or fluid interfaces are required.

The IUS mission times and thruster duty cycles allow the use of warming pulses to maintain minimum propellant valve and catalyst bed temperatures during the mission without prohibitive expenditure of propellant. For the 12 thrusters on IUS and 10-hour thermal-time requirements, the propellant valves can be maintained above 40°F with about 3 lb of propellant. The warming pulse concept has been selected over individual catalyst bed and valve electric heaters. Low wattage heaters will be installed on the REMs to keep propellant passage temperatures above freezing during the period prior to RCS activation. These heaters will be on for early Orbital operations, particularly during the time that the Orbiter payload doors are open and the IUS is still in the payload bay. At RCS activation after deployment, the REM heaters will be turned off and will not be re-energized.

The RCS is the only pressurized element of the IUS vehicle. This system consists of Reaction Engine Modules (REMs) connected by redundant valves to a distribution manifold. The manifold is connected to propellant tank assemblies (PTAs), separated from the manifold by explosive-actuated isolation valves. The manifold is pressurized to 25 psig to prevent intrusion of external gases. The RCS design accommodates either two or three 21-inch-diameter spherical tanks with a usable capacity of approximately 122 lb of propellant each.

The installation approach is illustrated in Figure C10-10. Tank material is 6Al-4V titanium, and the elastromeric positive-explusion diaphragm is AF-E-332. Mounting is on trunnions at the poles of each hemisphere, with gas and liquid ports located in the trunnions.

Tank modules consisting of tank, GN_2 and N_2H_4 fill valves, isolation valve, and related plumbing are welded and tested as an assembly. The tank modules are installed in the IUS by connecting the mechanical fitting to the feed manifold and bolting the tank module to the trunnion mounts.

The feed system as shown in the RCS Schematic, Figure C10-7, consists of service valves, filters, and the hydrazine manifold. The propellant tank service valves are used only on the ground to prefill the tanks with hydrazine and pressurant. The valves are closed and capped after the servicing operation. The isolation valves keep the manifold dry until the IUS is deployed from the Orbiter payload bay, thus minimizing leakage. All fittings upstream of the isolation valves are welded. Mechanical fittings downstream of the isolation valves permit tank installation, removal, and replacement. System filters are in-line units installed in the manifold downstream of the isolation valves. A manifold fill and drain valve is provided to allow pressurizing the manifold for connector leak tests and REM valve checkout. The circular manifold and the lines to the tanks are 1/2-inch-O.D. 304L stainless steel tubing. The REM legs are 3/8-inch-O.D. 304L tubing.

Thermal control provisions consist of propellant tank heaters and insulation blankets, propellant line heaters and insulation blankets, REM heaters and REM "Warm" pulsing. The tank and line heaters are low-wattage redundant element units that are energized for the entire mission. The REM heaters are provided to keep the REM propellant passages above the freezing point of hydrazine prior to deployment, particularly while the Space Shuttle payload bay doors are open. After the RCS is activated, the thrusters will be pulsed periodically to maintain valve temperatures above freezing. Table C10-1 summarizes the electrical interface data.

C10.2.2 Avionics Module

The avionics module includes the following subsystems: computer, antenna, power, and command and control. The avionics module is connected through umbilicals to the Airborne Support Equipment (ASE) and to each spacecraft (S/C).

Redundant programmable computers support guidance, navigation, data management, and housekeeping functions. The computer monitors pertinent equipment for status, which is used in housekeeping functions and/or transmitted to the Orbiter and/or ground control.

Antennas are located on opposite sides of the IUS. The antenna subsystem is composed of a transponder, power amplifier, RF switch, diplexer, and the antennas.

There are four sources of power for the IUS from Orbiter liftoff until safe separation distance has been achieved; namely, the avionics battery set, the utility battery set, ASE batteries, and orbiter power. The avionics batteries are used primarily as a source of power for electronics. This is unregulated; however, a dc-dc converter is available as a kit if required by the S/C. The utility batteries provide a separate power source for TVC motors, RCS valves, motor-driven switches, and all vehicle ordnance devices except for the destruct charge on Titan missions. This reduces the probability of conducted interference to the electronics. AS atteries and orbiter power are external to the IUS vehicle and provide power before deployment from the Orbiter.

The power subsystem provides for the distribution, control, and regulation of orbiter or ground power to the IUS vehicle/spacecraft combination. Further, this subsystem supplies battery power to supplement Orbiter power on a preplanned basis. The power subsystem consists of a power control unit, an orbiter power control panel, ASE cables, ASE batteries, and dc-to-dc converters. The ASE power control functional interface is shown in Figure C10-11.

Table C10-1 RCS Electrical Interface Data

REM Input Voltage Required	22.5-32 Vdc
Thruster Requirements	
Valve Pull-In	15 Vdc max
Valve Drop-Out	1 Vdc min
Max Power, per Thruster (28 Vdc)	72 Watts
Heater Power Allotments (28 Vdc)	
REM Heaters	3.2 Watts Each (Two per REM)
Tank Heaters	5.8 Watts Each (Two per Tank)
Line Heaters	2.7 Watts Each

IUS vehicle and spacecraft power is supplied by IUS batteries. The IUS power subsystem can also use electrical power supplied by the ASE batteries, Orbiter power or ground power. The power subsystem provides for switching and distribution of electrical power to the IUS vehicle and spacecraft. Dual buses ensure that no single failure can disable both A and B channels of avionics. For the two-stage vehicle, three batteries (two avionics, one spacecraft) are carried in the first stage; five batteries (two avionics, two utility, and one spacecraft) are carried in the second stage; and three batteries are mounted on the ASE. All relays and switches used on IUS are hermetically sealed.

The Power Distribution Unit (PDU) provides for electrical power transfer switching between ASE power and internal IUS battery power. Separate avionics and utility power buses are provided in each of the two PDUs for the redundant IUS avionics subsystems.

The Power Transfer Unit (PTU) provides the Stage 1 and Stage 2 S/C battery and ASE input power switching functions. An interface is provided for optional inclusion of a dc-to-dc converter if regulated power is required by the spacecraft. Current and voltage sensing of the S/C power bus is also provided.

The Pyro Switching Unit (PSU) provides power for all ordnance events except Flight Termination. Ordnance Bus A receives power from Utility Battery A and is controlled through a motor drive switch. Ordnance Bus B is similarly arranged. The PSU provides for ordnance safety interlocks, event unable, and firing of EED bridgewires to initiate desired ordnance events.

The major IUS software areas encompass the verification and validation simulator software (V&VSS), the mission operations segment (MOS) data load software and the MOS post-processing software. Errors in any of these software systems or their associated translators, compilers, loaders host various enable/disable flags; start and stop times; and process selection commands:

1. Plot parameter values

2. Plot bilevel parameter line-charts

3. Plot single-parameter distribution/histogram

4. Plot multiple distributions/histograms

5. Tabulate parameter values

6. Monitor for out-of-limits parameters

7. Record reduced parameter file

8. Perform SRM burn vehicle-weight calculations

9. Perform parameter change calculations

10. Perform velocity change loss calculations

11. Perform RCS burn calculations

12. Perform RCS burn time delay penalty calculations

C10.2.3 Airborne Support Equipment (ASE)

The IUS ASE is a subsegment of the IUS system segment. Specifically, the IUS ASE provides for support of the IUS vehicle (with attached spacecraft) while in the STS Orbiter payload bay, as shown in Figure C10-1. The IUS ASE consists of the structure, batteries, electronics, and cabling to support the IUS vehicle/spacecraft combination, to enable the deployment of the combined vehicle, to provide and/or distribute and control electrical power to the IUS vehicle and spacecraft, and to provide communication paths between the IUS vehicle and/or spacecraft and the Orbiter. A discussion of each of the subsystems follows.

The ASE for the IUS STS vehicle/spacecraft consists of a cradle assembly whose structural and mechanical components are used to hold, tilt, and release the joined IUS vehicle/spacecraft from the STS Orbiter payload bay, Figure C10-12. The major components of the cradle assembly are the forward frame and aft frame. The forward frame engages the IUS trunnions with payload retention latch assemblies. The forward frame in turn is engaged by fittings on the Orbiter. The aft ASE frame provides the means of tilting the IUS to the checkout and deploy positions and releasing the IUS from the payload bay. The secured cradle assembly can withstand, without failure, the Orbiter-imposed ultimate environments of RCS loads, pressure, temperature, acoustics, shock, and vibration during all phases of shuttle operations, including emergency landing of the Orbiter. The X, Y, and Z deflections resulting from the various imposed loads are allowed to occur in certain attachments by the use of sliding surfaces and are restrained in other attachments. Stops are designed into the system, which stops limit travel so that the IUS system is contained within the geometrical design limits set forth in ICD-D-E0001. When tilted with the IUS, the cradle assembly can withstand Orbiter vernier RCS loads without failure.

C10.2.4 Structures and Mechanisms

Structures include Stage 1 and 2 assemblies, equipment support structure, interstage structure, and aft skirt structure. Mechanisms include staging equipment and mechanical elements, including ordnance initiated separation devices.

C10.2.5 T34D/IUS Flight Termination System

The T34D/IUS Flight Termination System (Fig. C10-13) is designed to destroy the pressure integrity of the IUS solid rocket motors in the event of inadvertent SRM ignition, premature stage separation, or issuance of a command destruct signal to the T34D by the Range Safety Officer. The major components of the FTS are the Titan Interface Unit (TIU), breakwires, utility and destruct batteries, Safe and Arm device, Explosive Transfer System train, and the motor case cutter (linear-shaped charge). Each stage of the IUS contains the required FTS hardware to operate independently of the other.

C10.3 MISSION SCENARIO

The IUS vehicle will be configured to support various DOD and NASA missions, which are primarily geosynchronous orbit insertion. The baseline vehicle is a two-stage system that supports the geosynchronous mission. With the exception of sizing the solid propellant stages and hydrazine quantities, the risks are similar for all vehicles exclusive of spacecraft. The sequence of events from launch complex arrival until a safe distance from the Orbiter is achieved following deployment is the same for all vehicles, except for a spacecraft integration. The IUS Mission Profile is shown in Figure C10-14, and Figure C10-15.

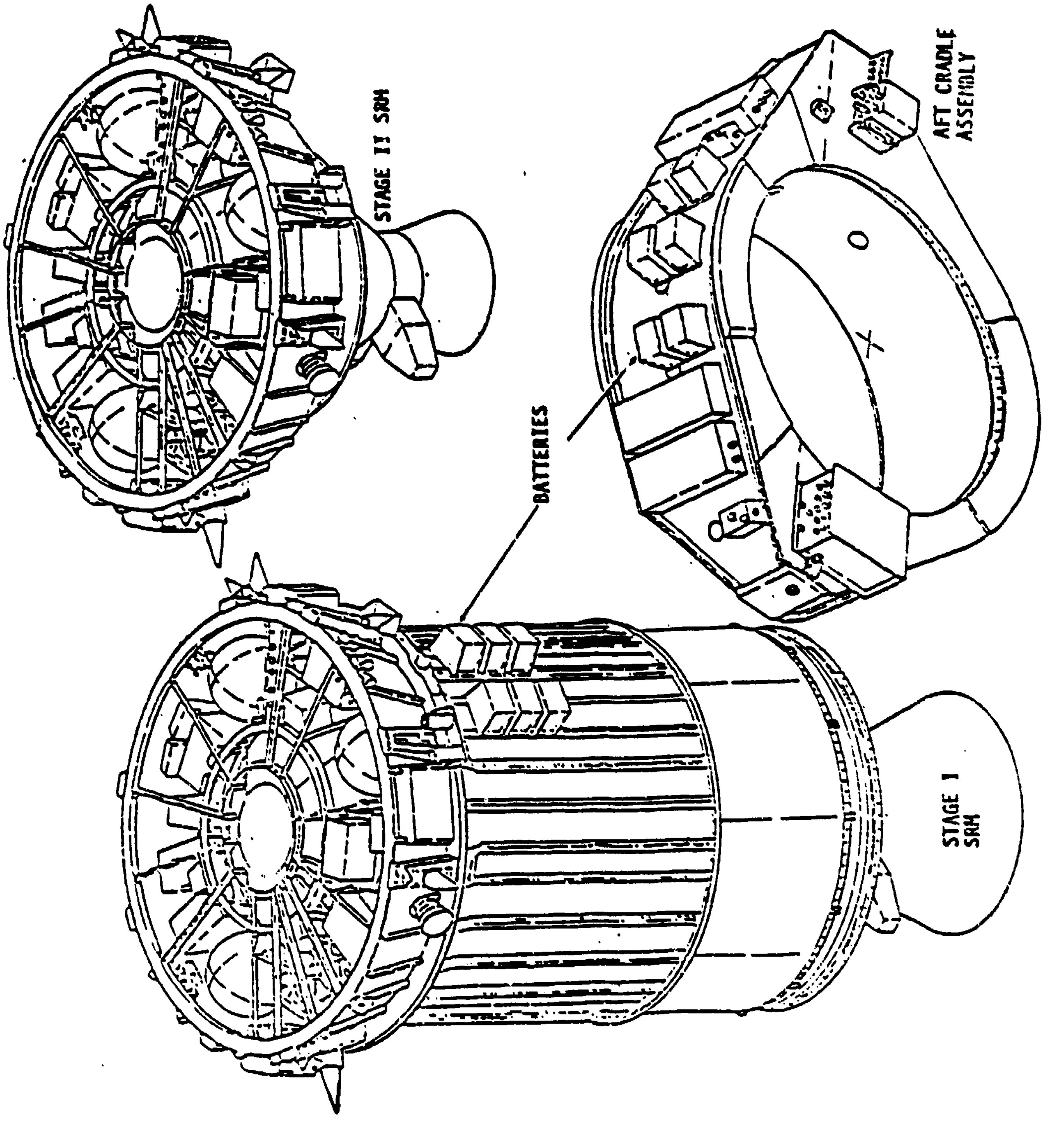


Figure C10-1 IUS Vehicle and ASE

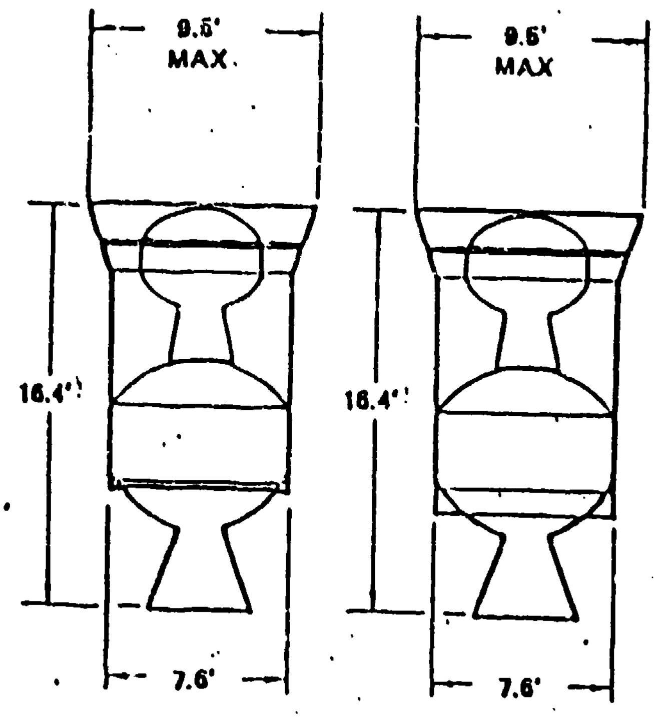


Figure C10-2 IUS Vehicle Configuration

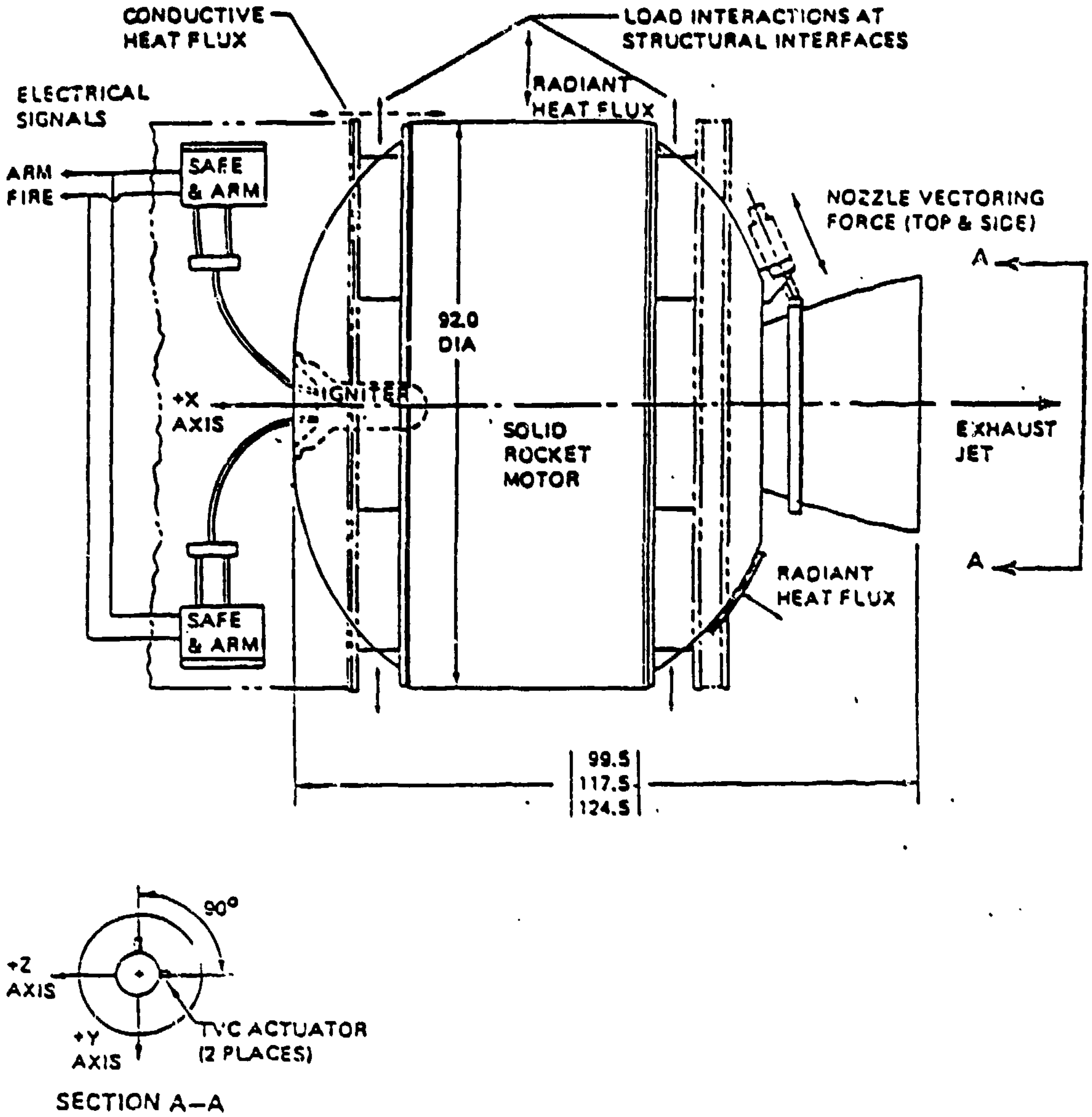


Figure C10-3 Functional Interface Diagram
(Large Rocket Motor)

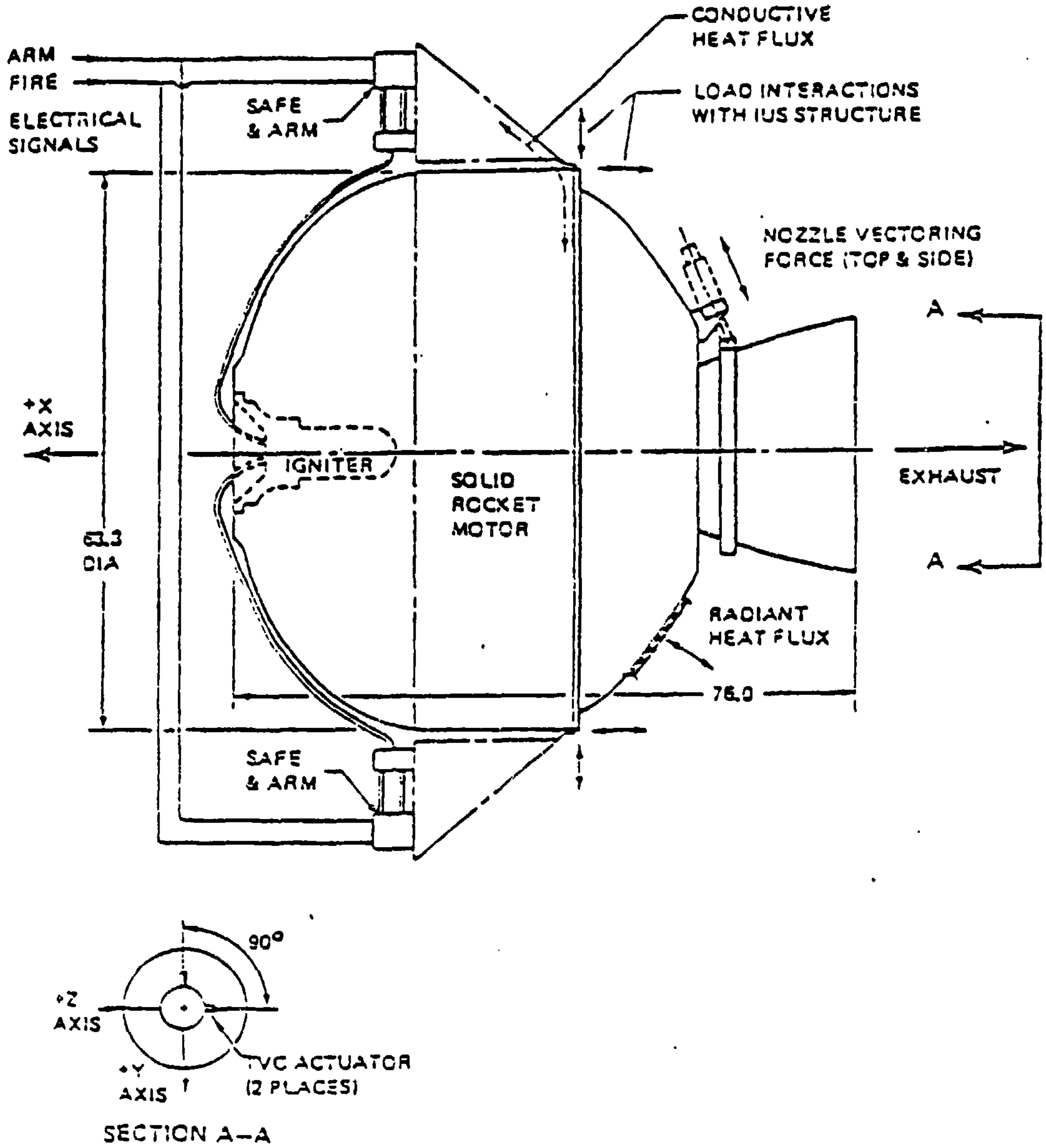


Figure C10-4 Functional Interface Diagram
(Small Rocket Motor)

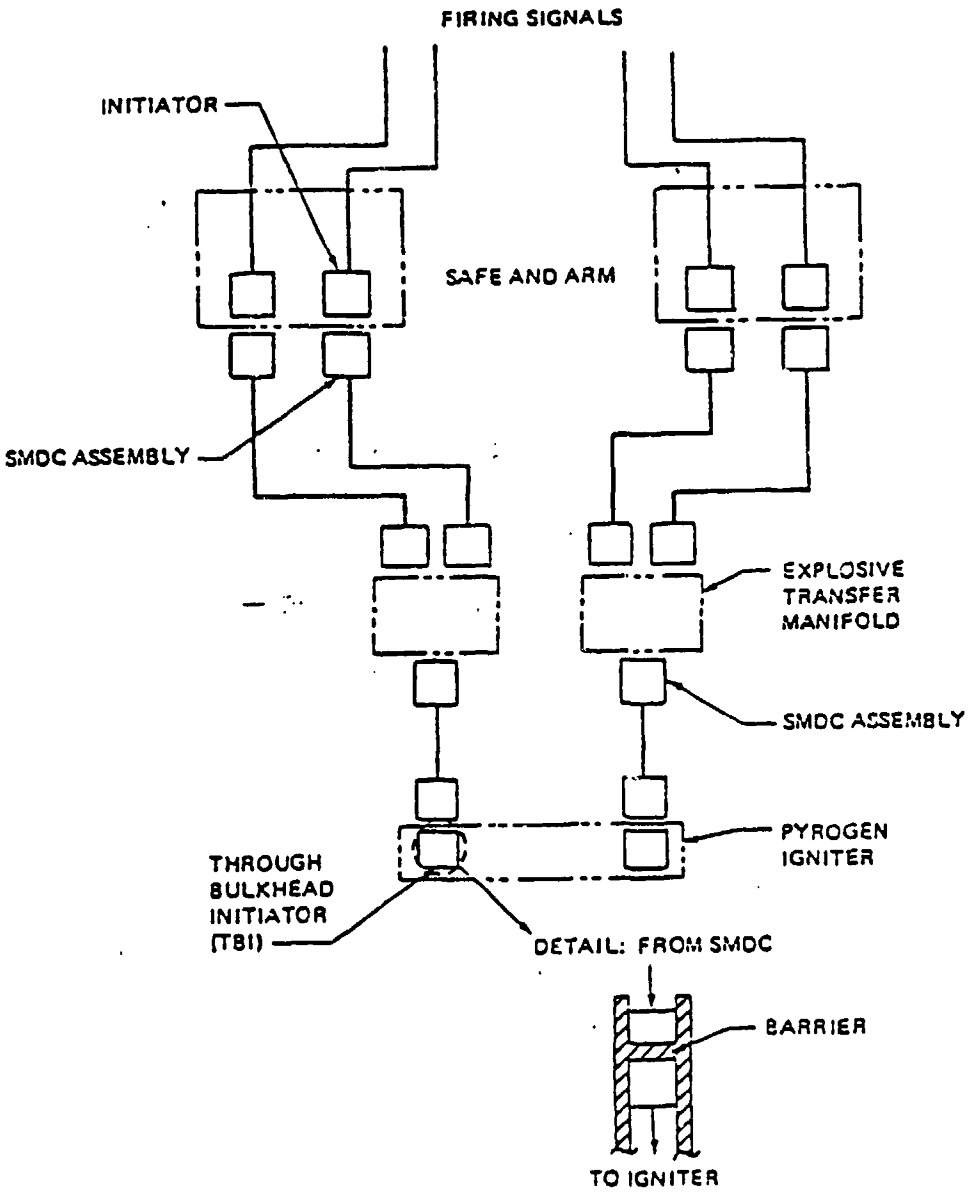


Figure C10-5 Ignition System Schematic

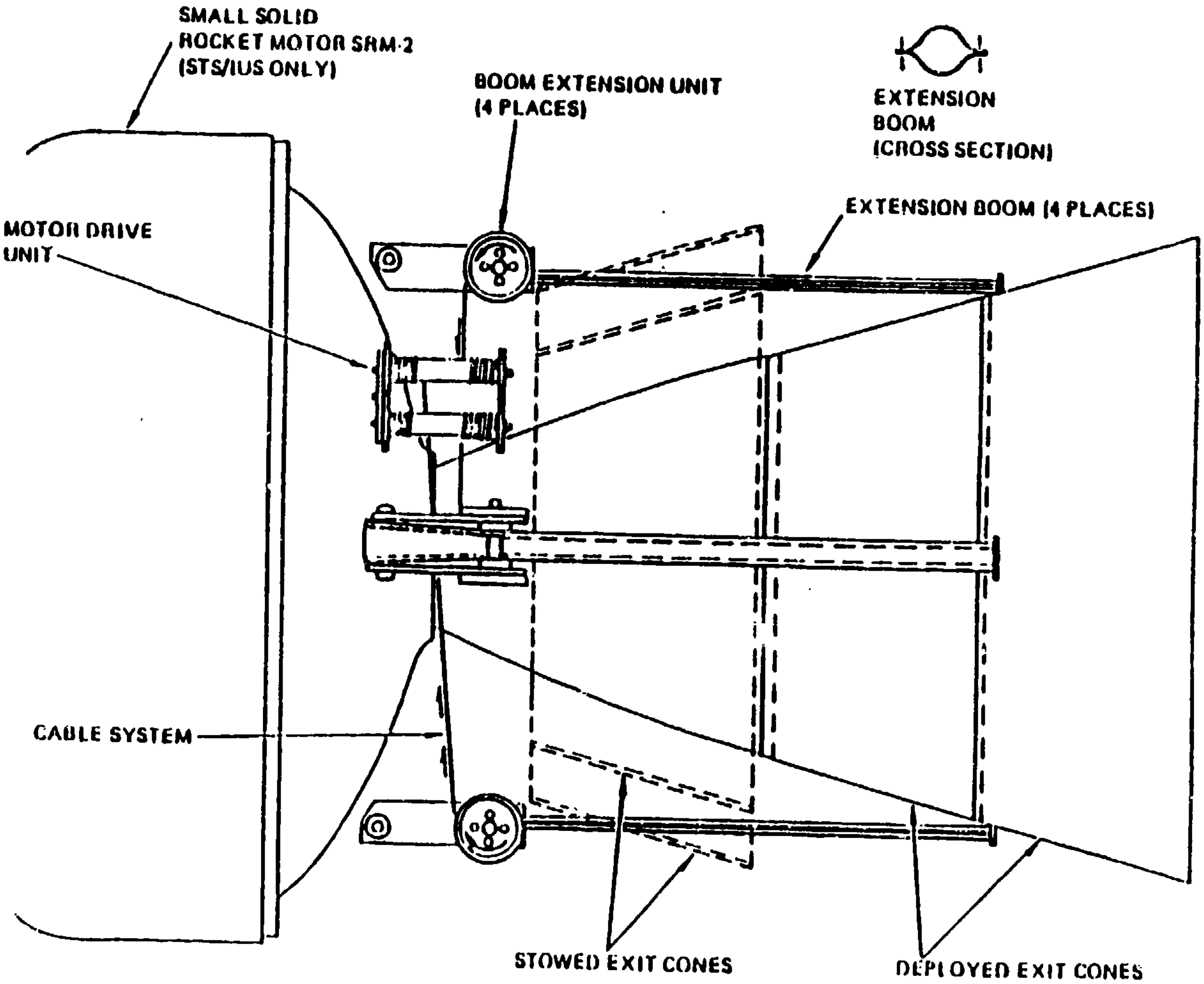


Figure C10-6 Extendible Exit Cone Deployment Mechanism

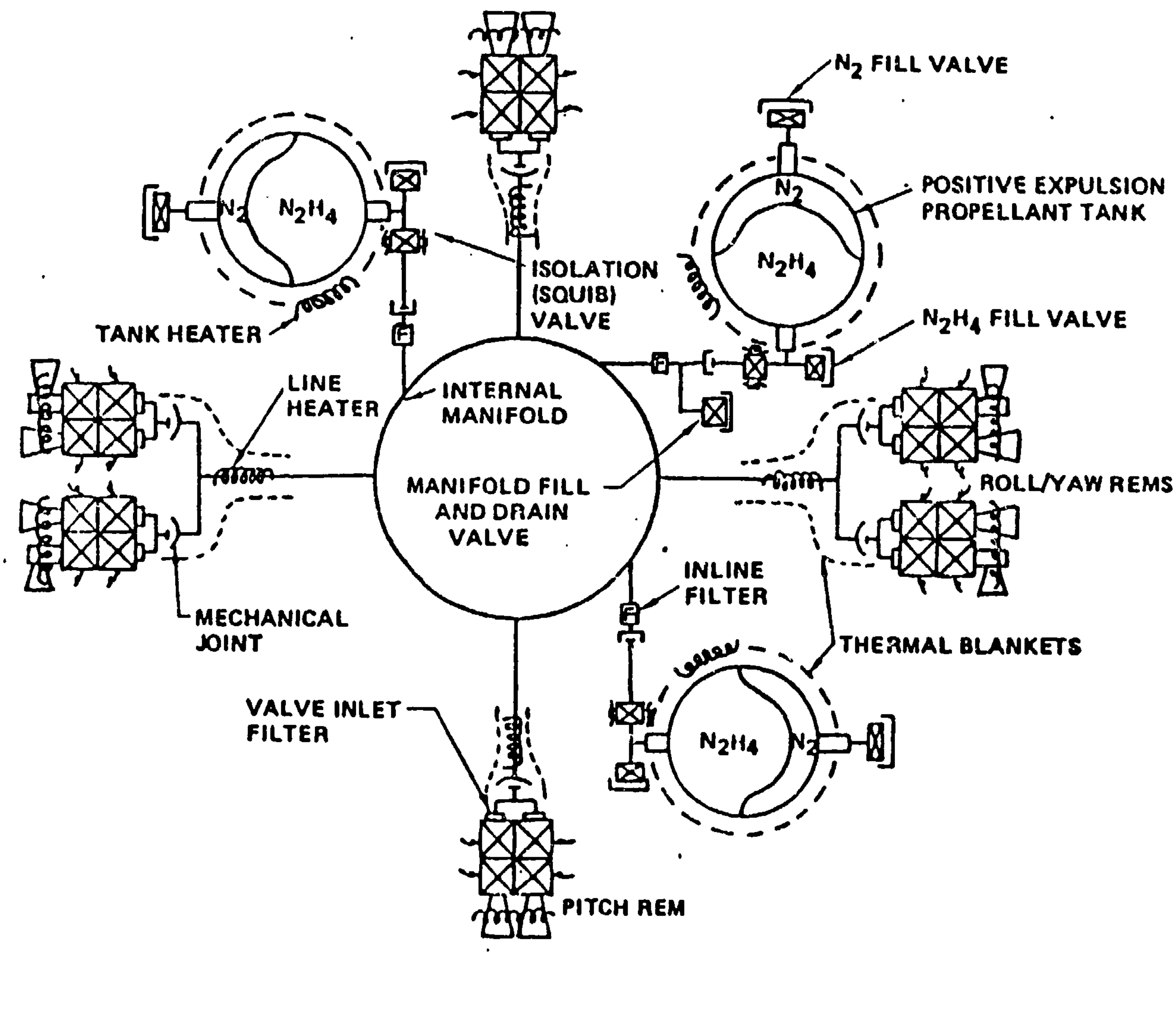
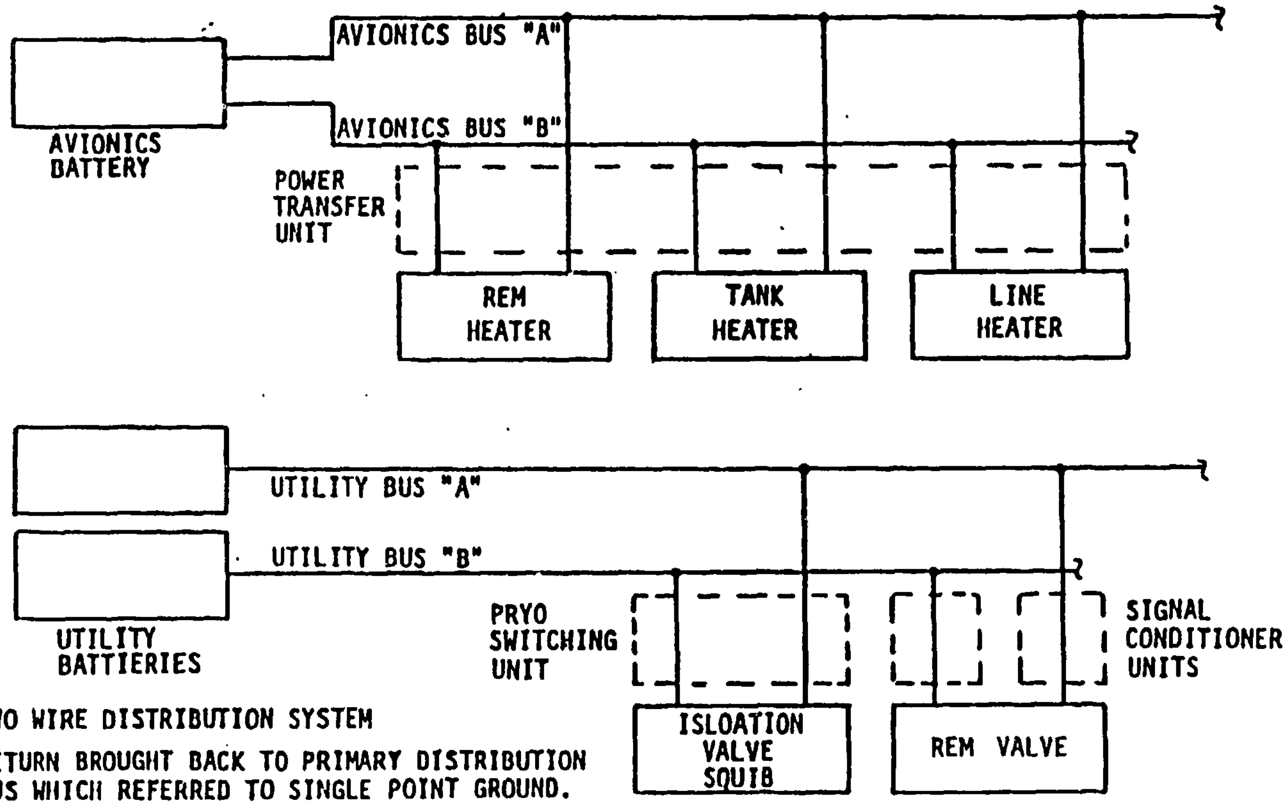


Figure C10-7 RCS Schematic Diagram



TWO WIRE DISTRIBUTION SYSTEM

RETURN BROUGHT BACK TO PRIMARY DISTRIBUTION BUS WHICH REFERRED TO SINGLE POINT GROUND.

SWITCH LEADS ARE TWISTED SHIELDED PAIRS FOR ORDNANCE AND REM VALVES

Figure C10-8 RCS Power Distribution Schematic

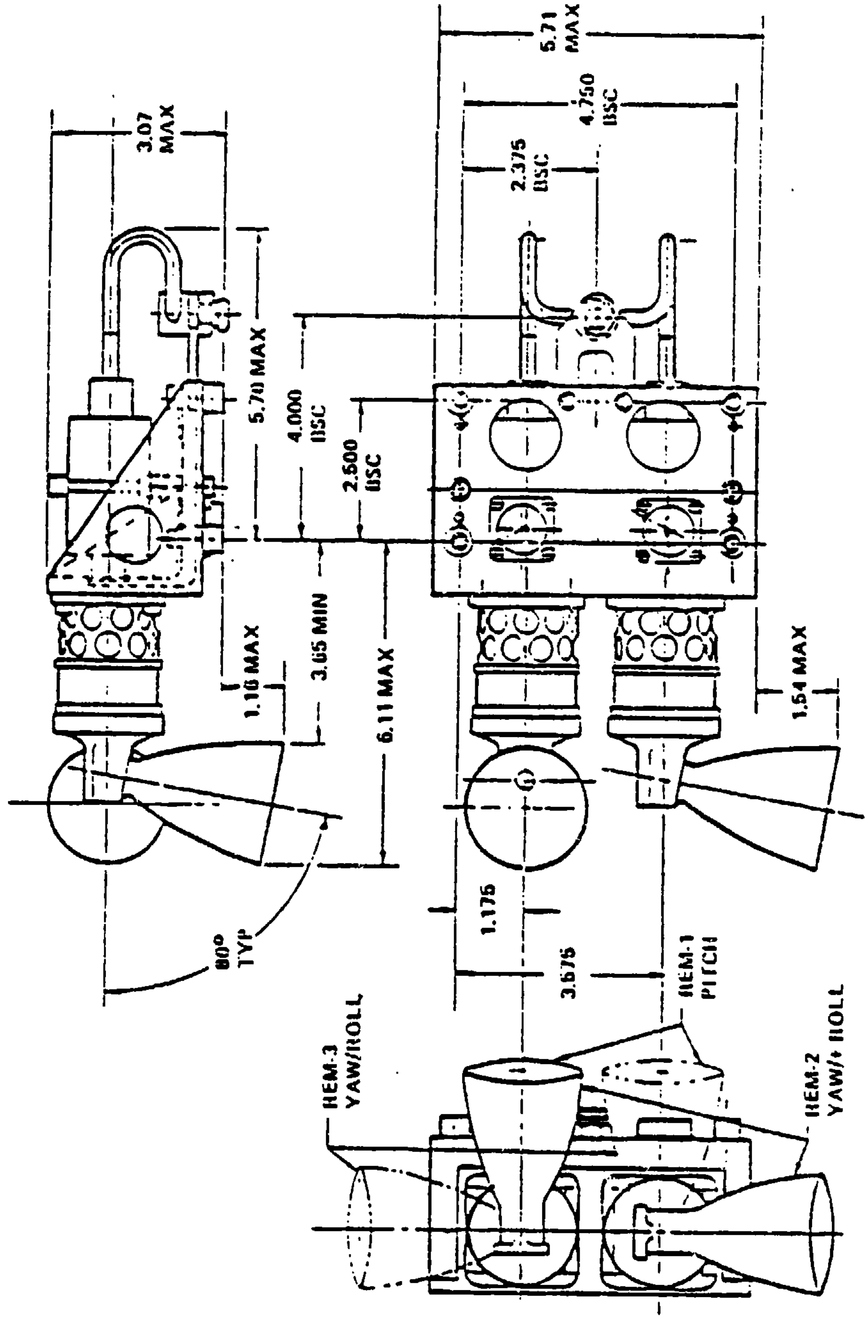


Figure C10-9 Reaction Engine Module

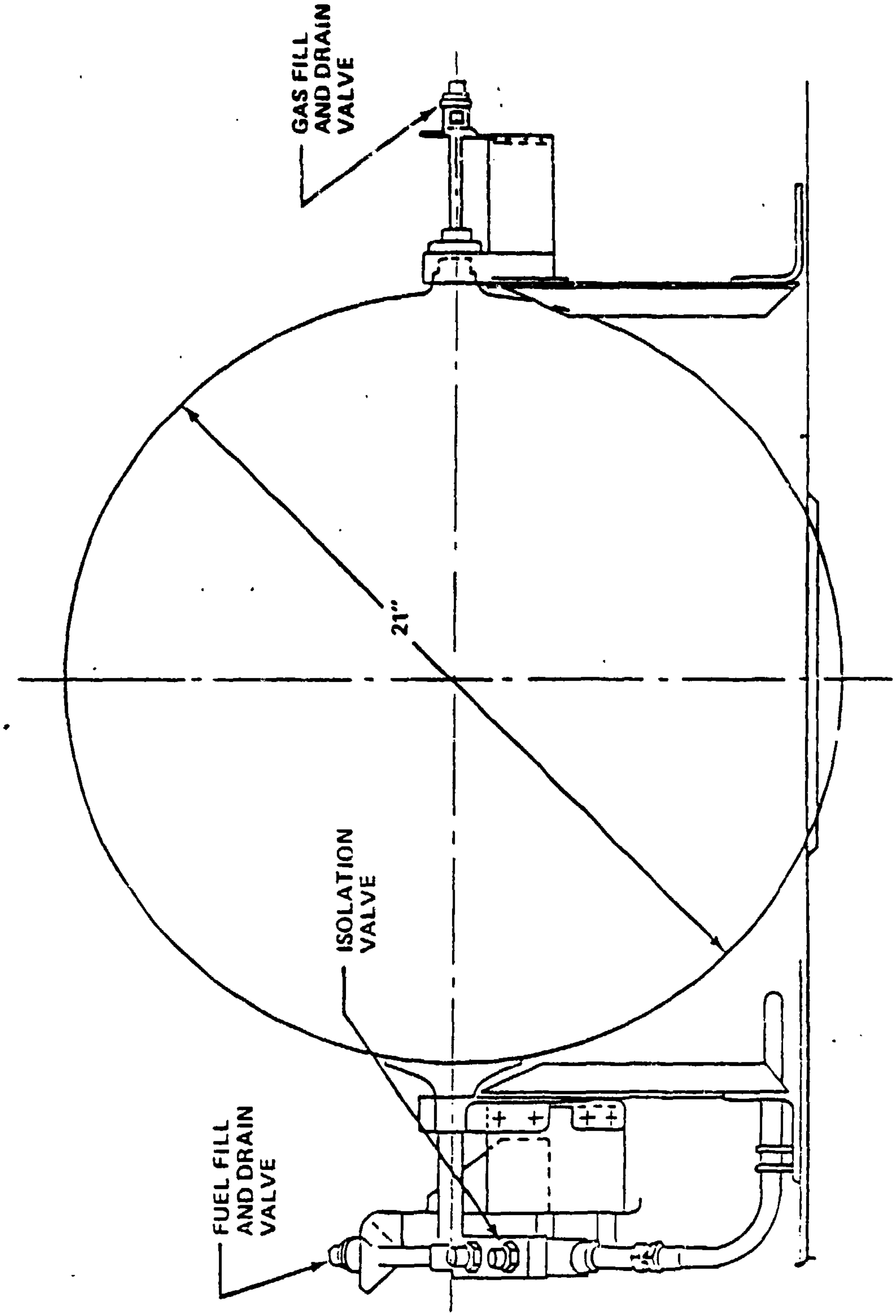


Figure C10-10 RCS Tank Installation

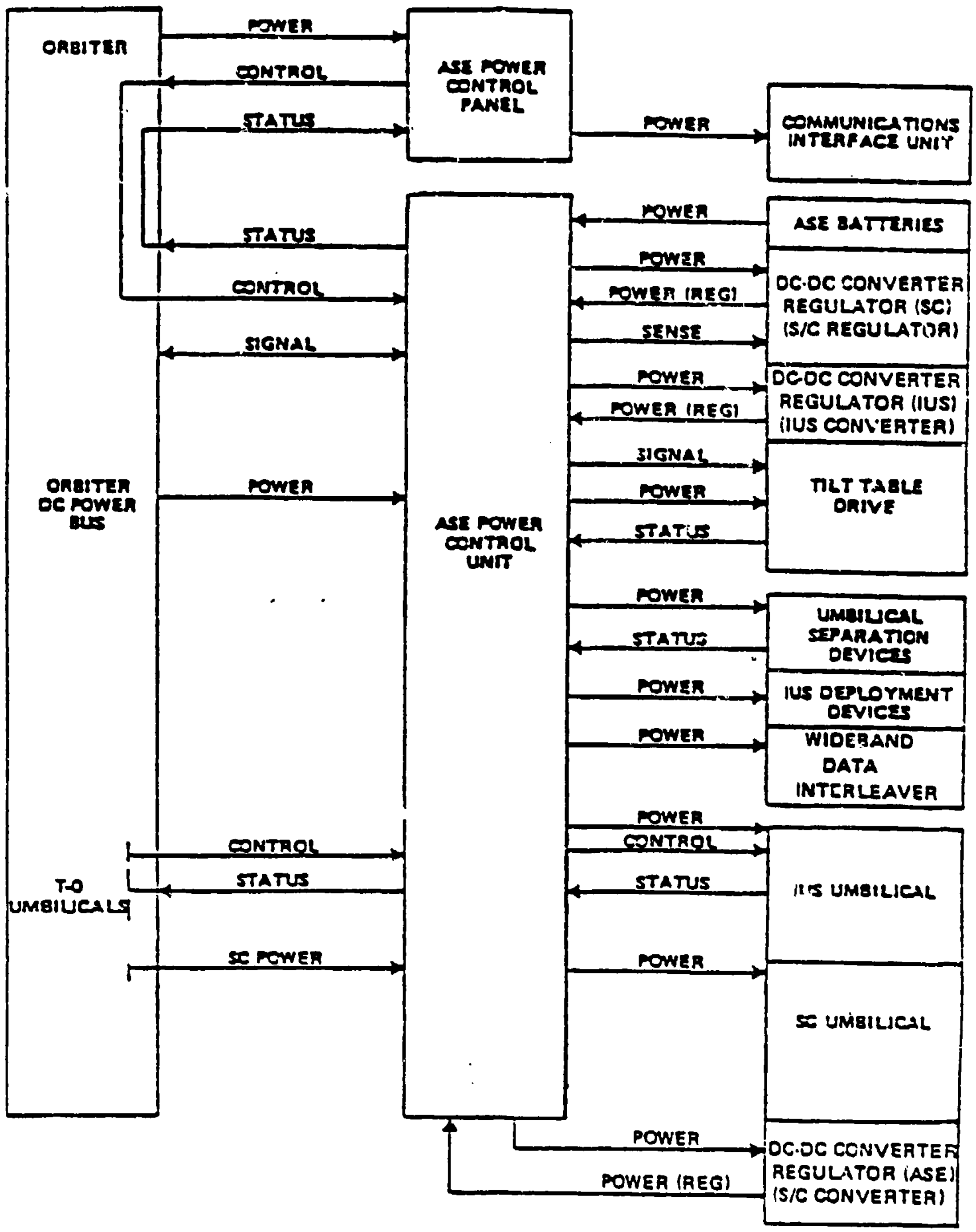


Figure C10-11 ASE Power Control Functional Interfaces

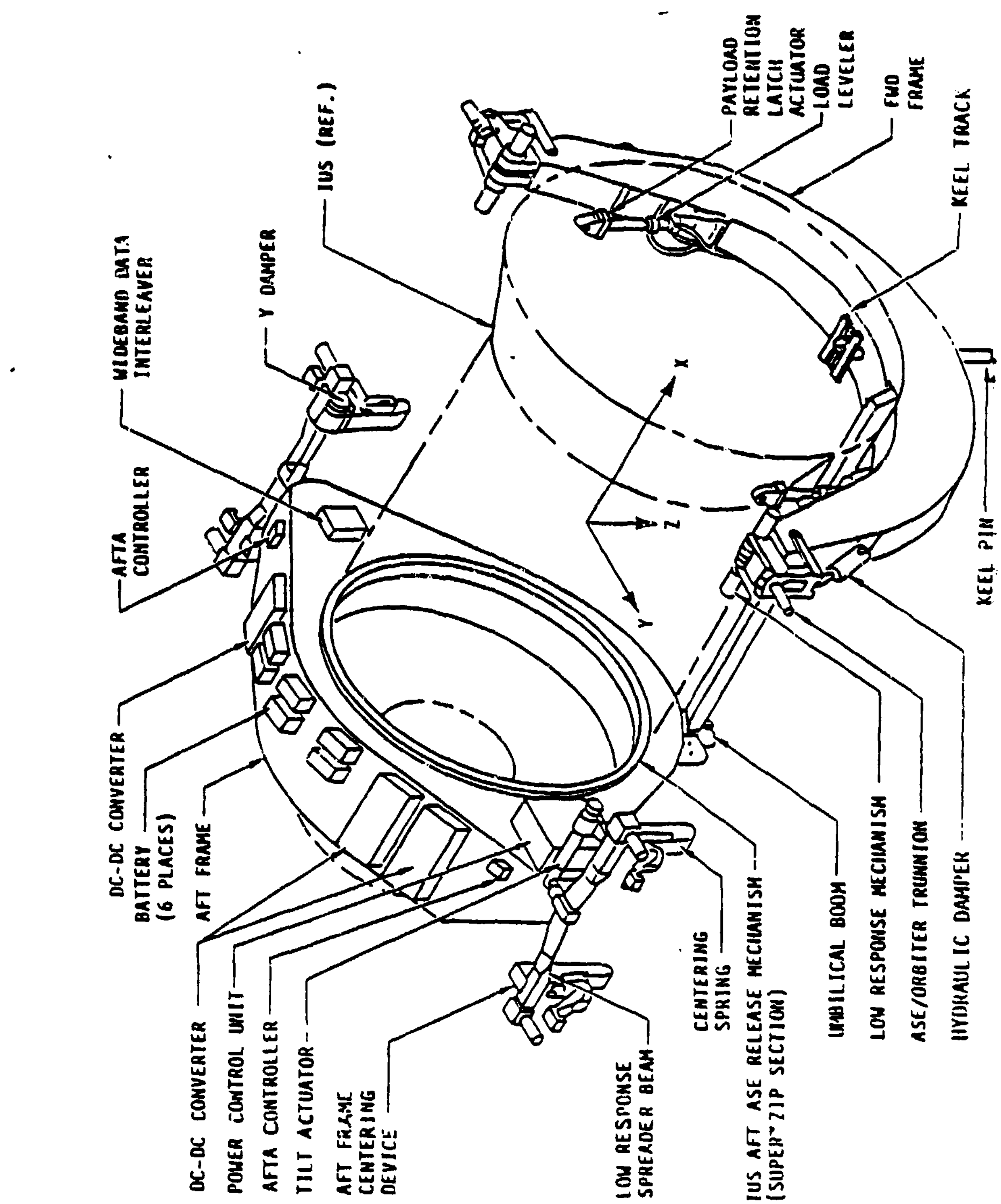


Figure C10-12 Cradle Assembly, IUS Two-Stage ˆ neral Configuration

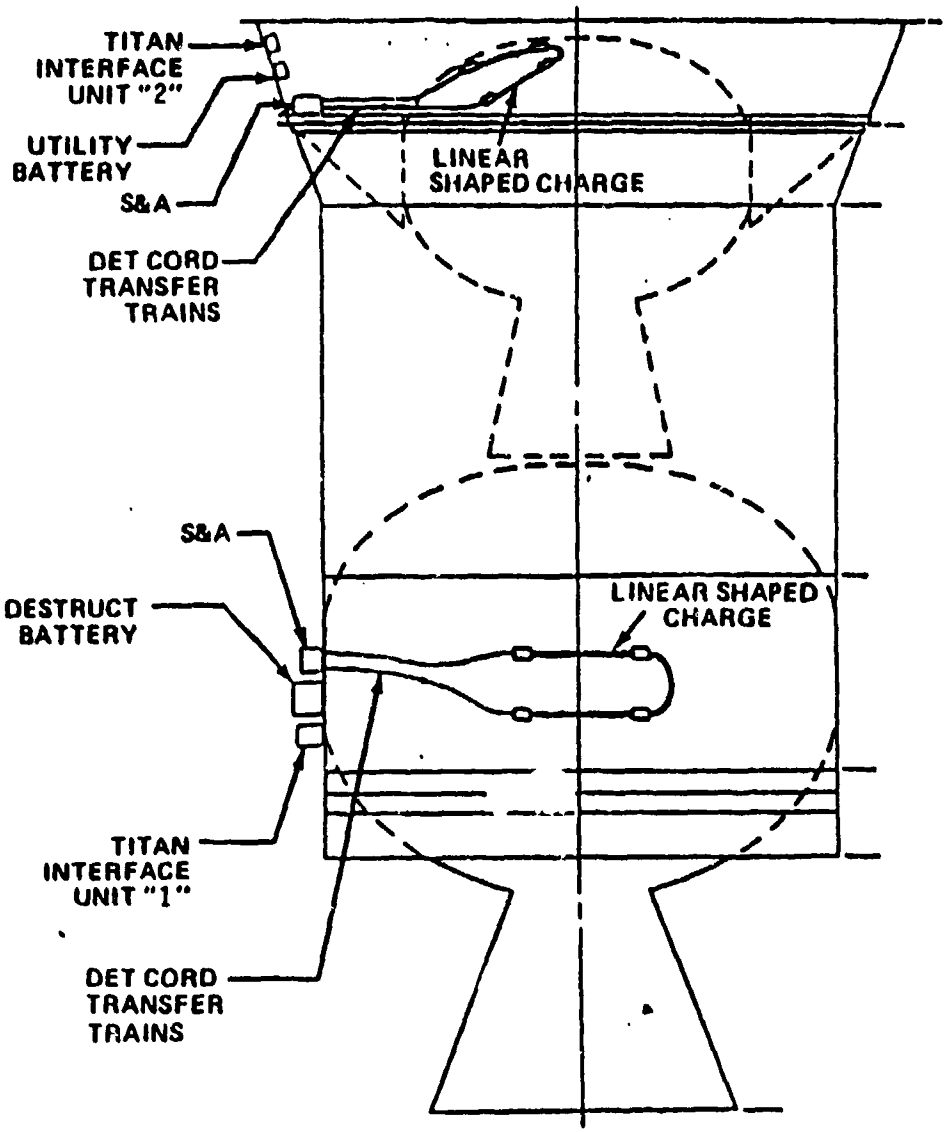


Figure C10-13 Independent IUS/Titan Destruct System

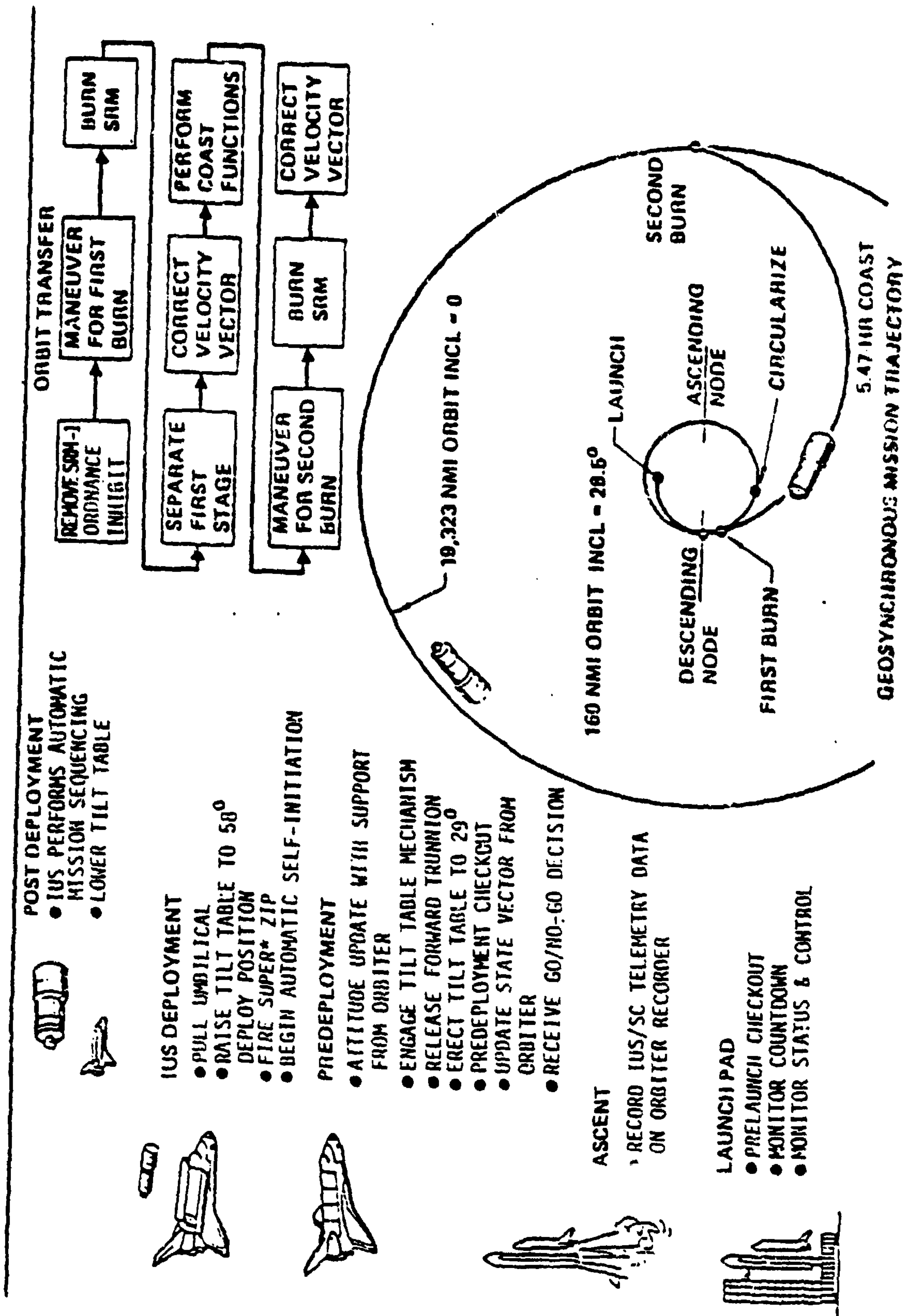


Figure C10-14 STS/IUS Mission Profile (Sheet 1 of 3)

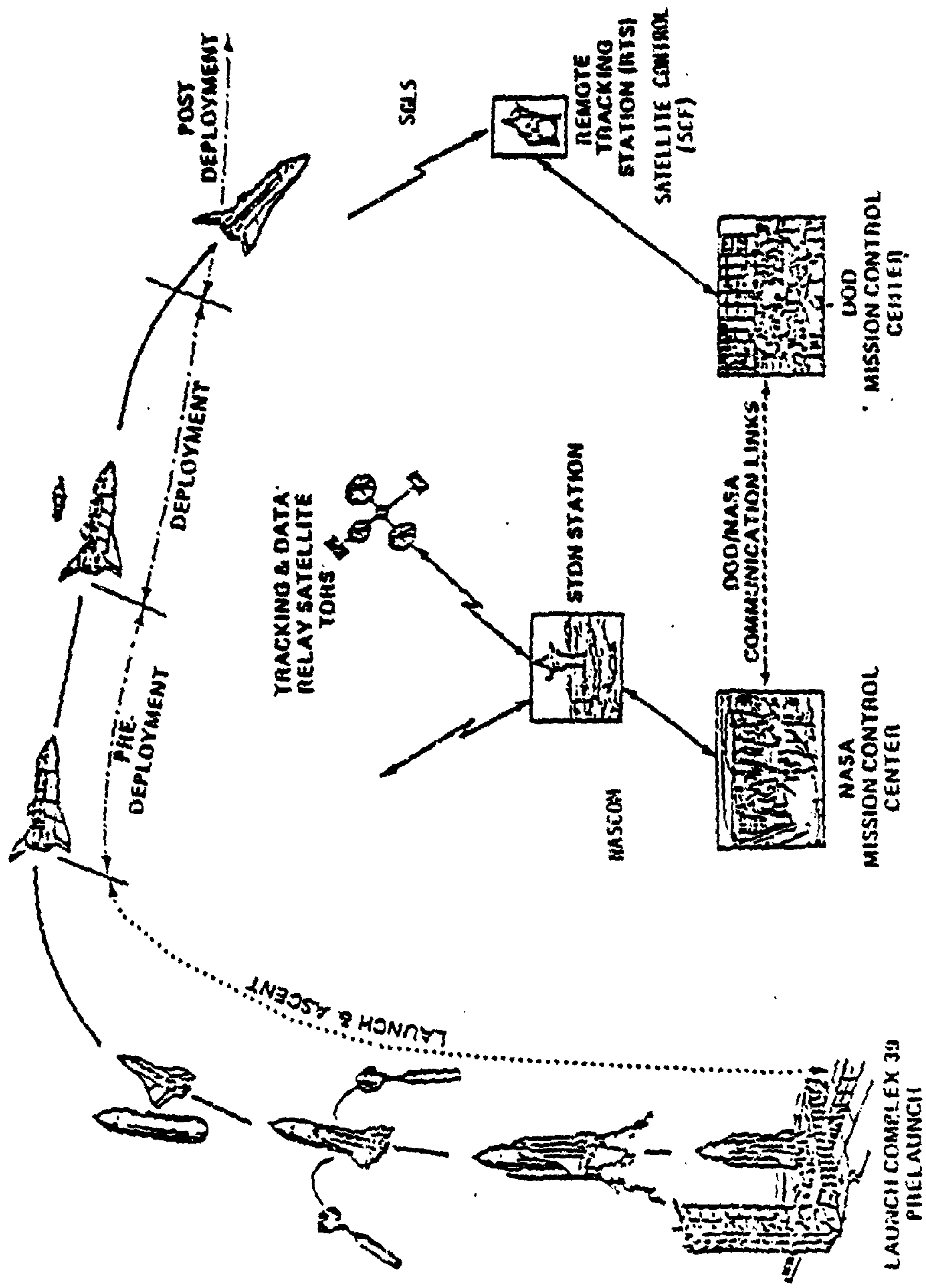


Figure C10-14 STS/IUS Mission Profile (Sheet 2 of 3)

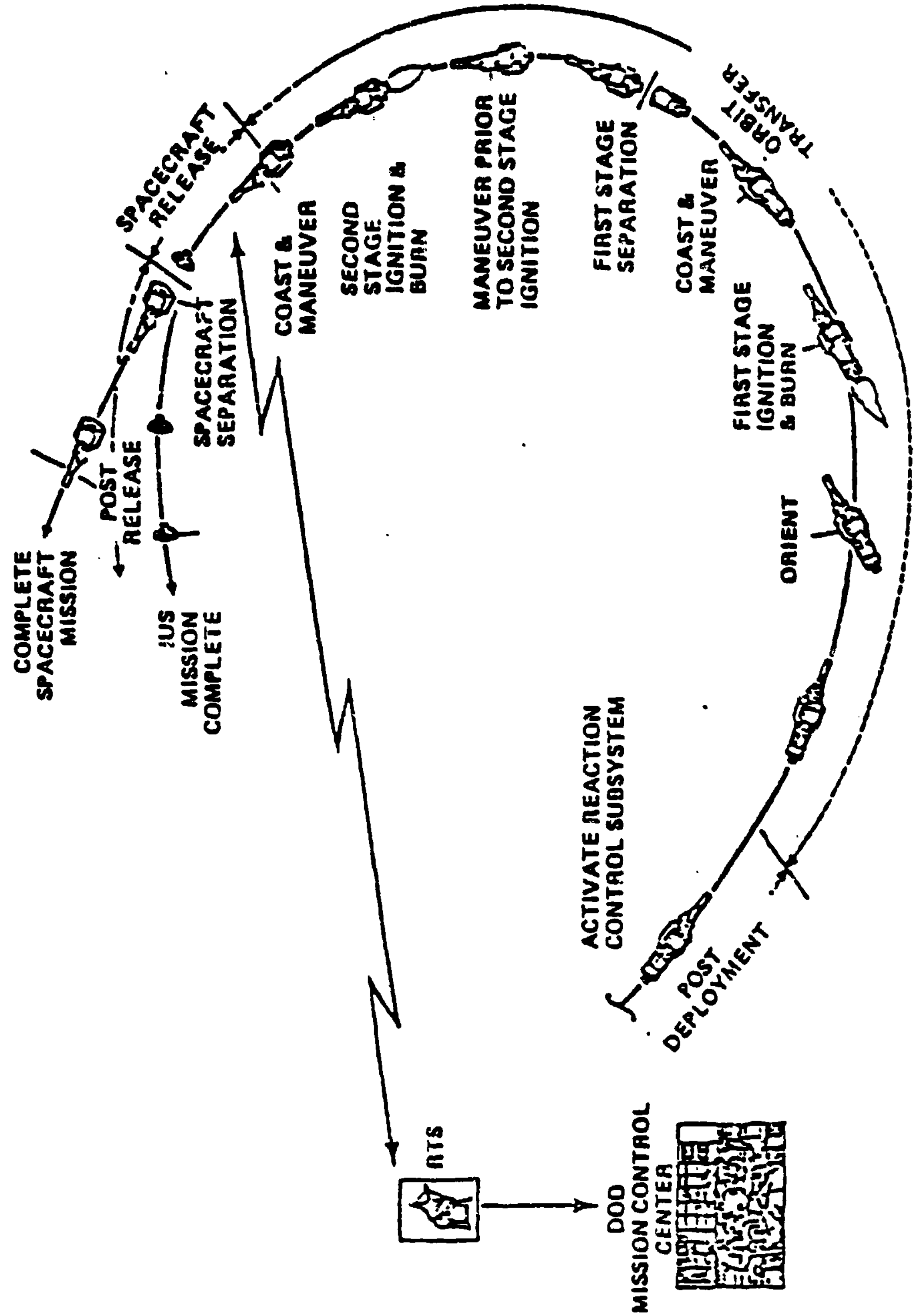


Figure C10-14 STS/IUS Mission Profile (Sheet 3 of 3)

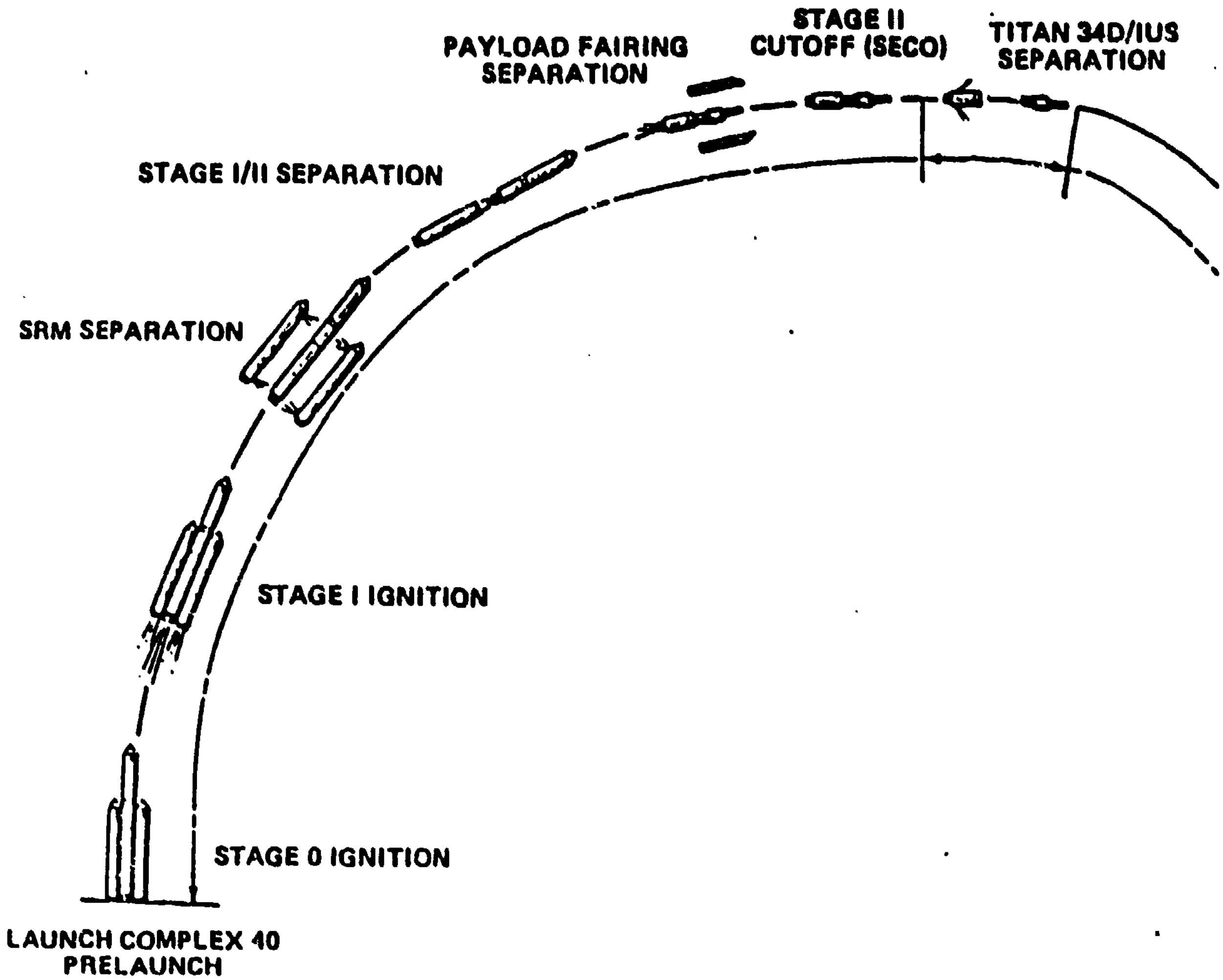


Figure C10-15 T-34D/IUS Mission Profile

Appendix C11
Reserved

APPENDIX C11

Not Available

Appendix C12
Orbit Transfer Vehicle

APPENDIX C12
ORBIT TRANSFER VEHICLE (OTV)

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APPENDIX C12
ORBITAL TRANSFER VEHICLE

C12.0 INTRODUCTION

The following data were extracted from the OTV Concept Definition and System Analysis Study, System Design Concepts," presented at a midterm review to NASA-MSFC by Martin Marietta, 4-7 March 1985, authors Glen Dickman - Cryogenic, and Art Inman - Storable; and also from the "OTV Concept Definition and System Analysis Study, Operations," presented at a midterm review to NASA-MSFC by Martin Marietta, 4-7 March 1985, authors Charlie Garner - Ground, and Jack Mitchell - Flight.

Presented here are the design concepts for a ground-based cryogenic OTV, a space-based cryogenic OTV, a ground-based (STS aft cargo carrier) storable propellant OTV (orbiter launched), and two-stage (Space-Station-based) servicing OTVs (manned and unmanned).

C12.1 GENERAL DESCRIPTION

C12.1.1 Ground-Based Cryogenic OTV

The general arrangement and weight breakdown of the ground-based cryogenic OTV is shown in Fig. C12-1. The four-tank single advanced technology engine configuration fits easily into the Aft Cargo Carrier (ACC). The 40-foot-diameter aerobrake folds forward while stowed in the ACC. It is discarded after flight and not stowed in the orbiter bay for retrieval. The aluminum/lithium propellant tanks are designed by engine inlet pressure requirements. Their thinnest gauges are 0.018 in. for the LO_2 tank and 0.014 in. for the LH_2 tank. The tanks are insulated with multilayer insulation. The hydrogen tanks are removed on-orbit and, with the core system (LO_2 tanks, structure, avionics, and propulsion), are stowed in the orbiter bay for retrieval after mission completion. The propulsion and avionics subsystems reflect the component counts previously discussed. The structure is of lightweight graphite epoxy. The propellant load was selected to enable full use of projected STS lift capability on GEO delivery missions. Note that the resulting payload capability is in excess of any identified mission model requirements.

The ideal OTV program would begin operations in a ground-based mode and shift to a space-based mode with no changes in the flight vehicle design. In this way, the vehicle could be flight proven in a familiar launch environment, and the stage development could be decoupled from the development of space-basing operations. This ideal situation is not realizable primarily because the ground-based system cannot be designed to perform advanced space-based missions within the limitations in the lift capability (size as well as weight) of the STS.

A compromise evolutionary approach meets several practical criteria: Basic subsystem designs should be proven while ground-based; commonality of hardware and operations should be maximized throughout the OTV family; the efficiency of the ground-based vehicle should not be penalized unnecessarily; capability should be added to a flexible system concept as required to meet advancing mission requirements.

Growth to achieve greater mission capability is accommodated through addition and modernization of avionics components within the modular architecture introduction of larger propellant tanks and introduction of more capable aeroshields. Two changes are necessary to achieve space-basing. One is the use of a more open general arrangement to support space maintenance; the other is to add meteoroid shielding to the propellant tanks to protect against long-term exposure. The open arrangement has a profound effect on the appearance of the space-based configuration and makes it necessary to use redesigned support structure.

C12.1.2 Initial Space-Based Cryo OTV

The initial space-based cryogenic OTV is derived from a ground-based concept, but there are several important differences. Propellant capacity has been increased to 55,000 pounds to enable GEO delivery capability to 20,000 pounds of payload. Minimum tank gauges have been reduced to 0.010 in. on the LO_2 tank and 0.012 in. on the LH_2 tank, reflecting lower engine inlet pressure requirements. Meteoroid shielding has been added to the tanks. The general arrangement has been opened up to permit servicing at the Space Station, when necessary, by a space-suited astronaut. Redundancy, including two main engines, has been added to increase mission reliability. Avionics units have been mounted on an avionics ring at the forward end of the vehicle to simplify space-based maintenance. The aeroshield is designed to withstand a peak pressure of 35 psf, enabling retrieval of the unmanned servicing spacecraft.

C12.1.3 Growth Space-Based Cryo OTV

The general arrangement and weight breakdown of the Growth Space-Based OTV is shown in Figure C12-2. In most respects, this vehicle is identical to the initial space-based OTV. The basic structure is identical; the level of subsystem redundancy is the same. Because the electrical power subsystem and reaction control subsystem are fed from the main propellant tanks, no subsystem changes are needed to accommodate different mission durations. Design variation does result from changes in propellant load and heating environment resulting from the delivery and retrieval of heavier spacecraft. Tank size is increased to accommodate an 81,000-pound propellant load. This is large enough to perform the largest lunar missions in a two-stage configuration without excessive compromise in meeting the manned GEO sortie mission requirement. Since the manned return increases the retrieved payload to 14,000 pounds, the peak design pressure of the aerobrake is increased to 63 psf. This results in an increase in TPS thickness and aerobrake structural strength.

C12.1.4 Ground-Based ACC-Storable OTV

The aft cargo carrier configured storable OTV as shown in Figure C12-3 is built around a subsystem module and an airframe truss. The subsystem module provides mounting space for the avionics components and support for the main propellant tanks. The airframe truss supports the tanks laterally and provides the attachment for the main engines and the aerobrake. Both the subsystem module and the airframe truss will be constructed of graphite epoxy composite materials to minimize weight. The main tanks are sized to contain 37,300 pounds of storable propellants and will be constructed of 15-3-3-3 titanium. The tank size selected is adequate to perform the mission model missions in 1993 and 1994 within the projected 72,000-lb lift capability of the Space Transportation System (STS) in that timeframe. The main propulsion system engines are two XLR-132 engines scaled up to 7500 pounds thrust. Extendable exit cones will be provided to an exit ratio of 600 to 1. The nozzles will be extended through the aerobrake while firing and retracted inside the brake contour during the aerobraking maneuver. The selected aerobrake is 23 feet in diameter and will be constructed of a multilayer fabric material of Nicalon, Q-felt, and Nextel sealed with RTV. The fabric will be supported on a graphite polyimide frame or honeycomb. The center section will contain the door through which the engine nozzles will extend and retract. The physical dimensions and weight of the storable OTV require the aerobrake to be no more than 23 feet in diameter. This allows the fully deployed aerobrake to fit within the dimensions of the aft cargo carrier. No deployment mechanisms will be required. At the end of the mission the outer torus of the aerobrake will be jettisoned, and the remainder of the OTV will be installed in the Orbiter bay for return to Earth. No further disassembly of the OTV is required to fit within the envelope of the P/L bay. The with statement is given in Table C12-1(a).

C12.1.5 Ground-Based P/L-Bay-Storable OTV

A minimum-length OTV as shown in Fig. C12-4 has been designed. It will fit within the envelope of the STS P/L Bay and still leave adequate space for the longest payloads identified in the Mission Model. The overall length has been held to approximately 13.5 feet, which leaves 46.5 feet for payload and ASE. The four main propulsion propellant tanks, sized for 37,300 pounds of storable propellant, are of 15-3-3-3 titanium alloy and are supported in a truss and skin structure of graphite epoxy. The subsystem equipment is fitted into the quadrants between the tanks. The main propulsion system will use the XLR-132 engines with 3750 pounds of thrust. Fixed nozzles with an expansion ration of 400 to 1 were selected to minimize length. A 23-ft-diameter deployable aerobrake was selected using the same multilayer fabric selected for the ACC OTV aerobrake design over graphite polyimide support structure. The center support structure is honeycomb, and the outer portion is rib construction. At the end of the mission, the outer portion of the brake will be discarded, and the remainder of the OTV will be installed in the Orbiter P/L bay for return to Earth. The weight statement is given in Table C12-1(b).

Table C12-1 Weight Statement

(a) Ground-Based ACC Storable

Structure		826 lb
Subsystem Modules	208 lb	
Airframe Truss	120	
Tanks	218	
Engine Mounting	101	
Other	179	
Aerobrake Assembly (23 ft Dia)		538
Environmental Control		117
Main Propulsion System		939
Reaction Control System		116
Electric Power System		510
Avionics System		422
Dry Weight		3468
Contingency (15%)		520
Total Dry Weight		3988 lb

(b) Ground-Based Orbiter P/L Bay -

Structure		1348 lb
Airframe	410 lb	
Tanks	458	
Engine Mount	375	
Other	105	
Aerobrake Assembly (23-in Diameter)		542
Thermal Control		117
Main Propulsion System		975
Reaction Control System		116
Electric Power System		422
Avionics System		510
Dry Weight		4030
Contingency (15%)		605
Total Dry Weight		4635 lb

C12.1.6 Space-Based - (GEO) Delivery Vehicle

Either evolving from an initially ground-based geosynchonous OTV as shown in Figure C12-5 or starting as a space-based OTV the basic configuration of a storable space based OTV should be essentially the same. The subsystem module/airframe truss forms an efficient structure to support the main propulsion system, propellant tanks, and avionics and electric power system equipment. Repackaging of the avionics and propulsion components is necessary to accommodate the space-based maintenance activities involving a limited number of technicians for a minimum time. Accessibility to the equipment both by astronauts in EVA gear and by robotics has been a consideration in configuring the space-based vehicles. The ground-based vehicle and the space based vehicles have much in common. The avionics components are the same except for added redundancy. The main engines and feed system are the same 7500-pound-thrust XLR-132 engines and feed system selected for the ACC ground-based OTV. The main propellant tanks are sized for the more ambitious missions identified for later years when the Space Station will be operational. The tank diameter is the same as selected for the ground-based ACC OTV in order that new tooling will not be required and that the assembled OTV can be delivered to the Space Station in the orbiter payload bay. The OTV configuration selected for delivery of payloads to GEO is shown on Figure C12-5. The vehicle will operate as a perigee stage, and an expendable AKM will be provided to insert the payloads into geosynchonous orbit. The aerobrake is constructed of the same materials as the ground-based ACC brake. The size has been increased commensurate with the return weight of the vehicle. Although shown with a 25-foot-diameter brake, this vehicle can be a 32-foot-diameter brake when required to return with additional equipment, such as the multiple payload carrier after delivering multiple payloads.

C12.1.7 Space-Based Unmanned Servicing Vehicle

The unmanned servicing missions to GEO will be performed with a two-stage vehicle, as shown in Figure C12-6, made up of the GEO delivery vehicle, described earlier, as the first stage and a smaller propellant capacity stage as the second stage. The first stage will perform the perigee burn, separate from the second stage and payload, coast out to GEO altitude, return for the aerobrake maneuver, and return to the Space Station. The second stage will continue the mission with the apogee burn to insert the payload into GEO orbit. The stage will stay in the vicinity of the servicer through the duration of the mission and then perform the deorbit burn to bring the servicer back to the Space Station. The smaller second stage is configured similar to the first stage but with smaller tanks and shorter airframe truss structure. The subsystem module, engines and feed system, avionics equipment, and electric power system are the same for both stages. Tanks for the EPS fuel cell reactants will be larger for the second stage because of the longer duration of the Stage 2 mission. The diameter of the tanks for the second stage are the same as the first stage tanks in order that the tooling for the domes can be common. By welding the domes of the fuel tanks together with no barrel section and by adding a short barrel section in the oxidizer tanks, the propellant capacity for the second stage is approximately 25,400 pounds. The aerobrake for the second stage is sized at 32 feet in diameter to bring the unmanned servicer back through the aeromaneuver. Construction of the brake is the same for both stages.

Table C12-2 - Weight Statement

(a) Space-Based GEO Delivery Vehicle

Structure		915 lb
Subsystem Module	208 lb	
Airframe Truss	153	
Tanks	209	
Engine Mount	120	
Other	225	
Aerobrake Assembly (25 ft Dia)		590
Maintenance Provisions		272
Meteroid Protection		157
Thermal Control		96
Main Propulsion System		1198
Reaction Control System		177
Electric Power System		460
Avionics System		569

Dry Weight		4434
Contingency (15%)		665

Total Dry Weight		5099 lb

Table C12-2 - Weight Statement

(b) Space-Based Unmanned Servicing Vehicle

	Stage 1		Stage 2	
Structure		915 lb		762 lb
Subsystem Module	208 lb		208 lb	
Airframe Truss	153		113	
Tanks	209		106	
Engine Mount	120		110	
Other	225		225	
Aerobrake Assembly (25'/32' Dia)		590		887
Maintenance Provisions		272		272
Meteroid Protection		157		108
Thermal Control		96		96
Main Propulsion System		1198		1098
Reaction Control System		177		177
Electric Power System		460		480
Avionics System		569		569
Dry Weight		4434		4449
Contingency (15%)		665		667
Total Dry Weight		5099 lb		5116 lb

Table C12-2 (Cont)

(c) Space-Based Manned Servicing Vehicle

	Stage 1		Stage 2	
Structure		1218 lb		915 lb
Subsystem Module	208 lb		208 lb	
Airframe Truss	224		153	
Tanks	392		209	
Engine Mount	169		120	
Other	225		225	
Aerobrake Assembly (32'x41' Dia)		887		1343
Maintenance Provisions		480		272
Meteroid Protection		226		164
Thermal Control		152		96
Main Propulsion System		1941		1198
Reaction Control System		197		177
Electric Power System		460		498
Avionics System		569		569
Dry Weight		6130		5232
Contingency (15%)		920		785
Total Dry Weight		7050 lb		6017 lb

C12.1.8 Space-Based Manned Servicing Vehicle

The manned servicing missions to GEO will be performed with a two-stage vehicle as shown in Figure C12-7 made up of the GEO delivery vehicle, slightly reoutfitted. The first stage has a larger propellant capacity stage. The first stage will perform the perigee burn, separate from the second stage and manned capsule, coast out to GEO altitude, return for the aerobrake maneuver, and return to the Space Station. The second stage will continue the mission with the apogee burn to insert the manned payload into GEO mission and then perform the deorbit maneuvers to bring the servicer back to the Space Station. The larger first stage is configured for maximum commonality with the second stage. As for the small stage for the unmanned servicing vehicle, the major difference is in the tank length and the airframe truss. The tanks are lengthened but retain the same diameter for tooling commonality to provide capacity for 90,000 pounds of propellant. Because of the mass of the complete vehicle/payload stack, two additional engines have been added. The four-engine arrangement uses the same 7500-pound thrust XLR-132 engine but will require a different feed system. The 53,000-pound propellant capacity second stage is the same basic stage as the GEO delivery vehicle and the first stage of the unmanned servicing vehicle except with a larger diameter aerobrake and larger capacity fuel cell reactant tanks. The fuel cell reactant tanks are sized for support of the 24-day manned mission. The aerobrake is increased to 41 feet in diameter because of the weight of the returning stage and manned capsule.

C12.1.9 OTV Launch Operations

Launch operations analyses have been performed for the following:
- a ground-based OTV launched from KSC in the dedicated ACC (DACC) (cryogenic and storable propellant versions analyzed together with differences noted)
- a ground-based storable propellant OTV launched from KSC in the orbiter payload bay
- a space-based cryogenic propellant OTV (KSC ground processing)
- a space-based storable propellant OTV (KSC ground processing)
- a space-based (cryogenic or storable propellant) OTV (Space Station processing)

C12.1.10 Space-Based OTV Ground Processing

The ground processing flow, Figure C12-8, shows the facilities used for "vertical processing." It was assumed for this analysis that this was the desirable mode, because it intersects the STS flow later in the count.

The space-based OTV presents a minimum amount of ground processing because the major buildup, test, and servicing will be performed at Space Station.

If the space-based OTV evolved from a ground-based program, it is likely that the space-based OTV would continue to use the same dedicated OTV facility.

C12.1.11 OTV Flight Operations Support

OTV flight operations, Figure C12-9, will involve a number of ground operations centers as well as several spaceborne capabilities. Nominal operations for ground-based missions will tie the OTV into operations centers at JSC, GSFC, and JPL as well as an OTV operations center. It may be possible to use a JSC payload MPSR for OTV control functions. Communications capabilities between the OTV, TDRS, DSN, and orbiter will be used. Navigation data will be derived by use of the GPS satellites for missions carrying DOD satellites. Tie-ins with the CSOC and/or the STC will also be necessary.

For space-based missions, additional involvement will include Space Station support for initial and final mission phases.

C12.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C12.2.1 Avionics and Power Equipment - Cryo Stages

The avionics and power equipment used in the selected family of cryogenic stages is summarized. The component redundancy levels are indicated. The level required for the short-duration ground-based missions is somewhat less than that necessary for the space-based vehicles. In the space-based vehicles, the redundancy required by man-rating (a failsafe return philosophy) is somewhat greater than the redundancy suggested by mission "lost cost" considerations. Man-rating redundancy is incorporated in all space-based configurations as indicated in the space-based column of the chart because the analyses indicated it was not economically desirable to maintain two different avionic configurations in the space-based program.

Guidance and navigation information is provided by an IMU supported by star-sensed attitude and global positioning system position updates. Ring laser gyros will be used in space-based configurations because of their long-term stability. Data management is provided by redundant central computers and mass memories. A condition monitor is provided for space-based vehicles. Command and data handling is single string in the ground-based vehicle, redundant in the space-based vehicles. A redundant deployment timer provides safe deployment from the orbiter in the ground-based case, but is not involved in the Space Station deployment scenario. Communications and tracking equipment support a communications interface through both STDN and TDRS. Fuel cells fed from the main cryogenic propellant tanks provide power--redundant radiators are required for the space-based vehicle. The ground-based vehicle requires end-of-mission cryo storage tanks to provide power after main propellant tanks are purged for return into the orbiter.

Table C12-3 - Avionics and Power System Equipment - Cryo Stages

SUBSYSTEM	COMPONENT	NUMBER OF COMPONENTS GROUND BASED	SPACE BASED
GUIDANCE NAVIGATION AND CONTROL	STAR SCANNER	1	N/A
	STAR TRACKER	N/A	2
	IMU	1 DTG	2 RLG
	GPS RECEIVER	1	1
	GPS ANTENNA - LOW ALT	2	2
	GPS ANTENNA - HIGH ALT	1	1
	FLIGHT CONTROLLER	1	2
DATA MANAGEMENT	EXECUTIVE COMPUTER	2	2
	CONDITION MONITOR	N/A	1
TELEMETRY AND COMMAND	COMMAND & DATA HANDLING	1	2
	TLM POWER SUPPLY	1	2
	DEPLOY TIMER	2	N/A
COMMUNICATIONS AND TRACKING	STDN/TDRS XPONDER	1	2
	20W RF POWER AMP	1	2
	S-BAND RF SYSTEM	2	2
ELECTRIC POWER SYSTEM	FUEL CELL (FC)	2	2
	FC RADIATORS	1	2
	EOM REACTANTS TANKS	1 SET	N/A
	FC WATER STORAGE	1	1
	POWER CONTROL & DISTRIBUTION	2	2

C12.2.2 Propulsion System Equipment - Cryo Stages

The propulsion system component count for ground and space-based cryogenic stages is summarized. The basic redundancy philosophy followed in the propulsion system is analogous to that used in the avionics subsystems. Specific identification of components is presented.

The cryogenic stages selected use advanced technology engines delivering 7500 pounds of thrust at a specific impulse of 487.8 seconds. One engine is used on the ground-based vehicle, two on space-based. Both ground and space-based vehicles use an autogenous pressurization subsystem and a thermodynamic vent subsystem. The propellant feed and venting subsystems on the ground-based vehicles require more valves than the space-based vehicles because the hydrogen tanks are purged and removed for retrieval in the orbiter on each flight. The pneumatic control subsystem on the ground-based vehicle is powered by compressed helium, while pressurized hot oxygen and hydrogen from the RCS provides pressure gas for the space-based system. The component count in the ground-based RCS subsystem is particularly low because it uses a simple monopropellant hydrazine system. A more complex main tank fed bi-propellant RCS system was selected for the space-based vehicle because of its more demanding attitude control requirements.

C12.2.3 Ground to Space Evolution - Storable OTV

The goal of identical stages for both modes of operation, is not fully realizable. The limitations of STS lift capability as well as the confines of the ACC and the Orbiter Payload Bay dictate certain compromises in the ground-based configurations. As a result, we have developed an evolution philosophy that will capture many of the advantages of developing and flight proving the stage in the familiar ground environment while not requiring such sweeping changes when transforming to space operations.

The philosophy can be stated as follows: Prove basic subsystems while in the ground-based mode; do not penalize stage efficiency unnecessarily; incorporate redundancy and capacity as mission requirements develop; and maximize commonality of hardware and systems for all stages in the OTV family.

Implementation of the ground to space-based evolution incorporates standardization of structural configurations in the supporting structure and tanks. The tanks for the ACC and all space-based configurations are the same diameter and vary only in the barrel section length. This will allow development of common tooling for production of all tanks. All aerobrakes are blunt body with lift vector pointing and constructed of the same materials. Commonality in the main propulsion system is built around the standard XLR-132 engine with a configuration and all space-based configurations except the 90K stage, which has a four-engine arrangement. Avionics commonality and flexibility is achieved by incorporation of a modular architecture with a plug-in box capability for adding redundancy and advanced technology at the box level when available.

TABLE C12-4 - Propulsion System Equipment - Cryo Stages

SUBSYSTEM	COMPONENT	NUMBER OF COMPONENTS GROUND BASED	SPACE BASED
MAIN ENGINES	COMPLETE ASSEMBLY	1	2
PRESSURIZATION	CONTROL VALVES	16	16
	SCREENS & FILTERS	6	6
	QUICK DISCONNECTS	2	8
VENT	VALVES & FILTERS	40	34
	QUICK DISCONNECTS	3	6
	THERMODYNAMIC VENTS	4	4
PROPELLANT FEED	VALVES & FILTERS	20	17
	QUICK DISCONNECTS	4	12
	PROP UTILIZATION PROBES	4	4
	TOTAL ACQUISITION DEVICE	N/A	1
PNEUMATIC CONTROL	CONTROL VALVES & FILTERS	47	47
	HELIUM BOTTLES	1	N/A
REACTION CONTROL	VALVES & FILTERS	39	94
	TANKS	1	3
	GH_2/GO_2 CONDITIONING	N/A	2
	RCS THRUSTERS	14	14
	QUICK DISCONNECTS	3	30

NOTE: GROUND BASED RCS IS MONO-HYDRAZINE,
SPACE BASED IS MAIN TANK FED BI-PROP

When the OTV operation is moved to the Space Station, certain changes become mandatory to support the space-based operation. For instance, maintainability in the space environment requires component mounting and location compatible with robotics and EVA access. The longer-duration missions defined for the space-based OTV require the addition of protection from long-duration meteoroid exposure. The more ambitious space-based missions, in terms of duration and weight as well as adding man to the payload, dictate increased redundancy, greater tank capacity, and two-stage operation for round-trip servicing missions.

C12.3 MISSION SCENARIO

C12.3.1 Cryo Configuration Summary

Figure C12-10 shows the family of cryogenic stages recommended for a nominal mission model from 1993 to 2010. The ground-based stage is sized at 45,000-pound propellant capacity to fully use STS payload capability when launched in the aft cargo carrier. It will be used to perform single-number and multiple-delivery missions from 1993 until the initial space-based configuration is introduced in 1995. We recommend that this stage employ a single 7500-pound-thrust advanced technology engine. The configuration is tightly packaged to fit assembled in the aft cargo carrier and uses a foldable 40-foot-diameter fabric-covered aerobrake. The aerobrake is designed to support empty stage return at a maximum surface pressure of 23 psf.

The initial space-based configuration is derived from the ground-based stage. Its 55,000-pound propellant load was selected to support the driving 20,000-pound delivery mission. It uses two engines of the same type and most of the same avionics components as the ground-based vehicle. We believe that the arrangement must be opened up to facilitate maintenance in space. Mission duration is increased to 10 days to support the unmanned servicing mission. A 44-foot aerobrake is required to protect the open configuration, and its 35-psf design pressure supports return with the unmanned servicer.

A growth stage is needed to support manned GEO and the larger lunar missions. Both initial and growth stages will be maintained at the Space Station from IOC of the growth stage (1997, per the nominal mission model) throughout space-based operations. The growth stage's 81,000-pound propellant load was selected to support the driving Manned Lunar Sortie in a two-stage configuration. This is slightly larger than the 75,500-pound load required to perform the Manned GEO Sortie, but our preliminary programmatic trades indicate that the selection of the slightly larger stage is cost effective. The mission duration of the growth stage is up to 24 days as required by the Manned GEO Sortie. A 44-foot aerobrake designed for a 63-psf peak pressure enables return with a manned capsule. The two-stage configuration is required to support the Manned Lunar Sortie, where 65,000 pounds of payload is delivered in conjunction with a 15,000-pound round-trip manned sortie.

This table C12-5 is a listing of the major components of the electrical and avionics systems and shows a comparison of the quantities for the ground-based and space-based stages. Additional components are added in the space-based stages to increase the probability of mission success and satisfy the man rating requirement of "no single failure preventing safe return of the astronauts." A condition monitor is added in the space-based systems to facilitate space monitoring and maintenance.

Table C12-5 Avionics and Power System Equipment - Storable Stages

Subsystem	Component	Number of Components Ground-Based	Space-Based
Guidance	Star Scanner	1	N/A
Navigation	Star Tracker	N/A	2
and Control	IMU	1 DTG	2 RLG
	GPS Receiver	1	1
	GPS Antenna - Low Alt	2	2
	GPS Antenna - High Alt	1	1
	Flight Controller	1	2
Data Management	Executive Computer	2	2
	Condition Monitor	N/A	1
Telemetry	Command and Data Handling	1	2
and	TLM Power Supply	[illegible]	2
Command	Deploy Timer	2	N/A
Communications	STDN/TDRS Xponder	1	2
and	20W RF Power Amp	1	2
Tracking	S-Band RF System	2	2
Electric	Fuel Cell (FC)	2	2
Power	FC Radiators	1	2
System	FC Reactants Tanks	2	2
	FC Water Storage	1	1
	Power Control & Distribution	2	2

Table C12-6 presents an approximate count of the propulsion components for the ground-based and space-based storable OTV configurations. The differences are the number of engines on the stages and the number of quick disconnects required to facilitate space maintenance of the systems. For the ground-based configurations, the ACC version has two XLR-132 engines with 7500-lbf thrust while the orbiter payload bay configuration has four XLR-132 engines with 3750-lbf thrust. The orbiter payload bay OTV also requires additional valves for the dump system required for abort conditions. Total propellant acquisition devices have also been added in the propellant tanks of the space-based stages to assist in propellant handling at the Space Station propellant terminal. The selected reaction control system is a monopropellant hydrazine system for the ground-based stages and a main-tank-fed bi-propellant system for the space-based stages. Use of a bi-prop system for space-based stages eliminates an extra commodity to be handled at the Space Station.

Table C12-6 Propulsion System Equipment - Storable Stages

Subsystem	Component	Number of Components Ground Based		Space Based	
Main Engines		2	4	2	4
Pressurization and Vent	Control & Check Valves	17	17	21	21
	Filters & Diffusers	5	5	7	7
	Quick Disconnects	5	5	9	9
	Tanks	1	1	1	2
Propellant Feed	Valves, Filters, & Screens	18	30	14	18
	Quick Disconnects	2	4	10	14
	P.U. Probes	4	4	4	4
	Total Acquisition Device	0	0	4	4
Pneumatic System	Control Valves	6	10	6	10
	Accumulators	1	1	1	1
	Quick Disconnects	0	0	2	4
Reaction Control System	Valves & Filters	39	39	68	68
	Quick Disconnects	3	3	16	16
	ACS Thrusters	14	14	14	14
	Tanks	1	1	2	2

Note: Ground-based RCS is mono-hydrazine; space-based is main tank fed bi-prop.

C12.3.2 GEO/HEO Delivery Capability - Ground-Based Cryo

GEO delivery capability with STS projected lift capability (72,000 lb; East launch to 140 nmi; JSC; Aug 1984) is in excess of 16,500 pounds. This capability is in excess of maximum requirement derived from the mission model. This capability reflects a flight performance reserve of 2% on all required mission velocity requirements; fluid residuals at burnout of 1.5% of loaded propellant; mission expendables totaling 738 pounds over a 3-day mission; net trajectory shaping losses of 358 ft/sec; and an advanced engine delivering 7500 pounds of thrust at a 487.8-second specific impulse.

A summary of the high Earth orbit (HEO) capability of the selected ground-based cryo is presented. The GEO capability is in excess of 16,500 pounds. The Lunar delivery capability reflects the same general ground rules as the GEO delivery mission, that except total mission duration is increased to 21 days and mission expendables to 1611 pounds. An approximate 17,000-pound payload capability results. The planetary delivery mission reflects a mission plan that drives the OTV and its payload (spacecraft plus kickstage) to a C3 of 9 (km/sec) squared; retros the OTV into a 4-day return orbit; uses a two-burn technique to turn the OTV into the regressed orbiter orbit; and returns to the orbiter using an aeroassisted maneuver. An approximate 18,000-pound payload capability to C_3 = 9 results.

C12.3.3 Manned GEO/HEO Sortie Capability - Growth Space-Based Cryo

The selected GEO stage size has a capability in excess of 15,500 pounds. The propellant penalty for the selected stage size with 14,000-pound spacecraft is approximately 170 pounds. Our preliminary programmatic analyses indicate that this is an acceptable compromise. These data were generated with the same general ground rules that were used for the initial space-based vehicle. The total mission expendables budget needed for the 24-day manned lunar sortie is estimated at 1886 pounds. The net trajectory shaping losses were 203 ft/sec. As in the initial space-based cryo, propellant for the electrical power subsystem and the reaction control subsystem is largely retrieved from MPS boiloff.

A summary of the HEO capability of the selected Growth Space-Based Cryo is presented. The growth space-based OTV is capable of rountripping a manned sortie spacecraft weighing in excess of 15,500 pounds. Lunar and planetary groundrules were as described for the initial space-based cryo. Lunar-one stage delivery capability is nearly 39,000 pounds. In its two-stage configuration, this vehicle was sized to perform the mission requirements of the Manned Lunar Sortie mission: 80,000 pounds up, 15,000 pounds back. The one-stage capability to a planetary (C_3 = 9 (km/sec squared) mission is approximately 44,00 pounds.

C12.3.4 Storage Configuration Summary

Figure C12-11 is a pictorial presentation of the complete storable OTV family of high potential stages. This family of stages will perform the missions identified in the Mission Model with the exception of the heavy lunar missions in the post 2006 timeframe. Two configurations for the early ground-based OTV are defined on the left of the chart: one carried aloft in the ACC and the other configured to fit in the Orbiter Payload Bay. Both are sized to take advantage of the total lift capability of the STS in the early 1990s and are outfitted to deliver unmanned single or multiple payloads, as identified in the Mission Model, to GEO operating as a perigee stage. The space-based family is built around three stages with propellant capacities carefully selected to most efficiently perform the broad range of identified missions. The 53,000-lbm capacity stage is the workhorse configuration that has an application in all GEO missions. Operating as a perigee stage, it is the GEO delivery vehicle for single and multiple payloads. For unmanned round-trip servicing missions, the 53,000-lbm stage is combined with the 25,000-lbm stage to form a two-stage vehicle. For the demanding requirements of the manned trips to GEO, the 53,000-lbm stage is mated to the 90,000-lbm stage to form another two-stage configuration. The 53,000-lbm stage will be fitted with appropriately sized aerobrakes for the size and weight of the body being returned from GEO. When only the stage is returning, as from delivery missions, only a 25-foot-diameter brake is required. When returning from delivery of multiple payloads, the multiple payload carrier returns with the stage, and therefore the required brake size is 32 feet in diameter. Bringing the manned capsule back from the manned servicing mission is the most demanding mission for the 53,000-lbm stage and requires a 41-foot-diameter brake. The 90,000-lbm stage is sized for the first-stage application on the manned servicing vehicle; however, it will be the primary vehicle for performing the planetary missions in the mission model. Some of the less demanding planetary missions can be performed by the 53,000-lbm and ground-based stages. The identified application for the 25,000-lbm stage in the current mission model is for the second stage of the two-stage unmanned servicing vehicle.

C12.3.5 Ground Based - ACC/P/L Bay Storable - Performance

A comparison of the performance capability of the ground based ACC OTV and the lift capability of the STS shows that the main propulsion system propellant capability is for payloads of 10,000 to 14,000 pounds delivered to GEO transfer orbit. Since the Storable OTV operates as a perigee stage for delivering payloads to GEO, the performance calculations have considered the impact of a solid apogee kick motor (AKM) to provide the energy to insert the payload into geosynchonous orbit. The "STS Cargo" curve includes the weight of the loaded OTV in the Aft Cargo Carrier and the weight carried aloft in the ACC and the Payload bay, the weight that remains in the orbiter during the mission, and the weight that returns to Earth in the Orbiter. The intersection of the "STS Cargo" and the "STS

Lift Capability" curves indicate that a payload of 13,150 pounds plus its AKM can be delivered to LEO, and the OTV can deliver the payload and AKM to transfer orbit using approximately 35,400 pounds of propellant. The propellant quantity considers the quantity required to return the OTV through the aerobraking maneuver and circularising at an altitude for pickup by the Orbiter for return to Earth.

A comparison of the performance capability of the ground-based OTV designed to be carried to LEO in the Orbiter payload bay and the lift capability of the STS shows the main propulsion system propellant capabilities for payloads of 10,000 to 14,000 lbm delivered to GEO-transfer orbit. Since the Storable OTV operates as a perigee stage for delivering payloads to GEO, the performance calculations have considered the impact of a solid AKM to provide the energy for insertion of the payload into geosynchonous orbit. The "STS Cargo" curve includes the weight of the loaded OTV, the payload, and the ASE in the Orbiter P/L Bay. The "STS Lift Capability" curve is 72,000 pounds for a 140-nmi orbit as defined by NASA for use in this study. The intersection of the "STS Cargo" and the "STS Lift Capability" curves define the maximum payload (13,500 lb), along with its AKM, that can be carried with the loaded OTV by the STS. The OTV, with approximately 35,000 pounds of propellant, can deliver the payload and AKM to GEO-transfer orbit and return for pickup by the Orbiter for return to Earth.

C12.3.6 <u>HEO Capability - 53K/90K Space Based Storable</u>

The capability of the selected 53,000-pound storable stage is to deliver, with the assistance of an expendable AKM, payloads to high-altitude circular orbits with an inclination of 28.5 degrees. It is significant to note the extremely heavy payloads that can be delivered to orbits up to 5,000 nautical miles. In excess of 60,000 pounds can be delivered to a 2,000-nmi circular orbit.

The capability of the selected 90,000-lb storable stage is to deliver, with the assistance of an expendable AKM, payloads to high altitude circular orbits with an inclination of 28.5 degrees. Of interest is the extreme capability to the lower circular orbits. This stage can deliver in excess of 109,000 pounds of payload to a 5,000-nmi orbit or as much as 62,000 pounds to a 10,000-nmi circular orbit. The GEO-transfer capability for the 90K stage is 38,700 pounds of payload plus an appropriate AKM. Lunar orbit capability when the stage is operating as a single stage is approximately 22,500 pounds. For planetary missions, the 90K stage has the capability of delivering 29,200 pounds to a C_3 = 9 (km/sec) squared.

C12.3.7 <u>Planetary Missions With Storable OTVs</u>

Performing planetary missions with a recoverable OTV requires complex flight maneuvers. Our scenario involves delivering the planetary spacecraft to C_3 = 9 (km/sec) squared with the OTV and then returning the OTV to the space station. The remaining energy required for the spacecraft will be provided by an expendable kick motor. Planetary missions in the Mission Model can be accomplished with a single-stage storable OTV using one of the stages in the space-based family. Evaluation of the planetary missions indicate that approximately 25,000 lbm to 76,000 lbm of storable propellant will be sufficient to accomplish each mission. Either the ACC-# or the payload-bay-configured ground-based OTV has adequate capability to perform the near-Earth asteroid rendezvous mission in 1993 (approximately 25,000 lbm of propellant for the ACC-configured OTV). The later planetary missions will require either the space based 53,000- or 90,000-lbm stages for sufficient propellant capacity to perform the missions.

C12.3.8 Lunar Missions With a Storable OTV

A scenario for performing the lunar missions in the mission model with storable or a combination of storable and minimum-development cryo stages is presented. The early lunar missions involve delivering payloads of 5,000 and 20,000 pounds to either a lunar orbit or to the vicinity of the moon. The heavy missions for manned sorties and lunar base delivery do not start until the year 2006. Evaluation of the capability of the storable stages in the space-based inventory shows that the early missions can be performed quite efficiently with a storable single-stage vehicle. The 53K storable OTV will perform the early 5,000-pound missions with just under 42,000 pounds of propellant. The 20,000-pound mission can be performed with the 90K stage using a little over 84,000 pounds of propellant. The post-2006 manned and lunar base missions will require a new development first stage to provide the energy to send the payload with a storable stage into translunar orbit. The stage can be a minimum development stage using the avionics components available on the storable stages and two existing cryogenic engines, such as the current RL-10A-3-3A. The 53,000-pound propellant capacity storable stage is adequate to perform the lunar maneuvers and return the spacecraft to Earth. The propellant quantities are shown for this combination of stages to perform the heavy lunar missions.

Lunar Base and Manned Mission (Post-2006)
- Use 2 Stage OTV
  - Storable Stage 2
  - Min Development Cryo Stage 1

	Stage I Cryo	Stage II Storable	Total Propellant
80,000 lb P/L to Lunar Orbit	147,561 lbm	31,178 lbm	178,739
80,000 lb up and 15,000 Return	157,994 lbm	39,422 lbm	197,174
80,000 lb up and 10,000 Return	154,500 lbm	36,674 lbm	191,174

C12.3.9 Ground-Based OTV Mission

The mission scenario developed for the ACC-launched OTV shown in Figure C12-12 differs in several important respects from that of a typical payload-bay-launched payload.

The orbiter OMS-2 burn establishes a 130-nmi intermediate rendezvous orbit at 45 minutes MET. This altitude was selected as a compromise to maximize injected payload weight while keeping drag on the orbiter at an acceptable level for subsequent LEO operations. At 1 hr 25 min MET the OTV completes a series of burns to circularize at 140 nmi and await rendezvous by the orbiter. A standard orbiter rendezvous sequence is initiated by an OMS burn at 4 hrs 9 min MET. At 6 hrs 24 min the orbiter achieves a stable position approximately 8 nmi behind and co-orbital with the OTV.

On the second mission day the orbiter accomplishes the final phase of the rendezvous. The OTV is grappled by the RMS after an approach that brings the orbiter from behind the OTV to a position 1000 ft ahead of the OTV before final closure to grapple range. The CTV is stabilized for payload mating by attaching it to the payload installation and deployment aid (PIDA) using the RMS.

Following OTV and payload mating operations as well as final pre-deployment checkout, the payload and OTV are released by the RMS. The orbiter performs maneuvers to provide safe separation distance and return to a 130-nmi orbit to await eventual retrieval of the OTV. The OTV performs a series of burns to place the payload in its mission orbit.

At the proper time for orbital alignment and phasing, the OTV performs a deorbit burn, which establishes the proper trajectory for aerobraking. The burns and aeromaneuver result in an OTV orbit at 140 nmi with the OTV phased properly for subsequent orbiter rendezvous.

Orbiter rendezvous is accomplished in a manner similar to the activity at the beginning of the mission. After grappling and placement on the PIDA, the H_2 tanks (for a cryo propellant OTV) are removed and stowed in the payload bay. The remainder of the OTV is then stowed for return to the launch site. For storable propellant OTVs, no disassembly is required. They can be stowed in the bay immediately after grapple without use of the PIDA.

Assessment of the computer-generated RMS motions for the OTV grapple through payload mating and redeployment indicate that there are no incompatibilities with the RMS. A forward bulkhead-mounted CCTV camera can provide visibility of the PIDA and OTV payload interface attachment areas.

Ku-band antenna blockage will be a problem for most orbiter attitudes, but the s-band communications capability should provide adequate back-up communications. The thermal environments induced on the OTV and orbiter while the OTV is mounted on the PIDA do not appear to cause problems for either vehicle. Propellant slosh effects on RMS operations have not been analyzed but could be alleviated by tank design if necessary. The operations appear to be feasible based on a development program for the PIDA.

C12.3.10 Space-Based OTV Mission

OTV missions originating from the Space Station shown in Figure C12-13 share many aspects with ground-based missions. There are also significant differences in a number of areas. The OTV is initially delivered to the Space Station in major subassemblies which are put together and checked out in the OTV hangar at the station. The STS continues to provide logistics support and ferries OTV payloads from the ground to the station. After payload mate and checkout, the OTV is fueled and deployed from the Space Station. Upon reaching a safe distance from the station, the OTV performs transfer and circularization burns to place the payload in the desired mission orbit. At the appropriate time for orbit alignment and rendezvous phasing, the deorbit burn, aeropass, and return orbit maneuvers are performed to bring the OTV back to the vicinity of the Space Station for retrieval and servicing for its next mission.

Several options for OTV deployment and return have been investigated. These include deployment by OMV, by tether, and by OTV separation mechanism and RCS thrusting. Return options include OMV and tether rendezvous.

Tethered deployment of the OTV and payload offers an opportunity to reduce the OTV's mission velocity requirement and hence the propellant used. This deployment method involves the attachment of the OTV to the end of a tether on a payload interface module (PIM). The tether is reeled out to a length of as much as 80 nmi. The time required to deploy the tether is expected to require four to six hours.

The PIM releases the OTV and payload, which are instantaneously placed into 342 x 801 nmi orbit. At the same time the Space Station orbit is reduced to 204 x 261 nmi. It should be noted that the Space Station will probably be in an orbit 40 nmi higher than shown, assuming tether operations will also be used to assist deorbiting of the Shuttles.

After release by the tether, the OTV performs state vector and attitude updates in preparation for its transfer orbit burn at the next perigee passage.

C12.3.11 Ascent Profile

The ACC-launched OTV ascent, Figure C-12-14, is based on a standard STS trajectory leaving the ET suborbital at MECO. Performance estimates have shown a gain of several thousand pounds of payload when compared with a scenario which places the ET into orbit with the orbiter. The OTV is protected from ascent atmospheric environment effects by the ACC shroud, which is jettisoned shortly after SRB separation.

The OTV deployment sequence begins at STS MECO. The STS and attached OTV are in a 16x80 nmi sub-orbital trajectory designed to accommodate ET disposal. A coasting delay follows while the STS nulls out shutdown attitude transients. Downward-firing aft RCS jets are inhibited until after OTV is separated in order to prevent plume impingement on the exposed OTV located directly beneath the orbiter boattail. The OTV is separated by springs that provide a 2 ft/sec retrograde relative separation velocity. After 30 seconds of coast, sufficient clearance exists to re-enable the inhibited orbiter RCS jets. A normal ET separation sequence follows with ET impact occurring safely downrange.

C12.3.12 OTV/Orbiter Trajectory Plot

Figure C12-15 depicts a polar view of the orbiter and OTV trajectories leading to the initial rendezvous. As previously described, the orbiter completes OMS burns shortly after MECO to establish itself in an intermediate 130-nmi parking orbit.

Following navigation update using the GPS and stellar attitude update, the OTV completes burns that result in a 140-nmi orbit with proper phasing for subsequent orbiter rendezvous. NH-1, NC-3, and NH-2 orbiter burns result in an orbiter position 8 nmi behind the OTV at the same orbital altitude.

C12.3.13 Ground-Based Planetary Mission

Planetary missions flown by the OTV in a ground-based mode, Figure C12-16, will begin from an orbit with an inclination and launch window optimized to achieve the desired hyperbolic velocity vector. After deployment and separation from the orbiter, the OTV main engine burn places the OTV and spacecraft on a hyperbolic escape trajectory. The OTV then separates from the spacecraft and increases the separation rate by performing a small RCS burn. About 15 minutes after separation, the OTV performs another burn to place itself in a highly elliptical orbit. Three additional burns are performed on this ellipse prior to the aerobraking maneuver. Two of these burns accomplish a plane change to align the OTV's trajectory with the orbiter rendezvous plane, which is regressing relative to the OTV orbit. The third burn (actually the second burn in sequence) lowers the perigee of the ellipse to the appropriate altitude for the aerobraking maneuver. The plane change maneuvers are designed to minimize the energy required to transfer to the rendezvous plane.

C12.3.14 Ground-Based 12-Hour Circular Mission

Orbit transfer to high inclination orbits must also account for rendezvous orbit precession. The problem of deploying payloads to different inertial planes adds additional complexity. The orbiter is launched at the proper time and into the proper inclination so that the OTV makes an in-plane transfer to the first satellite plane. The plane change between the two inertial planes is optimized by employing a three-burn strategy. The first burn recircularizes at the payload deployment orbit altitude. Following deployment of the second payload, the OTV performs a small burn to set up proper phasing for rendezvous. A combined deorbit and plane change burn sets the OTV on a trajectory leading to the aerobraking maneuver and, ultimately, the rendezvous with the orbiter.

C12.3.15 Manned/Unmanned Servicing Mission

Other variations on the GEO mission are the GEO manned and unmanned servicing missions, Figure C12-17. The OTV is used in these missions as a basic orbit transfer vehicle to transport autonomous manned or unmanned servicing vehicles to a desired location in the GEO arc. Locations within about 23° longitude can be achieved by selection of the proper node for the perigee burn. Fine tuning is accomplished by using two perigee burns. The first perigee burn places the OTV and servicing vehicle on a phasing orbit with a period (90 to 180 minutes) selected to shift the nodal location for the second perigee burn to the longitude appropriate for the GEO orbit target.

Additional propellant can also be loaded to allow transfer between locations on the GEO arc.

C12.3.16 Space-Based Planetary Mission

The space-based planetary mission, Figure C12-18, uses a strategy similar to the ground-based mission. The difference is in the launch window and orbital inclination of the OTV deployment orbit. Because the Space Station is in a 28.5° inclined orbit, the OTV launch window will be determined to minimize the total plane change required to change planes and allow rendezvous in the Space Station orbit plane.

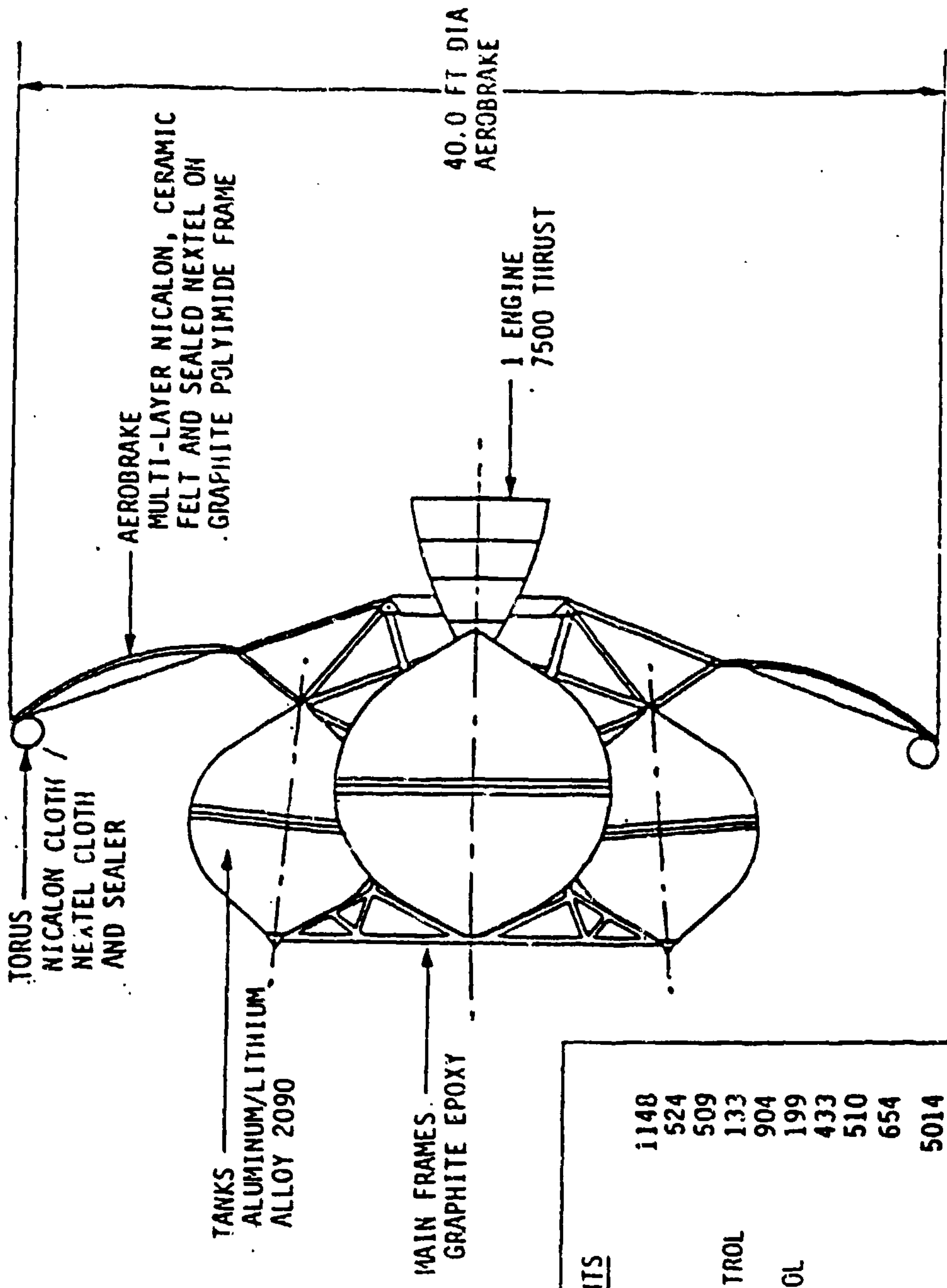
C12.3.17 Lunar Missions

Lunar missions using the OTV are accomplished stage vehicles. Two perigee burns are employed to reduce velocity losses for translunar orbit insertion. The second stage accompanies the lunar payload into lunar orbit and supplies necessary mid-course circularization in lunar orbit and Earth return velocity. Strategies for which stages to combine for the range of lunar missions have been previously discussed for cryo and storable propellant OTVs.

C12.3.18 Deorbit, Aeropasss, and Recovery Orbit

The return leg for space-based missions, Figure C12-19, is a variation of the previously described ground-based mission. In this case the OTV exits the atmosphere in an orbit with a 245-nmi apogee. The phasing orbit provides an opportunity to correct inclination and relative phasing with the Space Station.

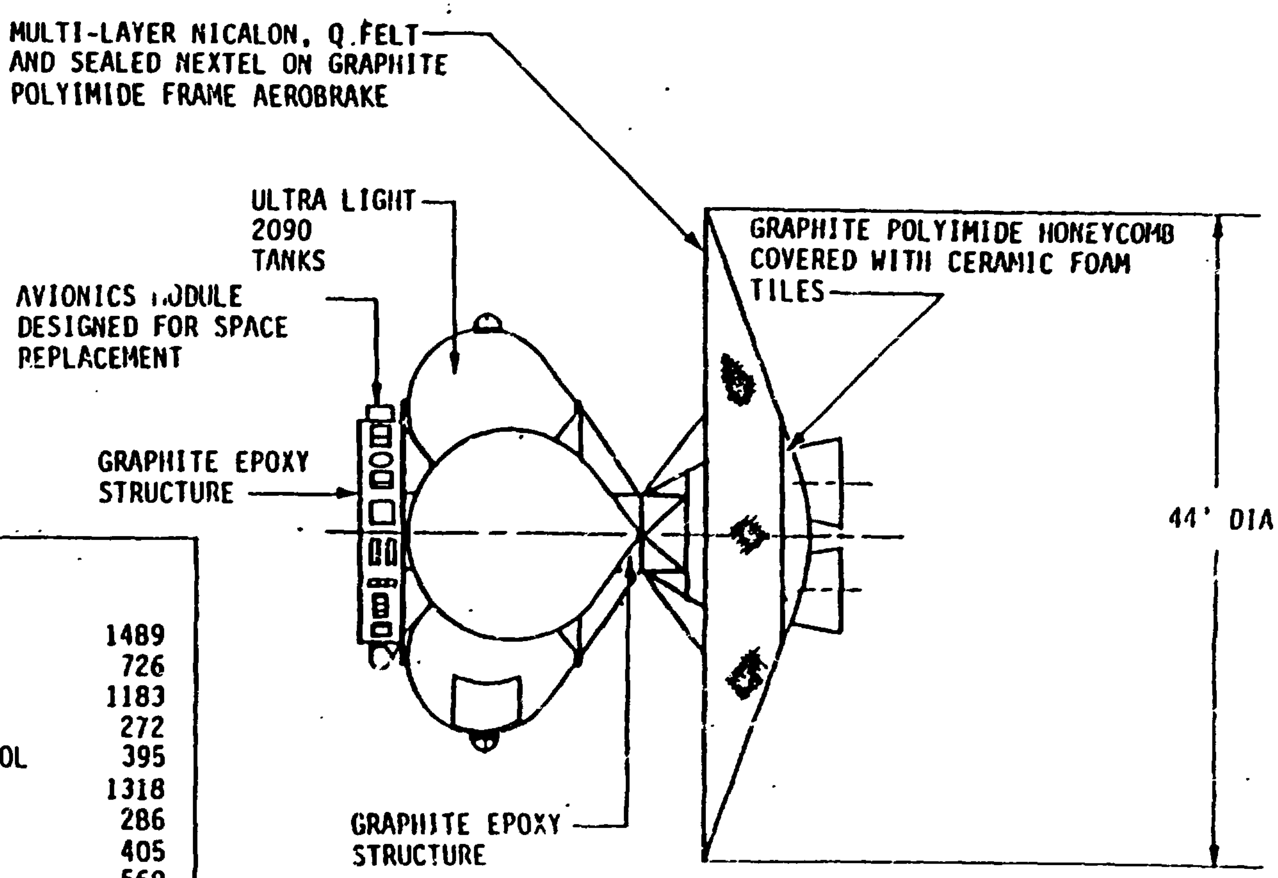
At the appropriate time the OTV does a Hohman transfer to the Space Station orbit.



WEIGHTS

AEROBRAKE	1148
TANKS	524
STRUCTURE	509
ENVIRONMENTAL CONTROL	133
MAIN PROPULSION	904
ORIENTATION CONTROL	199
ELECTRIC POWER	433
AVIONICS	510
CONTINGENCY (15%)	654
DRY WEIGHT	5014
PROPELLANTS, ETC	45349
LOADED WEIGHT	50363

Figure C12-1 Ground-Based Cryogenic OTV



WEIGHTS	
AEROBRAKE	1489
TANKS	726
STRUCTURE	1183
SUPPORT (ASE)	272
ENVIRONMENTAL CONTROL	395
MAIN PROPULSION	1318
ORIENTATION CONTROL	286
ELECTRIC POWER	405
AVIONICS	569
CONTINGENCY (15%)	997
DRY WEIGHT	7640
PROPELLANTS, ETC	91010
LOADED WEIGHT	88650

Figure C12-2 Growth Space-Based Cryo OTV

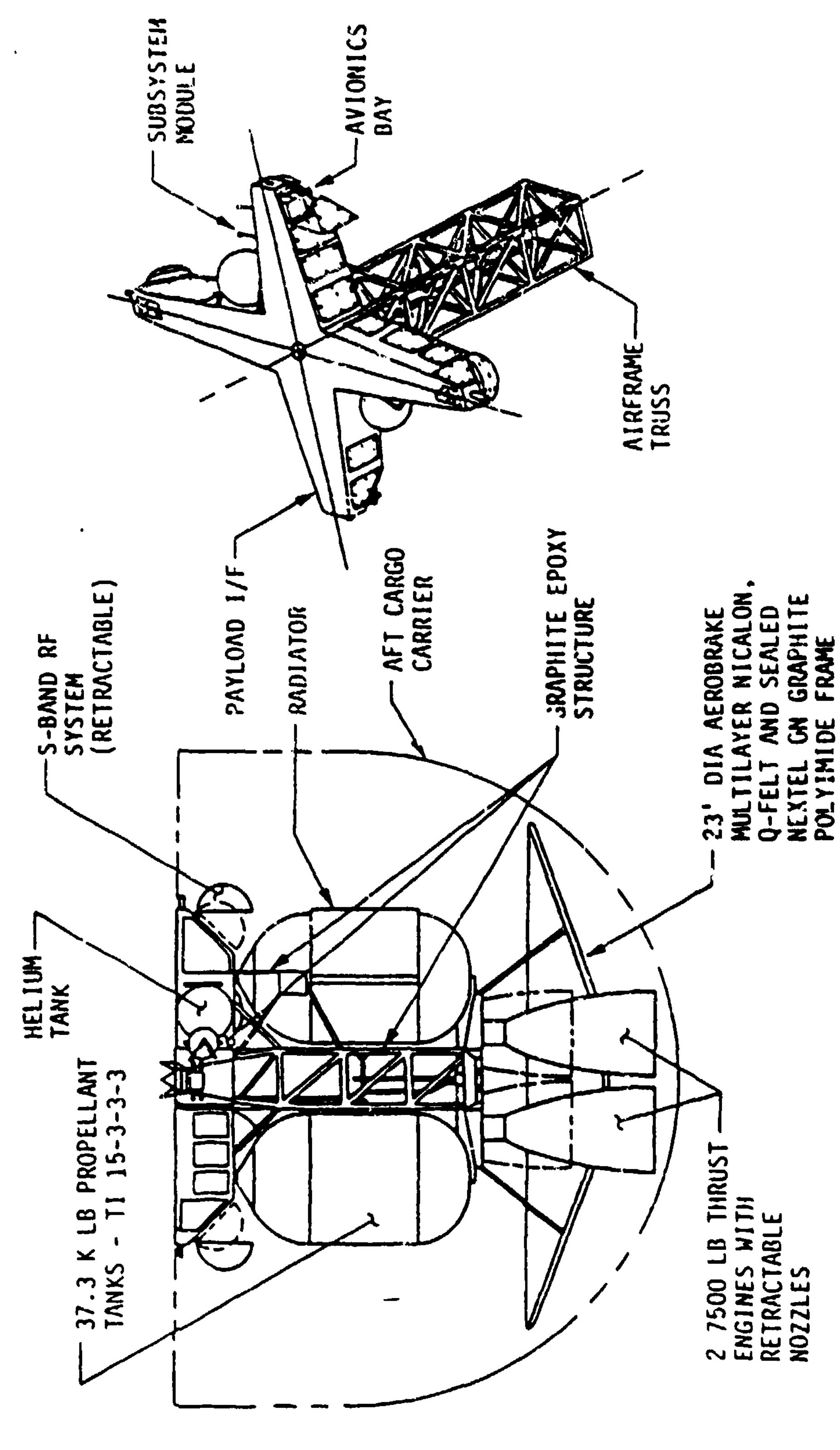


Figure C12-3 Ground-Based ACC-Storable OTV

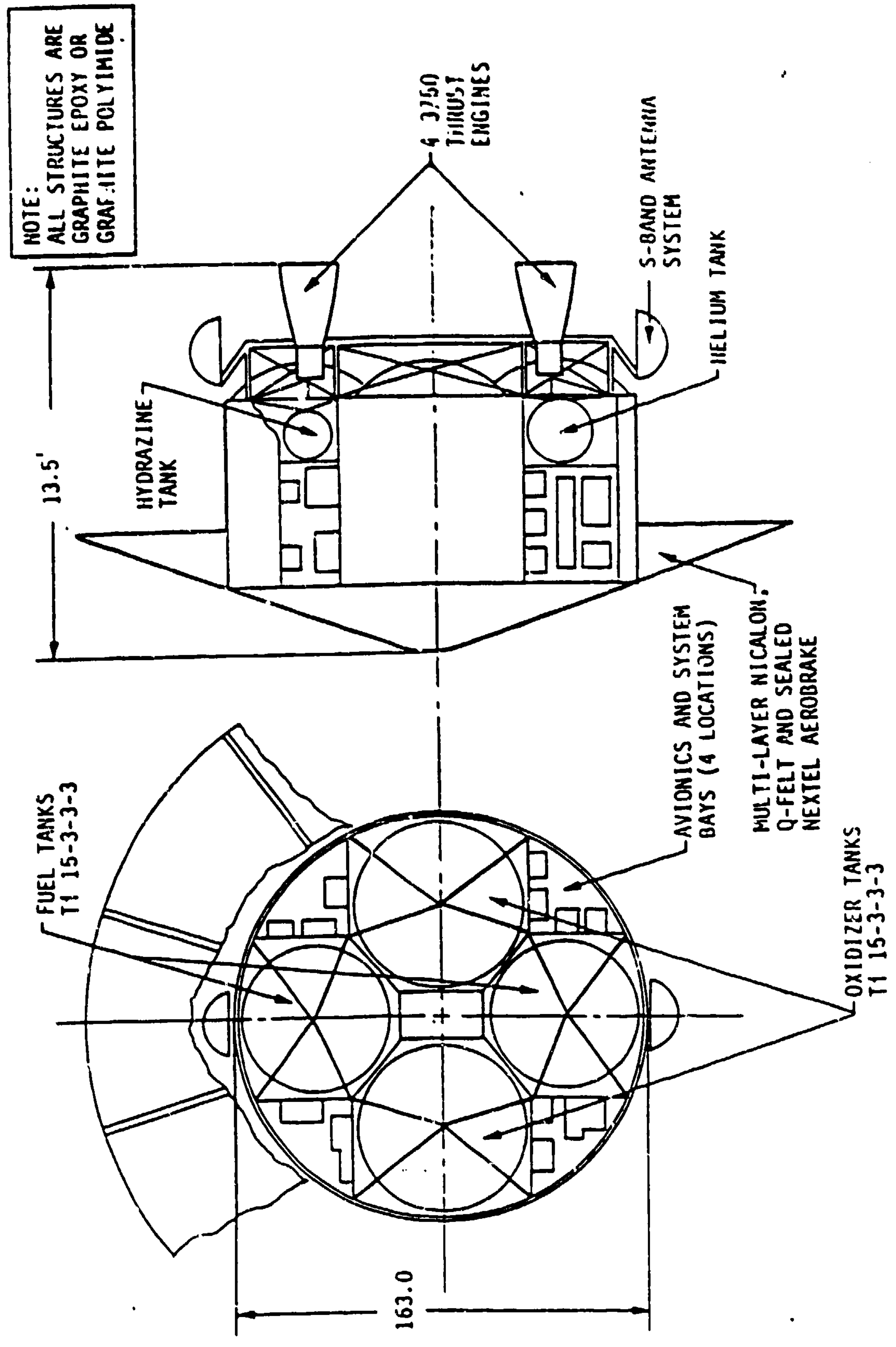


Figure C12-4 37.3K Ground-Based OTV (Orbiter Launched)

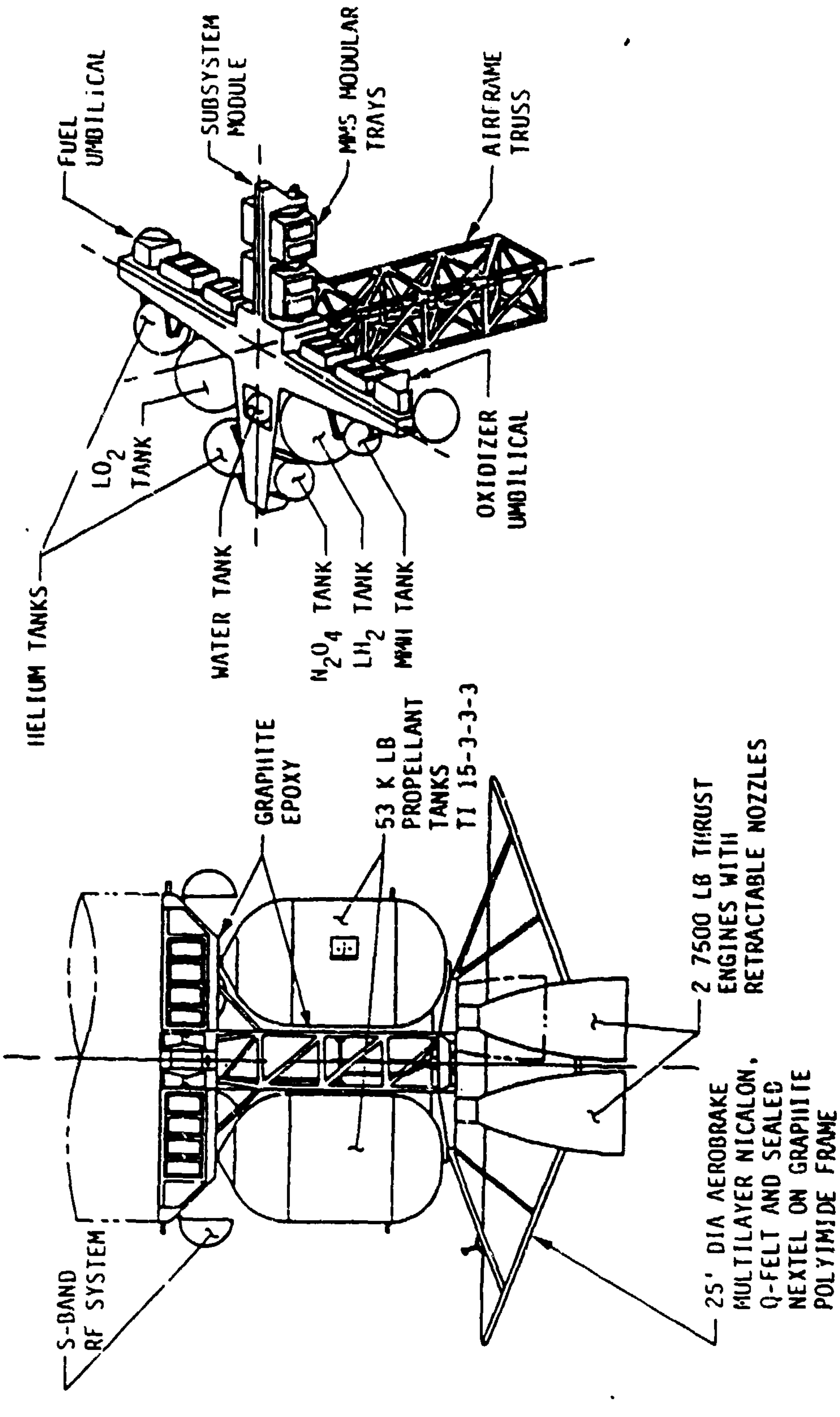


Figure C12-5 Space-Based GEO Delivery Vehicle

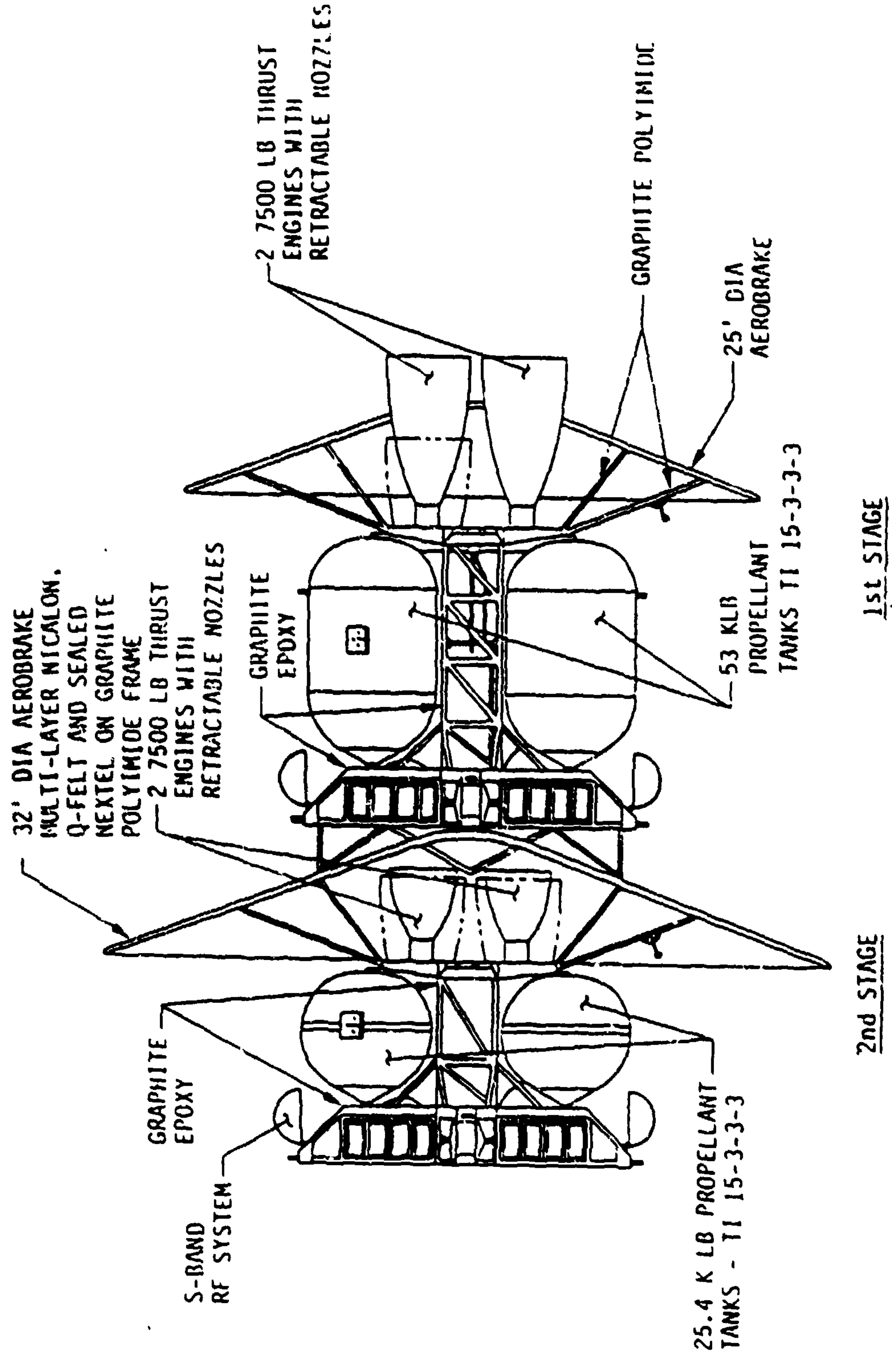


Figure C12-6 Space-Based Unmanned Servicing Vehicle

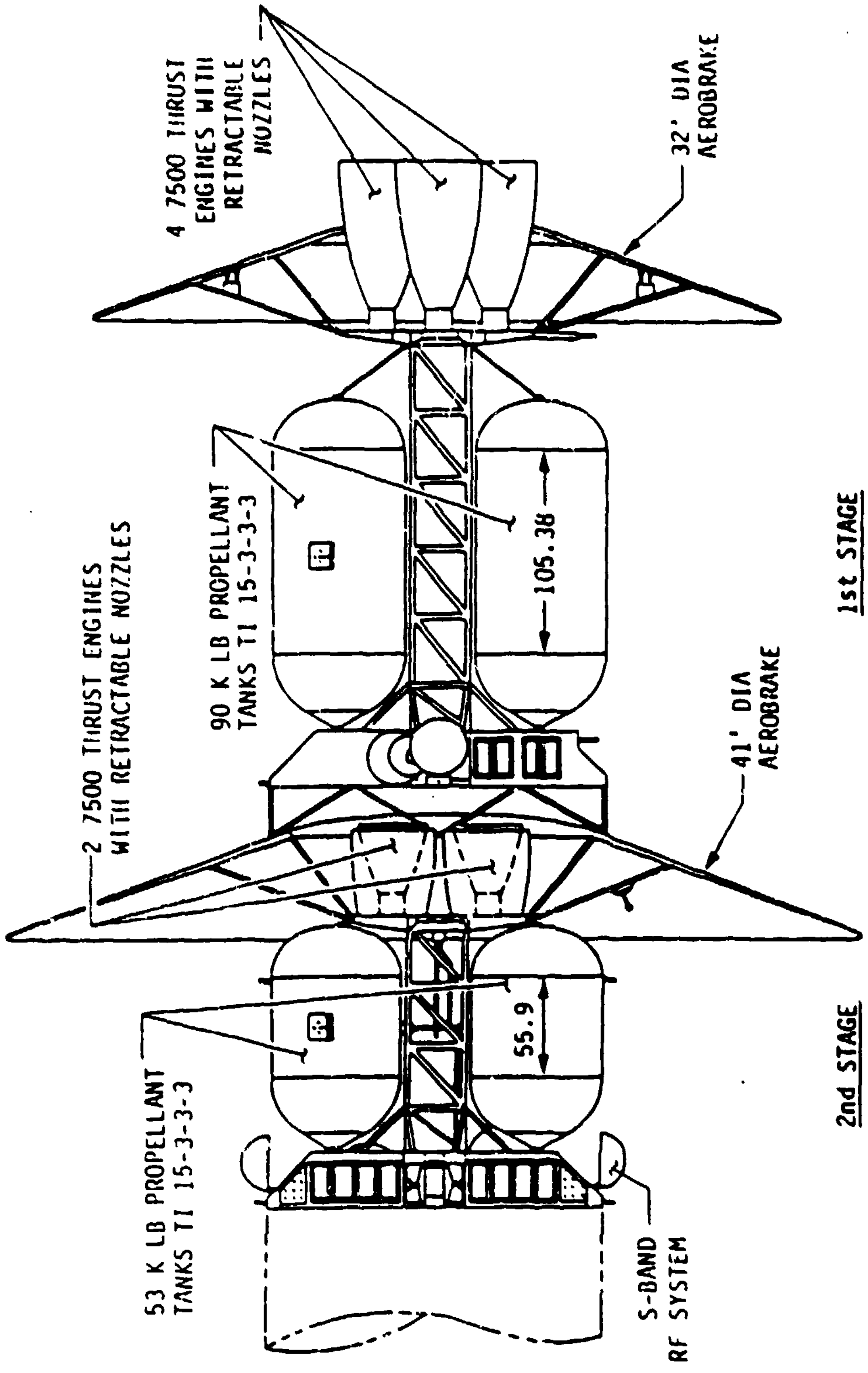


Figure C12-7 Space-Based Manned Servicing Vehicle

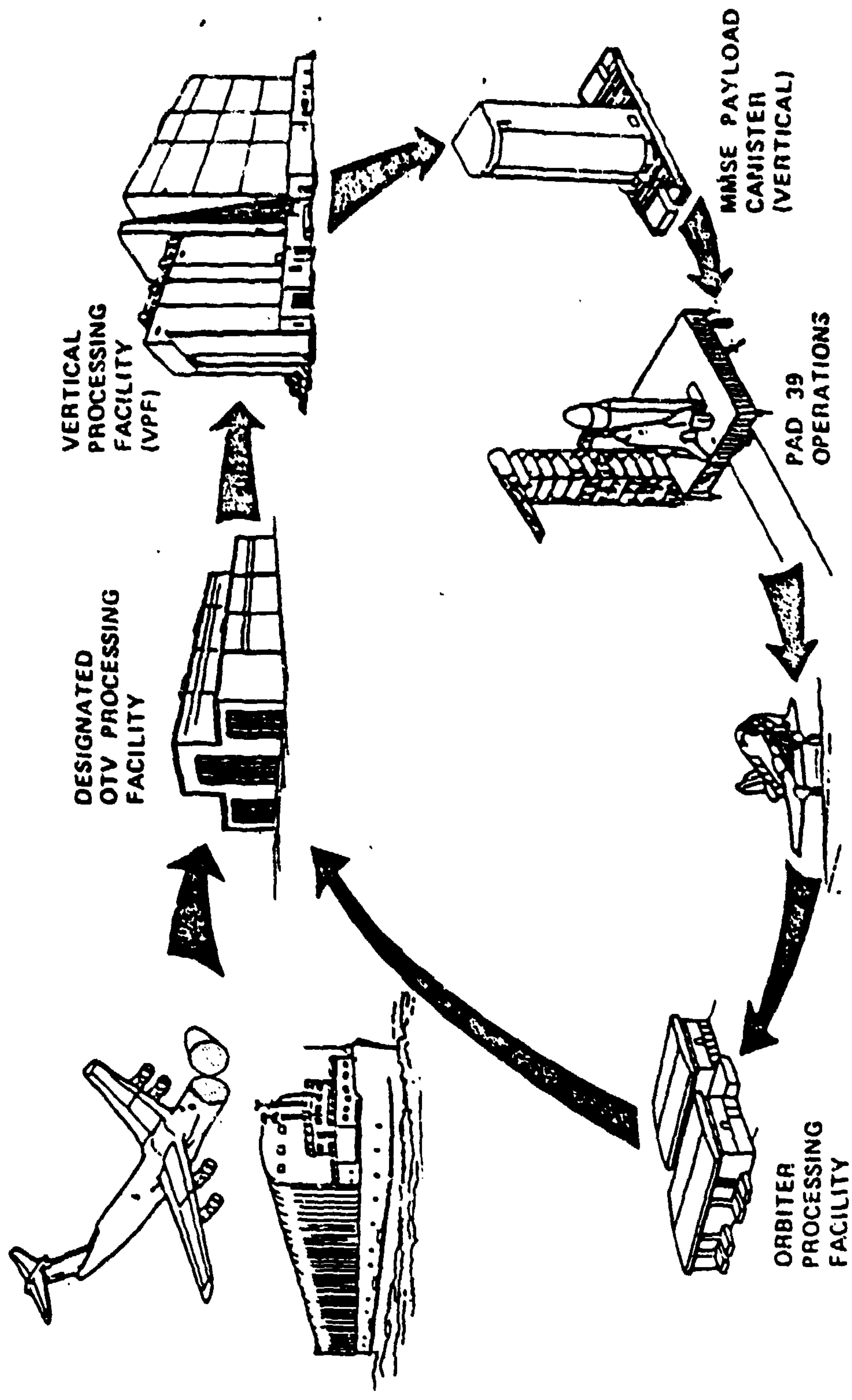


Figure C12-8 Space-Based OTV Ground Processing - Vertical

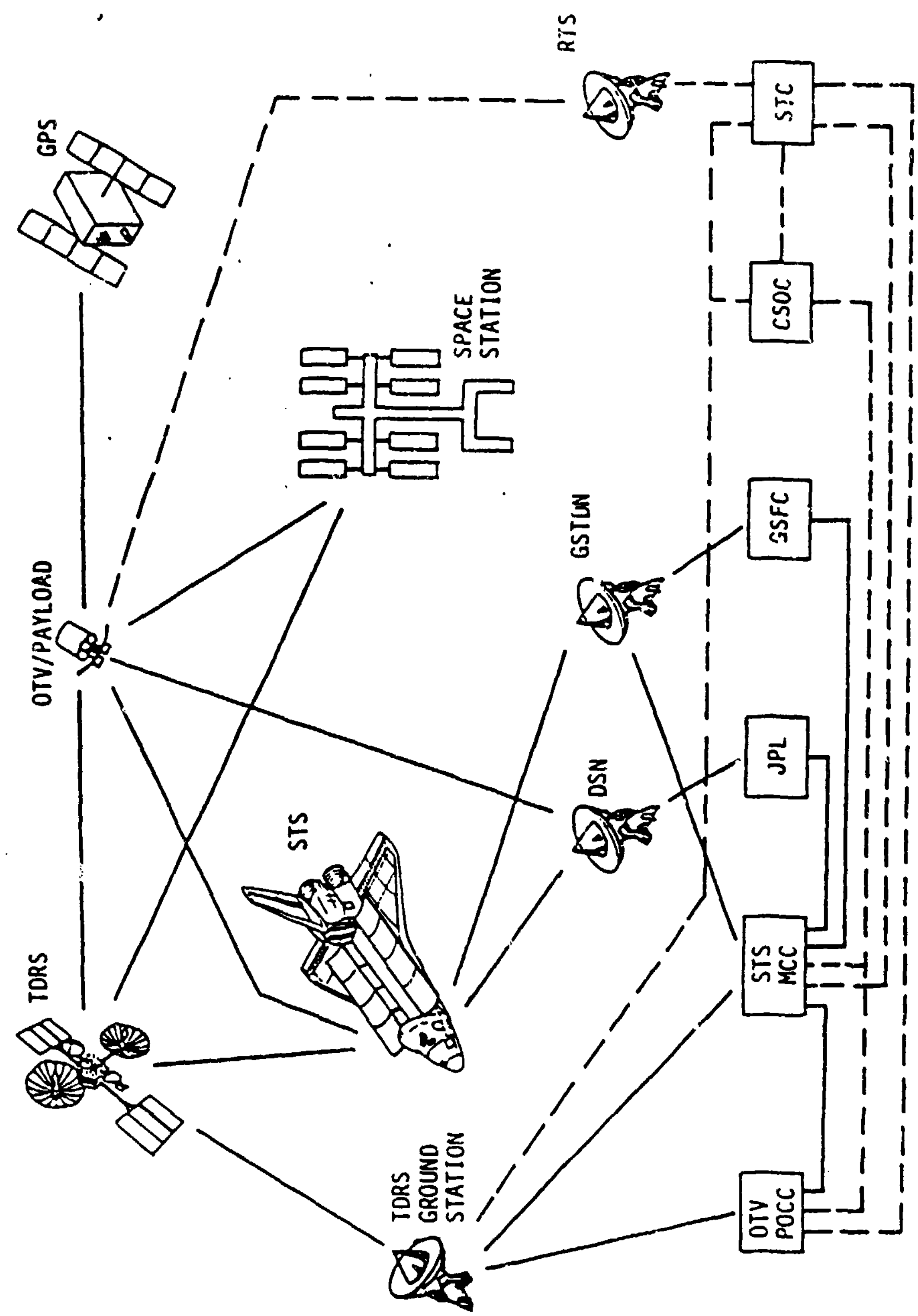


Figure C12-9 OTV Flight Operations Support

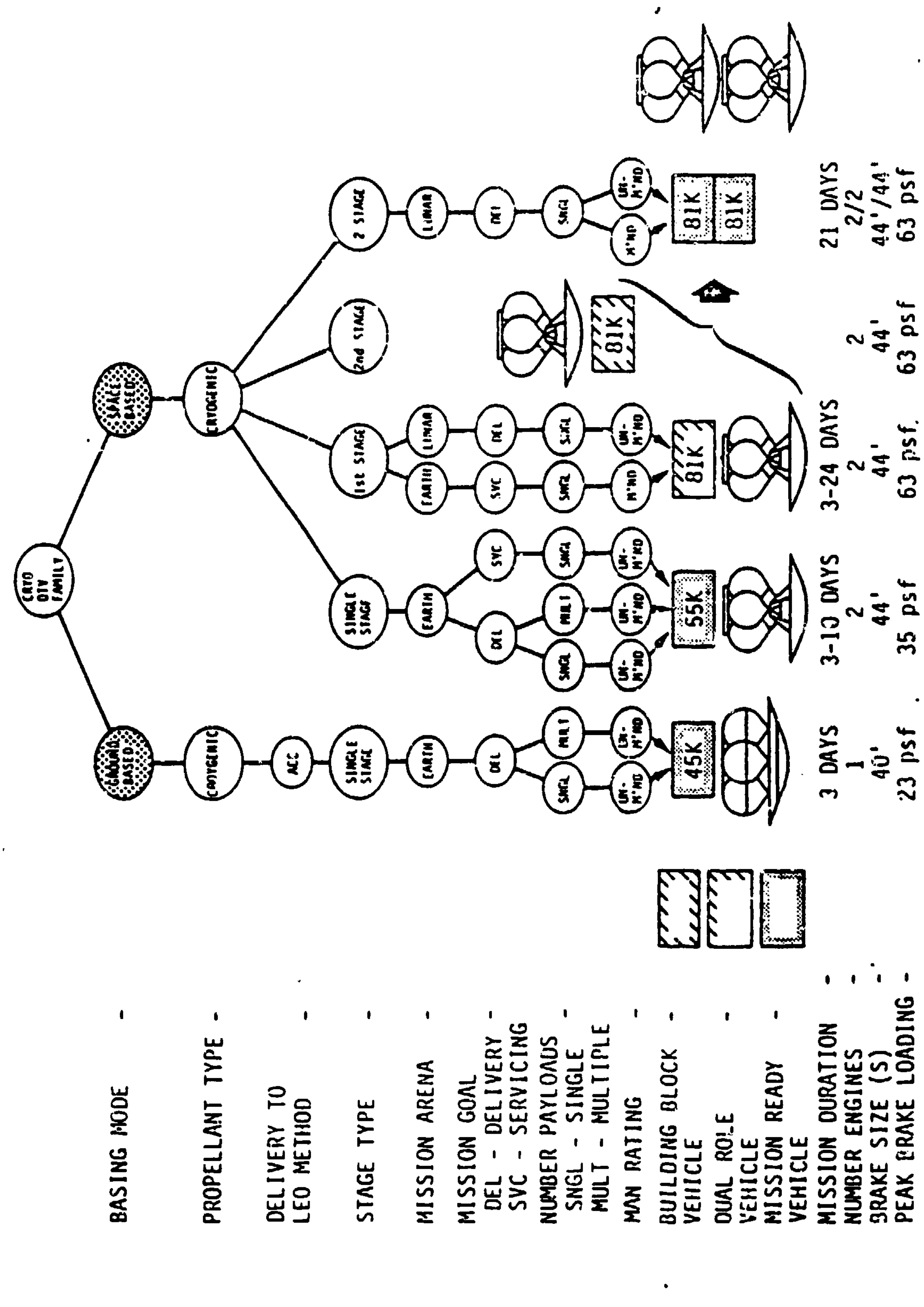


Figure C12-10 Cryo Configuration Summary

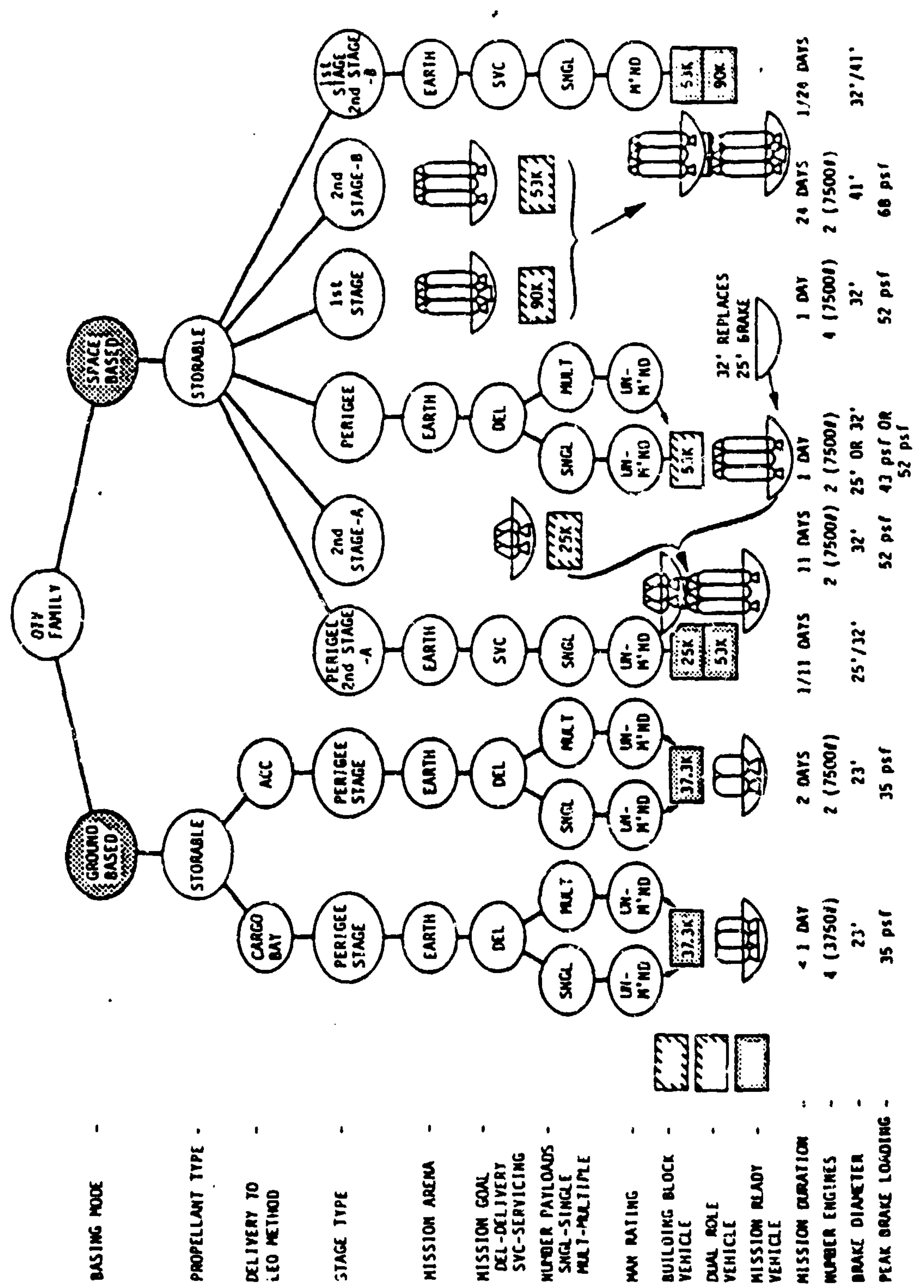


Figure C12-11 Storable Configuration Summary

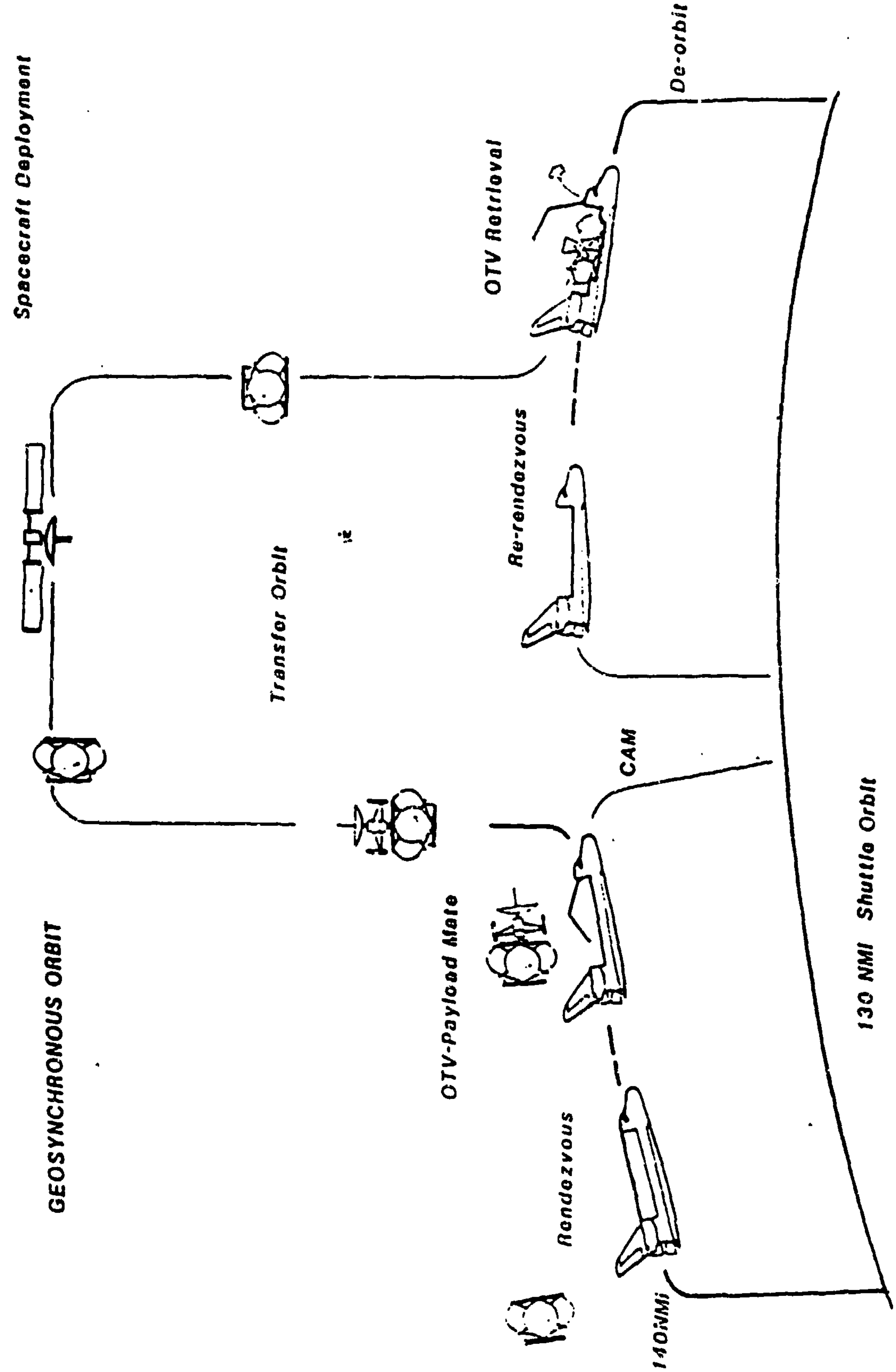


Figure C12-12 Ground-Based OTV Mission

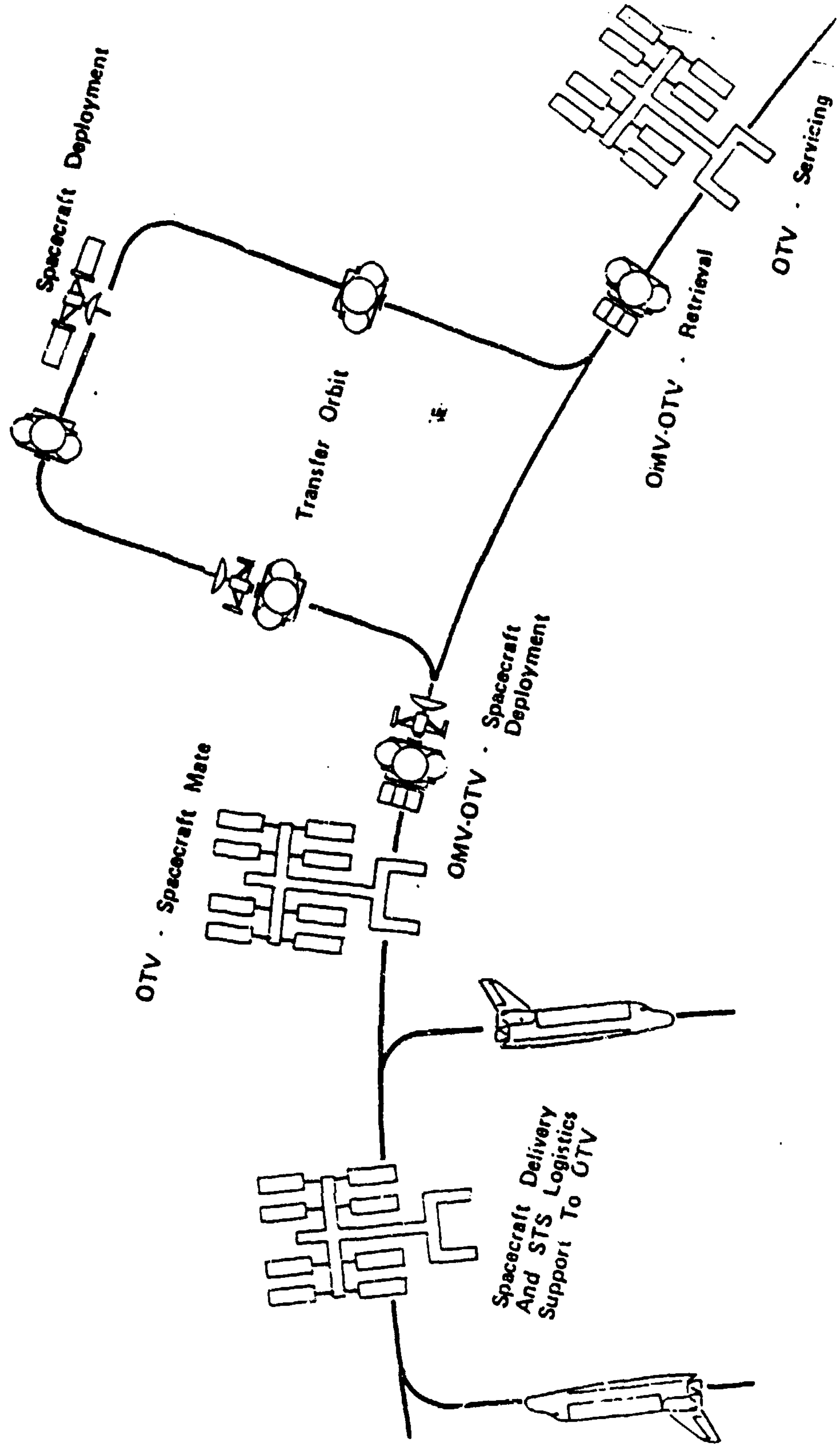


Figure C12-13 Space-Based OTV Mission

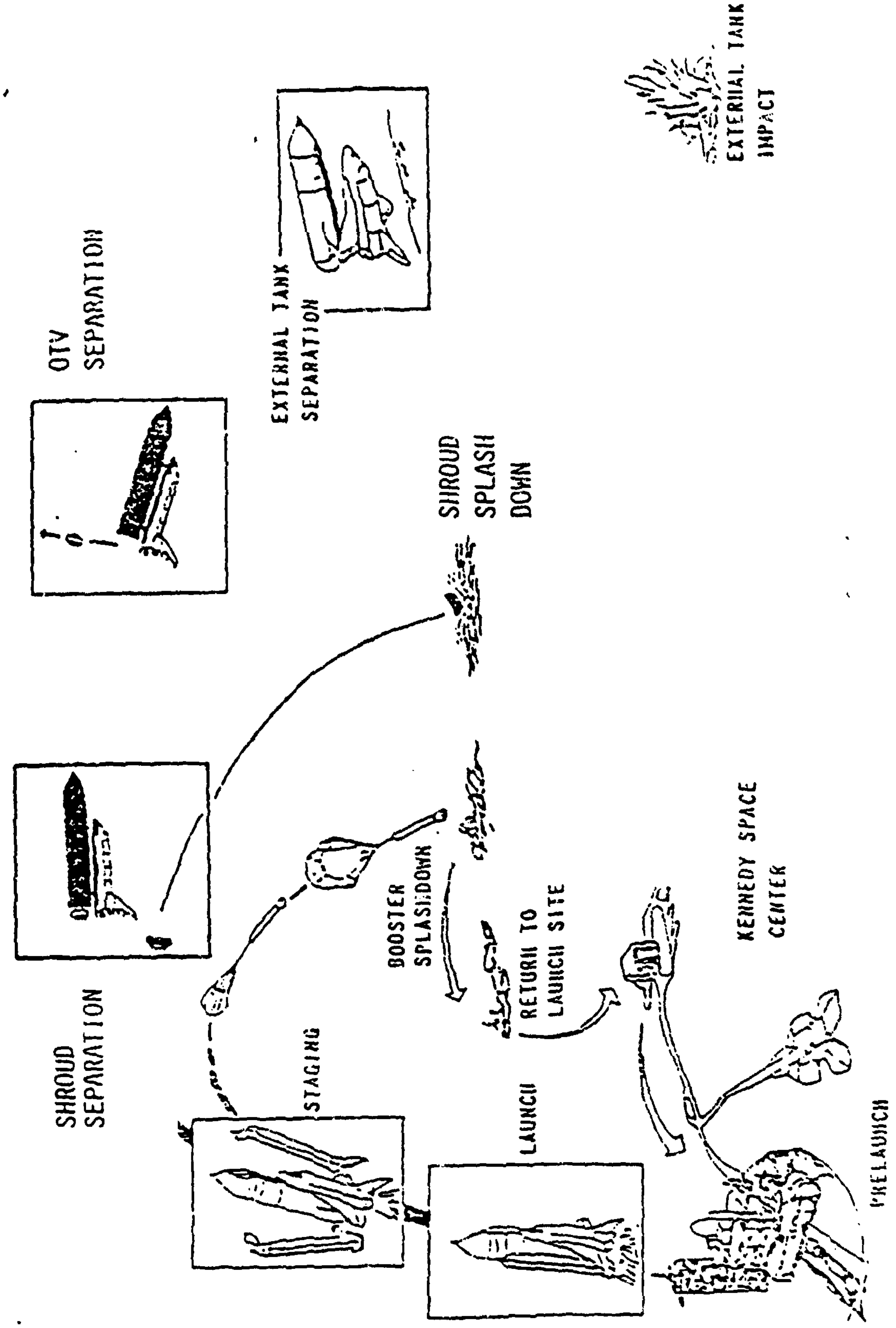


Figure C12-14 Ascent Profile

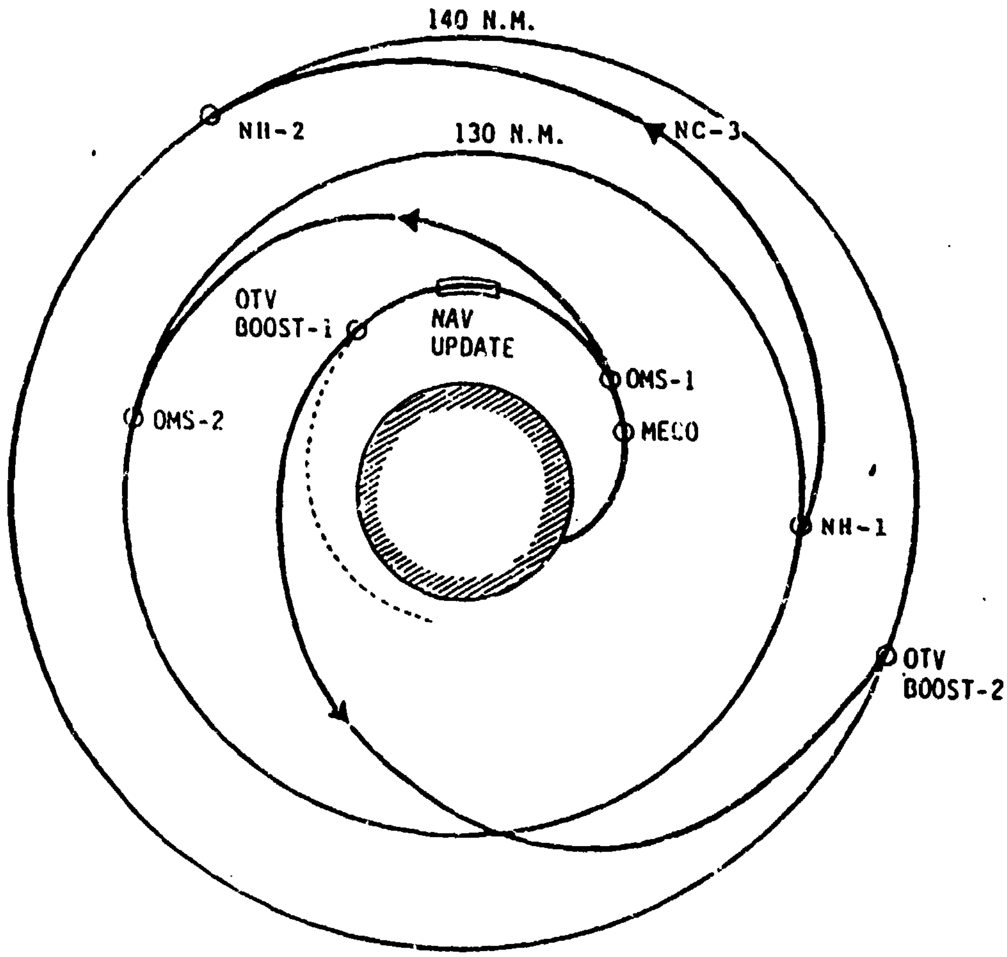
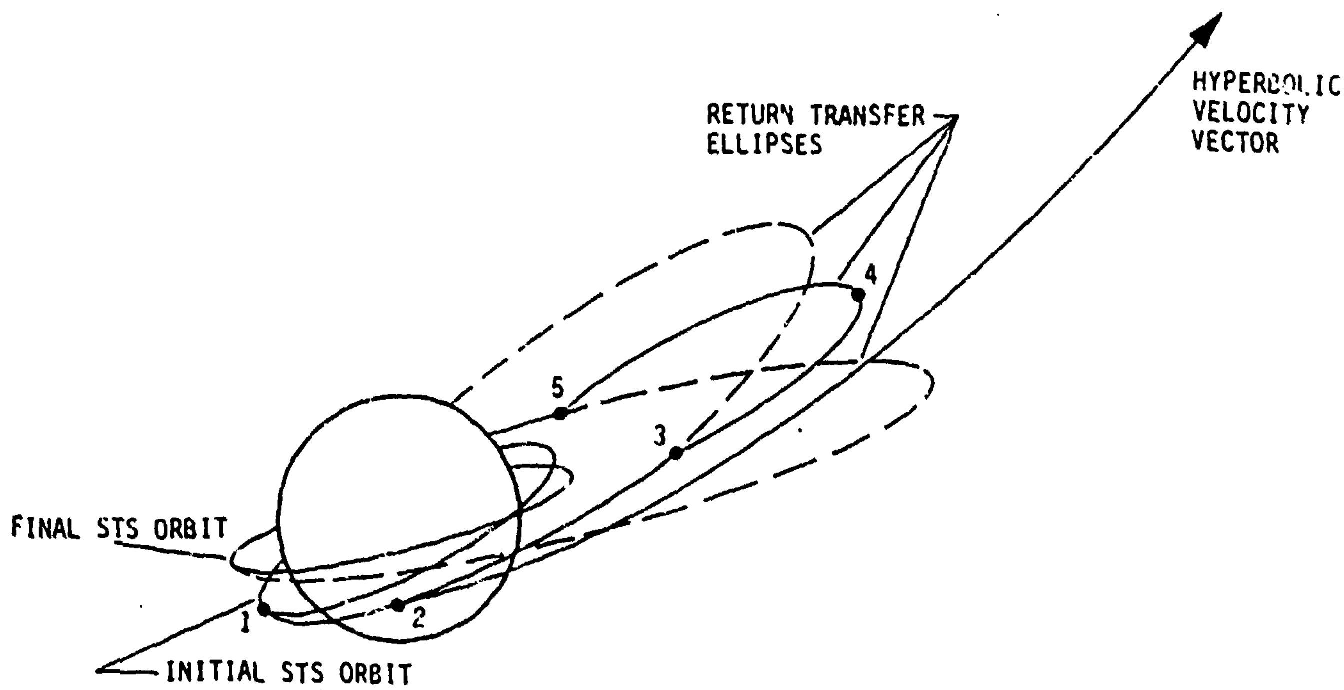


Figure C12-15 OTV/Orbiter Trajectory Plot



BURN: 1) HYPERBOLIC ESCAPE VELOCITY
2) RETURN ELLIPSE VELOCITY
3) ORBIT PLANE PHASING
4) PERIGEE LOWERING
5) ORBIT PLANE PHASING

Figure C12-16 Ground-Based Planetary Mission

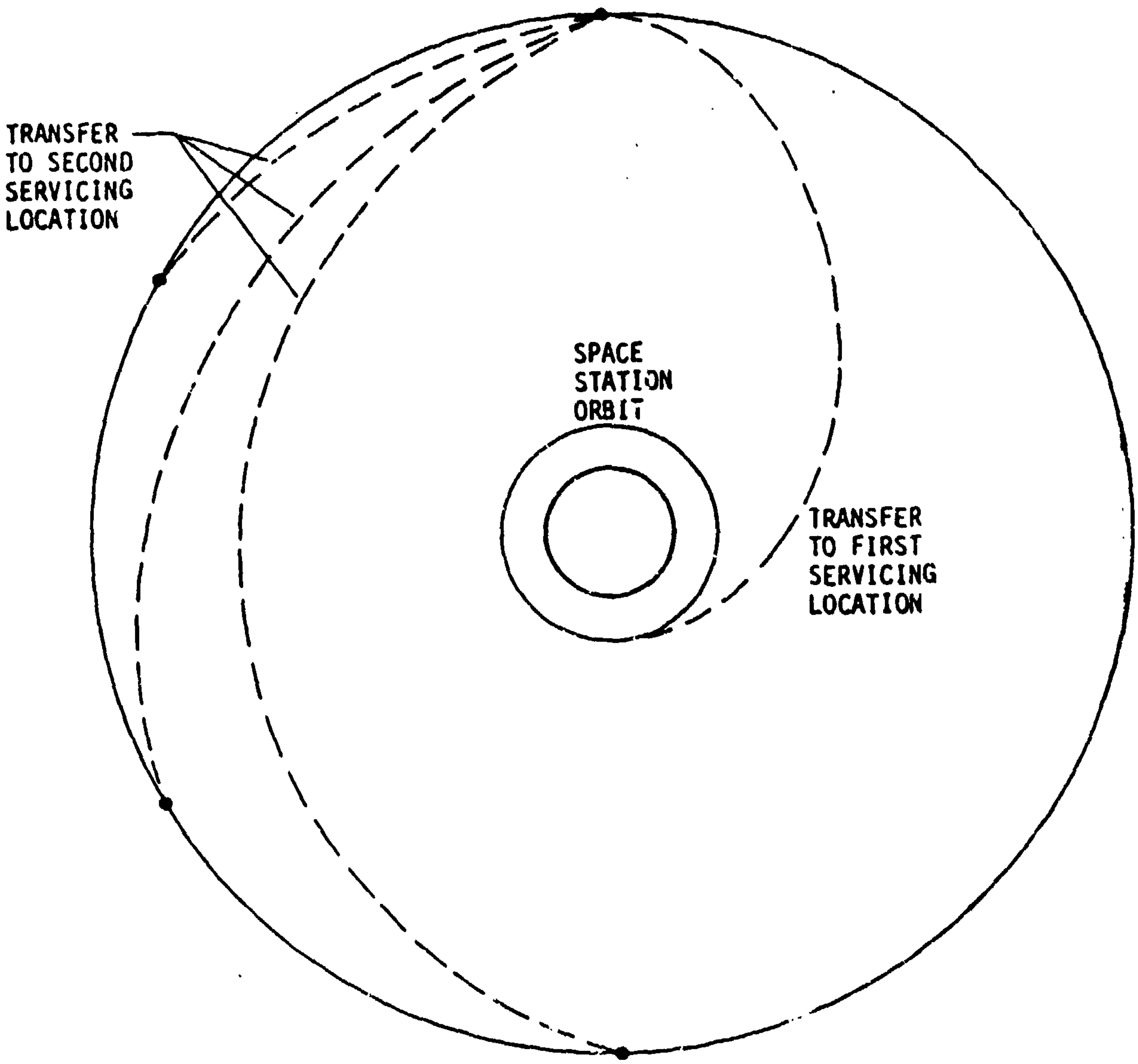
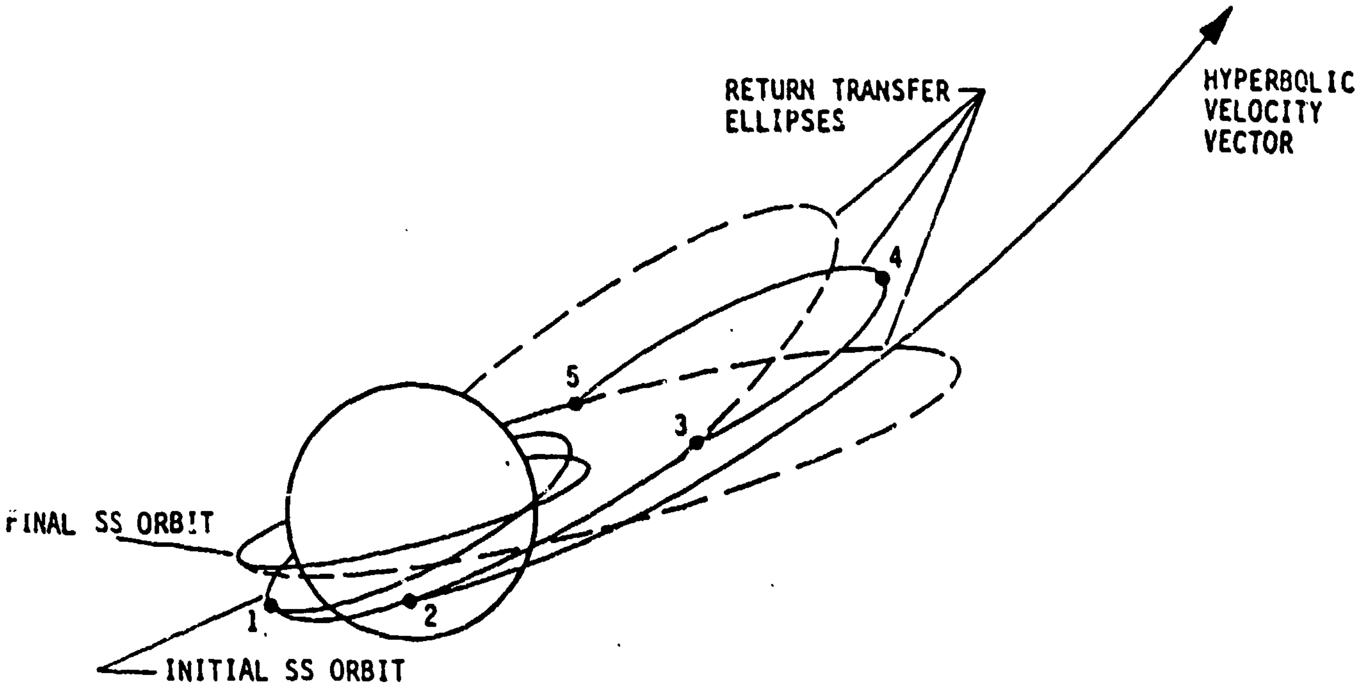


Figure C12-17 Manned/Unmanned Servicing



BURN: 1) HYPERBOLIC ESCAPE VELOCITY
2) RETURN ELLIPSE VELOCITY
3) ORBIT PLANE PHASING
4) PERIGEE LOWERING
5) ORBIT PLANE PHASING

Figure C12-19 Space-Based Planetary Mission

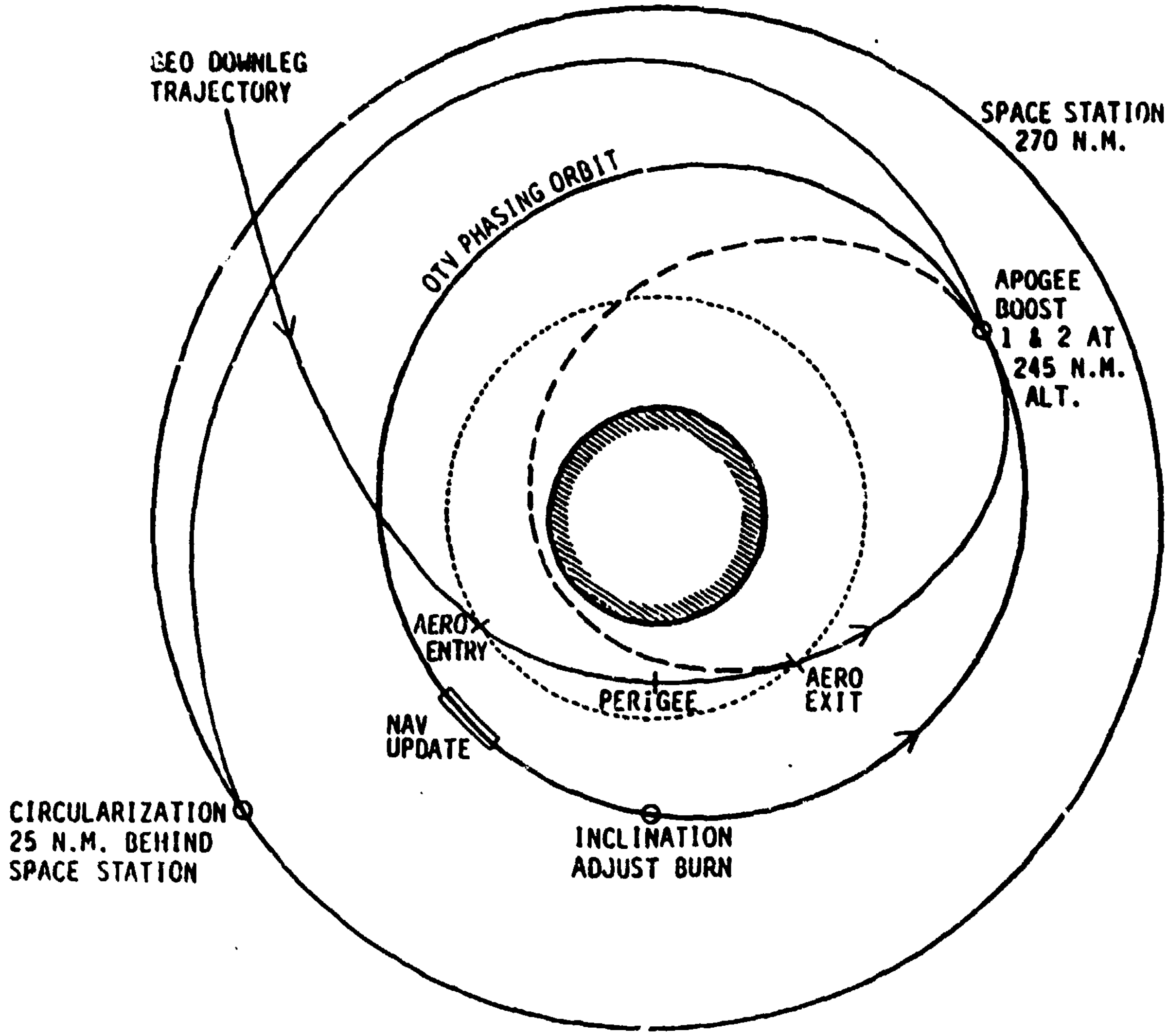


Figure C12-19 Deorbit, Aeropass, and Recovery Orbit

Appendix C13
Payload Assist Module

APPENDIX C13
PAYLOAD ASSIST MODULE - DELTA (PAM-D)

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APPENDIX C13
PAYLOAD ASSIST MODULE - DELTA (PAM-D)

C13.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref. 049.

The Payload Assist Module-Delta class (PAM-D) was developed by MDAC to interface with the STS and meet the needs of the satellite (hereinafter referred to as "spacecraft") users for increased launch vehicle performance.

The PAM-D system flight hardware consists of two major assemblies: the expendable vehicle (EV) and the airborne support equipment (ASE). An exploded view of the PAM-D flight hardware is shown in Figure C13-1.

C13.1 GENERAL DESCRIPTION

C13.1.1 Expendable Vehicle

The expendable vehicle segment of the PAM consists of two main elements:

Payload Attach Fitting (PAF)

Solid Rocket Motor (SRM)

C13.1.2 Payload Attach Fitting (PAF)

The PAF provides the structural mounting of the spacecraft at its forward end and provides the load paths to the SRM and to the ASE cradle assembly while attached to the Orbiter. The PAF is cylindrical and has two triangular fittings that provide the load-carrying paths to the cradle forward restraints. These restraints are retracted to permit spin-up of the PAM and spacecraft before deployment from the Orbiter.

The PAF also provides the mounting accommodations for the PAM functional subsystems, such as the spacecraft separation clamp band, separation springs, pyrotechnic assemblies (safe and arm device, explosive transfer assemblies, spacecraft separation ordnance, and yo-weight system ordnance), batteries, and the electrical control assemblies as portrayed in the block diagram shown in Figure C13-2.

C13.1.3 Solid Rocket Motor (SRM)

The SRM constitutes the propulsive element of the PAM-D stage. It is a Thiokol STAR-48 motor, which comes in different models containing varying quantities of propellant from approximately 3833 to 4410 pounds. The SRM contains through bulkhead initiators (TBIs) that interface with the internal toroidal igniter and provide the means of ignition from the remote safe and arm device via the explosive transfer assemblies (ETAs).

C13.1.4 Airborne Support Equipment (ASE)

The safety requirements specified in Paragraph 216, Reflown Hardware, NBH 1700.7A will apply to the ASE. The major elements of the ASE consist of: cradle structure assembly, spintable assembly, and ASE avionics.

C13.1.5 Cradle Structure Assembly

The cradle is an aluminum structure that mounts in the Orbiter cargo bay at four longeron and one keel attachment points: the cradle supports the PAM-D stage and the spacecraft through the spintable pallet at the bottom and through the forward restraints as shown in Figure C13-1. The forward restraints engage the fittings projecting from the PAF and are the main paths for carrying lateral loads to the Orbiter.

The cradle structure assembly includes a thermal control system (TCS) consisting of (1) thermal blankets and heater panels mounted on the cradle to provide thermal protection for the PAM-D, and (2) a passive sunshield mounted on the cradle to control solar input to and heat loss from the spacecraft when the orbiter bay doors are open. The TCS is shown in Figure C13-3.

C13.1.6 Spintable Assembly

The spintable assembly provides the structural and electrical interface between the cradle and the SRM; provides upperstage spin stabilization using spin system electrical drive motors, and provides Orbiter-to-PAM separation. The assembly includes a rotating cylinder to mate with the SRM using a "V" Block separation clamp band. The assembly also includes a fixed adapter that mates with the cradle, a spin bearing, separation springs, and spin drive system (including spin motors, braking and indexing mechanism, electrical disconnect brackets, and slip-ring assembly). The braking and indexing mechanism provides for reindexing of the PAF in the event of an aborted deployment.

C13.1.7 ASE Avionics

The PAM-D avionics system is shown in a functional layout in Figure C13-4. This system interfaces with the STS Orbiter, the PAM stage, and with the spacecraft and its ASE avionics. The ASE avionics system is designed to be largely self-contained and to impose a minimum requirement on the Orbiter for commands or crew actions. The system provides monitoring of all functions and has the capability for crew override to implement any actions required outside the normal sequencing. The ASE avionics derives raw power from the Orbiter and distributes it to the PAM-D system and to the spacecraft and its ASE.

C13.2 DESCRIPTION OF SAFETY CRITICAL SUBSYSTEMS, HAZARDOUS MATERIALS, SCHEMATICS

C13.2.1 Electrical System Description

The PAM-D electrical system consists of two major subsystems: Electrical Airborne Support Equipment (ASE) and the Expendable Vehicle Electrical System.

The electrical ASE consists of all the reusable hardware elements installed on the cradle and spin table. The electrical ASE performs the following functions:

A. Control and monitor the electromechanical subsystems which mount, protect, condition, and operate the expendable vehicle and spacecraft from lift-off through deployment from the Orbiter;

B. Control and monitor the expendable vehicle electrical systems as required to ready the vehicle for deployment;

C. Provide the power, command, and monitor interfaces with the Orbiter as required for in-Orbiter flight and ground operations of the PAM and Spacecraft systems.

The ASE is designed to be as self-contained as possible, thereby minimizing dependence on Orbiter or crew functions for its operation. Closed-loop automatic operation of most systems is provided by the ASE, while the Orbiter is required to command the start of each sequence.

The Expendable Vehicle electrical system consists of two major subsystems: Sequencing System and Telemetry System. The Sequencing System provides those timed functions required after deployment of the vehicle from the Orbiter.

A. Ordnance system arm

B. PAM motor ignition

C. Spacecraft separation

D. Yo-weight deployment

E. Spacecraft commands

Initiation of the Sequencing System is accomplished by physical separation from the Orbiter. Sequencing is strictly a timed function, with no up-link capability required.

The vehicle telemetry system, which is a mission option, includes a PCM encoder and an S-band transmitter/antenna in order to provide post-deployment data radiated back to the Orbiter or to ground stations.

The PAM electrical system is designed to meet the Orbiter safety requirements through the use of series functional elements in critical circuits, with crew interaction as required. Parallel redundancy is incorporated for mission reliability, with the design objective that no single failure will cause loss of the mission.

C13.2.2 PAM/Orbiter Interface

The interfaces between the PAM and the Orbiter are shown in block diagram form in Figure C13-5. These interfaces are designed in compliance with JSC-14005, Shuttle Orbiter/Payload Assist Module - Delta Class Cargo Element Interfaces.

C13.2.3 Spin Drive, Brake, and Index

C13.2.3.1 Mechanical Subsystem - The mechanical aspects of the PAM spintable subsystem include the following vendor-procured components:

1B99574-1 Brake Assembly (one required)

1B99517-1 Solenoid Index Assembly (two required)

1B99801 Spin Motor (two required)

Each spin motor is a 0.75-HP, 28-Vdc, gear-reduced assembly driving a 5.20-inch-diameter pinion gear through a Sprague-type clutch to drive the 52.0-inch-diameter ring gear attached to the rotating portion of the spintable assembly. The two motors are redundant, operation of either one satisfies the spin-up requirement. The Sprague-type clutch allows the alternate motor to operate independent of a "frozen" armature on the second motor.

The primary functions of the single brake assembly and two solenoid assemblies are to stop the spintable and to index and lock the spintable into the original prelaunch position in the event of an aborted PAM mission after spin-up. The brake is a multiple disc (alternate bronze inner and hardened steel outer discs) electric-coil-activated assembly driven from the spintable by a 5.20-inch-diameter pinion gear bolted to the freely rotating spider end-cap. Current applied to the coil forces the stacked discs together, creating sufficient friction to slow the pinion gear/spider assembly and bring the spintable to a stop. With no current to the brake coil, internal springs release the multiple disc stack, permitting the gear/spider to idle essentially free of frictional torque forces.

Each solenoid index assembly consists of two coils (push, pull) activating a rod end that, when extended, engages slot provisions in the bottom surface of the spintable ring gear. The solenoid assemblies provide talkback for each position (engaged, disengaged). This index provides the correct repositioning of the spintable after de-spin, permitting reinsertion of the payload restraints into the payload attach fitting (PAF). The solenoids are not engaged during initial shuttle launch and orbit.

C13.2.3.2 <u>Spintable Clamp Band</u> - The solid motor is atta[illegible] to the spintable by a two-piece "V" Block clamp band which is secured by two bolts. Separation is effected by actuation of ordnance cutters which sever the two bolts. The clamp band release system is redundant, and cutting of either bolt will permit upper-stage separation. A retention system is provided to retract and retain the clamp in the open position after release.

C13.2.3.3 <u>PAM Spintable Spring Separation System</u> - The spring separation system consists of four self-contained spring [illegible]tuators of 1400 pounds preload. They are bolted to the outside of t[illegible]ward spintable structure. A guided shaft in the actuator acts against a reaction fitting bolted to the solid motor. The spring actuator assemblies are preloaded against the motor push pads after mating.

Upon release of the spintable clamp band, a relative velocity is imported to the PAM expendable vehicle/spacecraft to deploy it from the Orbiter bay at approximately 2.5 ft/s.

C13.2.3.4 <u>Payload Restraint Mechanism</u> - The payload restraint subsystem provides for the restraint of the payload (spacecraft) for launch, boost, reentry, and landing loads.

The complete release of the restraint mechanism allows for the pre-deployment spin-up of PAM. The restraint assembly also provides the capability to resecure the payload after de-spin in case of an abort.

The restraint mechanism consists of two (180-degree opposed) three-link actuating assemblies. Each assembly is actuated by an electromechanical rotary actuator. The actuator is powered by redundant electrical motors coupled to a single actuator output shaft by separate gear trains through a differential assembly. With one motor operating and the other motor stalled or de-energized, the actuator can perform the required operation. Redundant limit switches, positioned at each end of the restraint mechanism stroke, control the linkage travel in both directions. The limit switches also provide an indication of the position of the restraint mechanism, e.g., inserted or retracted. Control and monitor of the sequence of the events that retract the support mechanism and, in case of abort, reinsert the support mechanism, is achieved by the sequence control assemblies (SCAs). In the case of an SCA failure, the backup SCA can control the entire spin abort/resecure system.

C13.2.4 <u>Propulsion Subsystem</u>

The STAR-48 PAM rocket motor, Figures C13-6 and C13-7, incorporates a high-performance solid propellant, a thin-walled titanium case, and a semisubmerged nozzle that includes a 2-D carbon/carbon exit cone and a 3-D carbon/carbon throat insert.

The major motor components include a 6Al-4V titanium case, silica-filled ethylene propylene diene monomer (EPDM) internal insulation, TP-H-3340 hydroxylterminated polybutadiene (HTPB) propellant containing 89-percent total solids and 18-percent aluminum in a head-end web grain, an expansion nozzle that incorporates a toroidal aft-end igniter, and twin initiator assemblies (one redundant) each with a through-bulkhead initiator (TBI). A remote safe-and-arm assembly and twin explosive transfer assemblies (ETAs) are provided with the motor.

Features of the STAR 47 motor include:

A head-end-web grain design;

An 89% solids propellant;

A carbon/carbon nozzle exit cone, high-strength titanium case, and low-density elastomeric insulation;

An aft-end toroidal igniter.

C13.2.4.1 Interfaces - The spacecraft attachment flange on the forward dome provides 24 equally spaced 5/16-24 UNF-3B holes on a 37.456-inch diameter. A "V" Block clamp-band flange for PAM spintable attachment is provided on a 46.100-inch nominal diameter on the aft dome, 41,500 inches aft of the forward dome apex. A handling with a 49.602-inch diameter is located in the case cylindrical section. The motor maximum diameter of 49.910 inches is located at a series of tabs 21.188 inches forward of the nozzle attachment boss.

C13.2.4.2 Case - The case is a 6AL-4V titanium sphere with a 6.5-inch cylindrical section. Its principal diameter is 48.970 inches. The case has integral antenna flanges, yo-weight tabs, and spring tabs located on the exterior of the motor case. The nozzle attachment flange has 44 mounting holes (5/16-24UNF-3B) located on a 17.392-inch diameter circle. A payload attachment flange, located on the forward end of the motor case, provides 24 holes (5/16-24UNF-3B) on a 37.456-inch diameter circle. A "V" Block flange with a 46.110-inch maximum diameter is an integral part of the aft hemisphere. An attachment flange for mounting the case handling ring with a maximum diameter of 49.602 inches is located in the case center section.

The motor case is fabricated from solution heat-treated and aged 6AL-4V titanium. Each hemispherical dome is machined from a closed-die hemispherical forging and the cylindrical section from a rolled-ring forging. The three sections of the case are joined by circumferential welds using the tungsten inert gas process. A single opening is machined in the aft hemisphere to admit the bolt-on nozzle igniter assembly. The tabs and flanges are machined and milled.

Cases are heat-treated to provide a minimum ultimate tensile strength of 165,000 psi and minimum yield strength of 155,000 psi. The case was designed to accommodate 1.25 times the maximum expected operating pressure

(MEOP) of 688 psia. The hydrostatic proof test pressure level for each case is 725 to 735 psig. These design criteria are summarized below:

Hydroproof pressure, equivalent to:

(1.05) (MEOP) at 100°F + 10 psig	725 to 735
MEOP at 110°F, psia	688
Minimum design burst at room temperature, equivalent to (1.25) (MEOP at 110°F), psia	860

C13.2.4.3 Insulation - The rocket motor case is internally insulated with silica-filled EPDM rubber to provide thermal protection for the pressure vessel during motor operation and to maintain the external surfaces of the case below 700°F during thermal soak. The case insulation consists of forward and aft internal insulators.

The aft insulator incorporates a boot to provide stress relief for the propellant grain shrinkage during cooldown from propellant cure temperature and during thermal cycling.

The interior and exterior of the aft closure and toroidal igniter housing components of the nozzle are also insulated with this material.

C13.2.4.4 Propellant and Liner - TP-H-3340 is a composite ammonium perchlorate propellant that uses HTPB fuel binder, isophorone diisocyanate (IPDI) curative, an imine (HX-752) bonding agent, and an aluminum fuel additive.

The line (TL-H-318) has the same elastomeric binder system as the propellant and uses a carbon-black filler. The liner is brush-coated onto the internal surfaces of the insulated case and then cured prior to propellant loading.

C13.2.4.5 Propellant Grain - The STAR-48 rocket motor has an internal-burning, case-bonded, cast-in-place, radial-slotted propellant grain (Fig. C13-8) with a nominal web thickness (including insulation) of 20.47 inches for the fully loaded configuration and 17.97 inches for the off-loaded configuration. A 7-inch section of the fully loaded grain, located just forward of the radial slot, is formed in a 10-point star to provide the surface area to achieve pressure-time neutrality early in operation.

A fully loaded STAR 48 contains 4410 pounds of TP-H-3340 propellant and the off-loaded version for PAM contains 3833 pounds of TP-H-3340.

C13.2.4.6 Nozzle - The rocket motor has a 39.7:1 (short exit cone) or a 54.8:1 (long exit cone) expansion ratio nozzle assembly (Fig. C13-9) with an integral toroidal igniter. Principal components of the nozzle assembly are a 3-D carbon/carbon throat insert, 2-D carbon/carbon exit cone, an exit cone carbon phenolic support insulator, a throat pack retainer and insulators of carbon phenolic, and an aft closure and igniter housing of 6Al-4V titanium. EPDM rubber insulates internal and external titanium surfaces, while the exit cone is covered with carbon-felt insulation. As shown in Figure C13-10, the toroidal igniter chamber is formed by the igniter housing and exterior of the insulated aft closure. Exhaust gases flow through twelve silica-phenolic exhaust ports from the toroid chamber onto the motor propellant grain.

C13.2.4.7 Initiator Assembly - Two initiator assemblies, one of which is redundant, are used to ignite the toroidal igniter propellant. The initiator uses a TP-H-3174 propellant grain bonded into a paper-phenolic tube. The body and headcap are made of titanium (6Al-4V). The initiators thread into the aft closure. Initiator exhaust gases are channeled into the toroidal igniter by a small stainless-steel tube.

C13.2.4.8 TBI and ETA - A through-bulkhead initiator (TBI) ignites each initiator assembly. The TBI contains PETN as donor and receptor charges and a pyrotechnic output charge. It mates to the explosive transfer assembly on one end and to the headcap of the initiator assembly on the other.

C13.2.5 Pyrotechnics Subsystem

The pyrotechnics subsystem is composed of the following elements:

A. Safe and Arm Device: Provides protection against inadvertent activation of the explosive train used to ignite the SRM and, conversely, ensures the timely ignition of the SRM when required.

B. Bolt Cutter Assembly: The bolt cutter assemblies have two applications:

   1. To release the clamp band assembly that secures the PAM to the spintable assembly for separation of the PAM from the STS (ASE).

   2. To release the clamp band assembly that secures the spacecraft to the PAM to provide for separation of the spacecraft.

C. Cable Cutter Assembly: To release the cable, allowing the yo-weight to deploy after spacecraft/PAM separation to tumble the expended PAM.

C13.2.5.1 Safe and Arm Device, P/N E29609 - The PAM S&A device, Thiokol P/N #29609, is used in the PAM solid motor ignition system. It is remotely located, and motor ignition takes place via redundant explosive transfer assemblies (ETAs) that initiate through bulkhead initiators (TBIs) installed in the pyrogen igniter in the aft end of the motor.

Table C13-1 - PAM-D Non-Electric Ordnance Devices

Name: PAM-D Solid Rocket Motor (SRM)
Part Number: 1B98497
Military Hazard Classification: 1.3
Military Storage Compatibility: B
DOT Classification: B
Propellant Composition and Weight:

Propellant	Weight
TP-H-3340	3833 to 4410 lbs

Quality Per Vehicle: 1
Manufacturer: Thiokol

Name: Through Bulkhead Initiators (TBI)
Part Number: Thiokol E25446-02
Military Hazard Classification: 1.4
Military Storage Compatibility: B
DOT Classification: C
Explosive Composition and Weight:

Explosive	Weight
PETN	0.130 grams
Titanium Cupric Oxide	0.610 grams

Quantity Per Vehicle: 2
Manufacturer: Teledyne McCormick Selph

Name: Explosive Transfer Assemblies (ETA)
Part Number: Thiokol E31115
Military Hazard Classification: 1.3
Military Storage Compatibility: B
DOT Classification: C
Explosive Composition and Weight:

Explosive	Weight
HNS-I, Grade A	62-68 Milligrams
HNS-II, Grade A	31-39 milligrams

Quantity Per Vehicle: 2 Long/2 Short
Manufacturer: Teledyne McCormick Selph

The S&A explosive train consists of two electrical detonators, two RDX transfer leads, and one crossover charge (consisting of a length of MDF with PETN-filled stainless-steel cups secured to each end of the MDF). The crossover charge provides a crosstrain for redundancy. The S&A is an electro-mechanical device that prevents inadvertent motor ignition by providing mechanical interruption of (1) the electrical circuit, and (2) the explosive train. In the safe position, the firing circuit is open (see Fig. C13-11) and the detonator leads are shorted together and grounded to the case. The explosive train interruption is provided by a mechanical barrier resulting from the rotation of the RDX transfer leads out of alignment with the detonators. The rotary switch and mechanical barrier (rotor assembly) are mounted on an integrated system of gears and clutch-driven shafts. Motive force is provided by a reversible DC motor. Additional S&A characteristics are indicated as follows:

A. The S&A can only be armed electrically but can be safed either manually or electrically.

B. A safety pin is provided to prevent inadvertent arming of the S&A. With the safety pin in place and the S&A electric motor energized, the pin is locked in place to prevent arming and to prevent pin removal.

C. A visual indicator and monitor circuits are provided to indicate "SAFE" or "ARM" status.

D. The S&A will not fracture or crack when initiated in either the safe or armed position.

E. All fragments of the explosion are contained within the S&A housing when initiated.

The device is rated at 1 amp - 1 watt for five minutes. In addition, it includes a safety design feature of protection for 25,000 volts electrostatic discharge (ESD), and each detonator is tested in accordance with the ESD procedure set forth in MIL-STD-1512, without the series 5,000 ohm resistor and with the pins shorted. The device satisfies the intent of USAF range safety requirements for S&A devices as specified in Paragraph 3.6.9.7 AFETRM 127-1 and NASA's requirements specified in Paragraph 5.3.3, JSC 08060.

A comparative investigation reveals that the detonators used in the S&A device are similar to NASA's standard initiator (NSI-1). The header, bridgewire, and initiating charge cup design, materials, and construction are identical to the NSI-1. Identical procedures for assembly and loading of the pyrochemical materials are used. The case configuration and adaptation of the output detonating charge are modified to facilitate internal installation in the S&A device. The electrical characteristics are also identical to the NSI-1.

C13.2.6 Structures Subsystem

The structural subsystem is comprised of two major elements:

A. The airborne support equipment that is flown, returned, and refurbished for subsequent missions. The cradle assembly is the primary structural element between the STS and PAM/spacecraft with the forward restraint system providing support for launch, boost, entry, and landing. The spintable assembly is the mating point of the PAM/spacecraft to the STS and induces spin stabilization of the PAM and separation from the STS.

B. The expendable vehicle is the solid rocket motor (SRM) and the payload attach fitting. The payload attach fitting is the interface between SRM and the spacecraft. The PAF also contains the reaction fittings that attach to the cradle assembly for transfer of internal loads.

C13.2.7 Thermal Control System

The PAM system (expendable vehicle and ASE) will be exposed to the external environment after the Orbiter payload bay doors are opened in STS orbit. Exposure of the PAM system to such an environment has led to the need for a thermal control system (TCS).

C13.2.7.1 Cradle-Mounted Thermal Control System (CTCS) - The purpose of the CTCS is to keep the solid motor, the electrical equipment in the ASE compartment, and the PAM vehicle at a proper temperature level. The system consists of multilayer insulation (MLI) blankets and heater elements attached to the inside of the MLI blankets. The MLI blankets enclose the cradle.

C13.2.7.2 Cradle-Mounted Sunshields - Two cradle-mounted sunshields are available to provide thermal protection to the spacecraft when the payload bay doors are open. The first, designated the "standard" sunshield, has flown on early PAM-D missions. The second, designated the "large" sunshield, is adjustable for spacecraft between 103 and 115 inches in diameter.

The two sunshields are functionally identical, and the following description is applicable to both.

The sunshield will be in the open position when the payload bay doors are closed and will be closed after these doors are opened on orbit, except during the time that PAM is being readied for deployment. The sunshield will be opened just prior to spin-up and closed immediately after PAM is deployed.

Multilayer insulation blankets of the same construction as previously described for the CTCS are used on the sunshield to provide passive thermal protection to the spacecraft and PAM components.

The normal positions for the sunshield are:

Open during orbiter ascent;

Closed within 25 minutes after orbiter payload bay doors are opened;

Open again before deployment of expendable PAM-DII/spacecraft;

Closed again after deployment;

Open again before orbiter payload bay doors are closed for reentry.

C13.3 MISSION SCENARIO

The STS PAM-D system provides the necessary injection velocities to place payloads (spacecraft) into their required orbits above the low-altitude parking orbit attainable from the STS. The nominal STS PAM-D flight sequence of events will be initiated at a preplanned time in the STS mission. This selected time will depend on the spacecraft requirements for perigee location and the requirements for other cargo scheduling. When the selected point of deployment is approached, a typical first-mission sequence of events, tabulated below, will be initiated.

Table C13-2 - PAM-D Mission Timeline

Time (min-sec)	Event
T - 60:00	Turn on STS PAM-D power
T - 45:00	PAM-D system status check complete
T - 40:00	Start maneuver to deployment attitude
T - 28:30	Achieve PAM/spacecraft deployment attitude
T - 15:00	Start mechanical sequence
T - 12:00	Start spin sequence
T - 11:30	Spin-up complete and verified
T - 3:00	Start terminal sequence
T - 0	Deploy PAM Initiate sequencing system by separation switches
T + 45:00	Achieve separation distance required by STS
T + 45:00	PAM motor ignition
T + 46:28	PAM motor burnout
T + 48:20	Fire PAM/spacecraft separation bolt cutters
T + 48:23	Fire tumble system yo-weight bolt cutters

The STS PAM-D ignition time is 45 minutes after separation, which results in a coast phase of about one-half an orbit period.

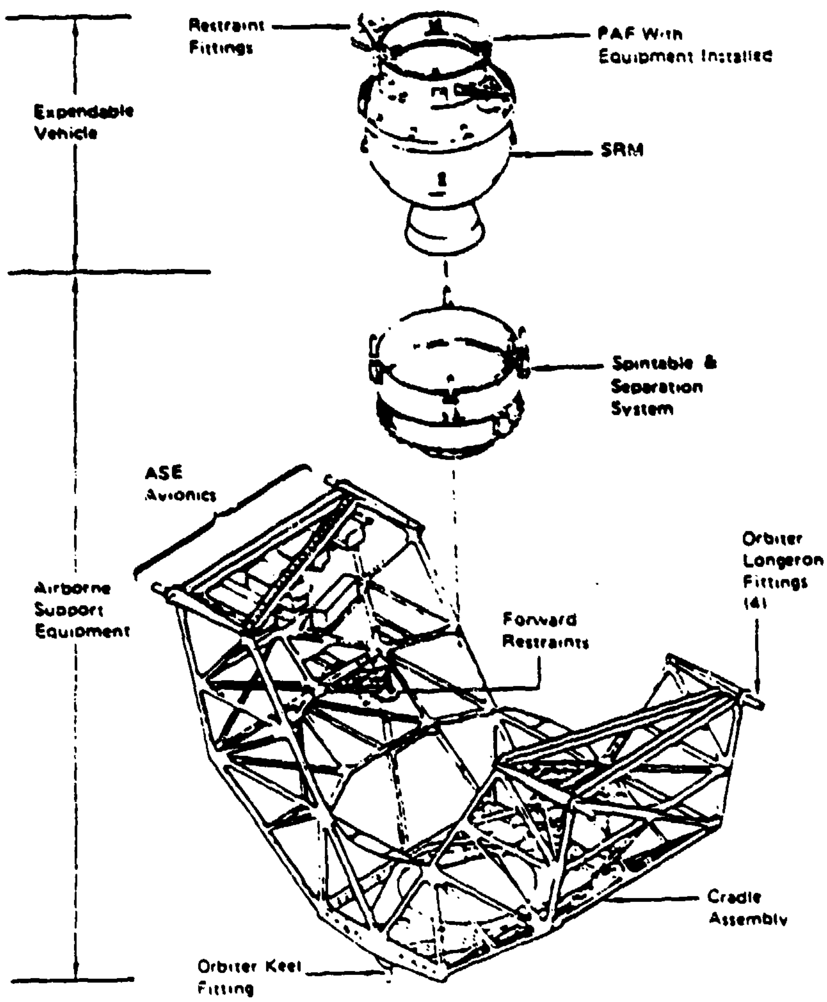

Figure C13-1 STS PAM-D System Hardware

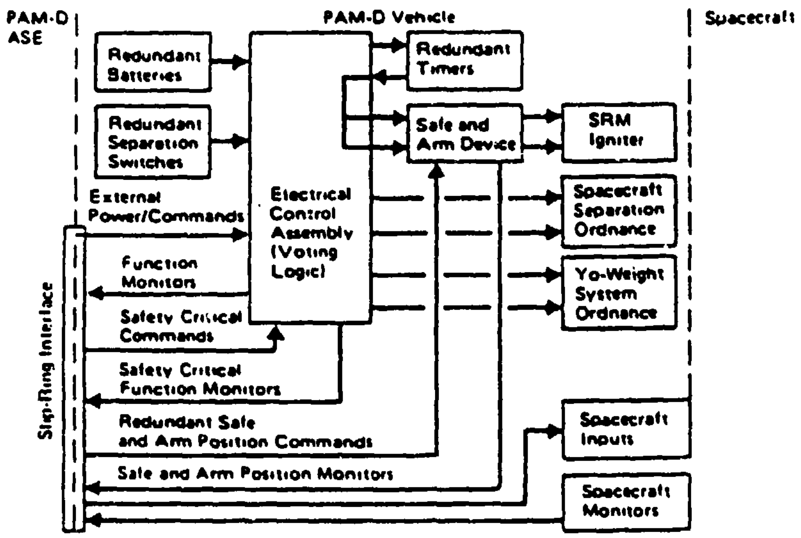

Figure C13-2 STS PAM Payload Attach Fitting Block Diagram

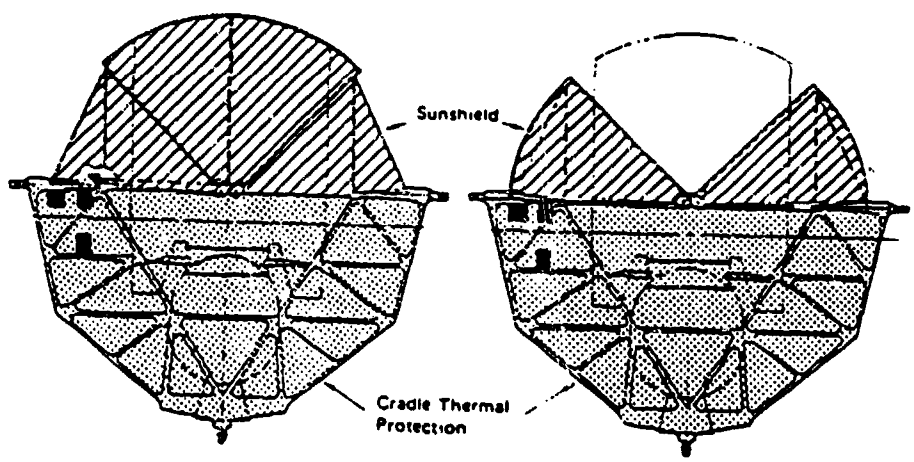

Figure C13-3 PAM-D Thermal Control System

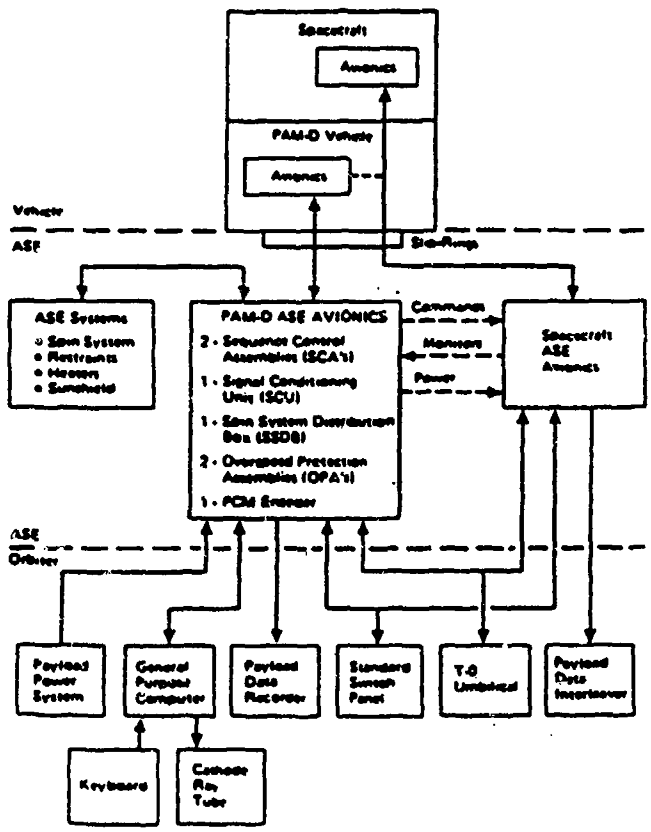

Figure C13-4 PAM-D Avionics

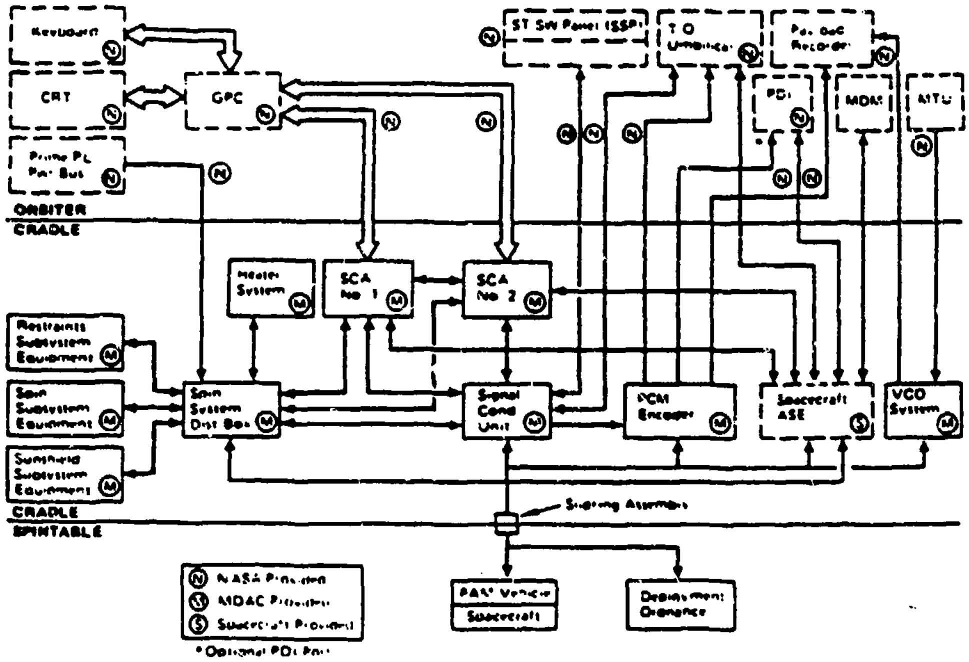

Figure C13-5 Block Diagram of ASE and Interfaces

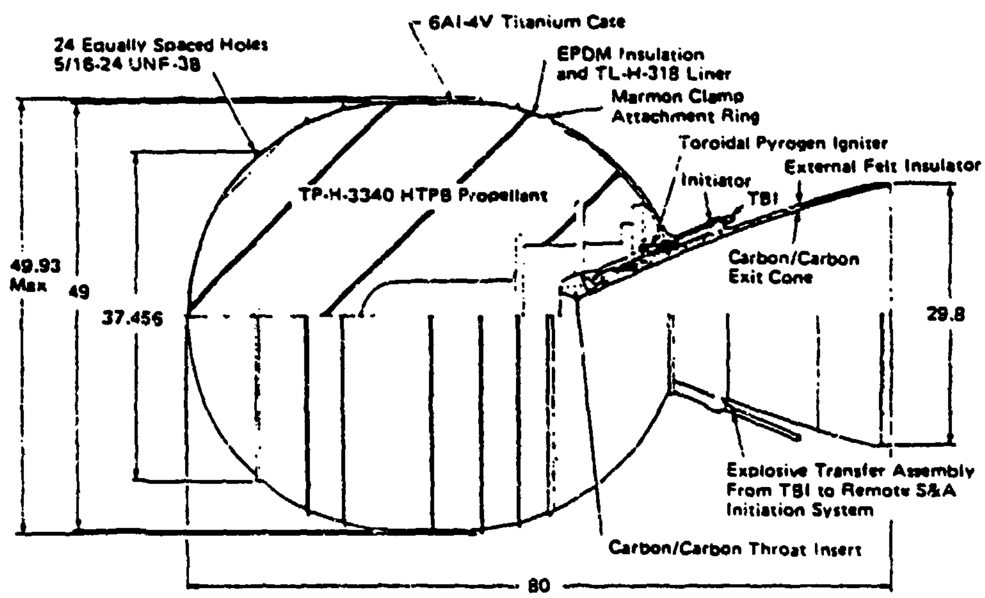

Figure C13-6 PAM Rocket Motor

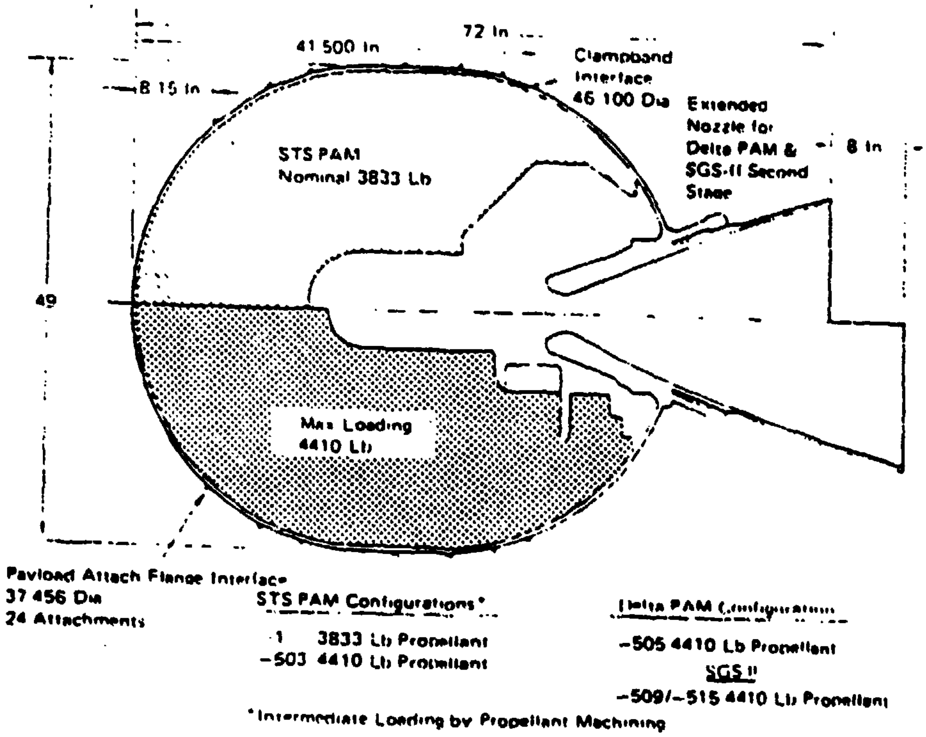

Figure C13-7 Comparison of STAR-48 PAM Motor Configuration

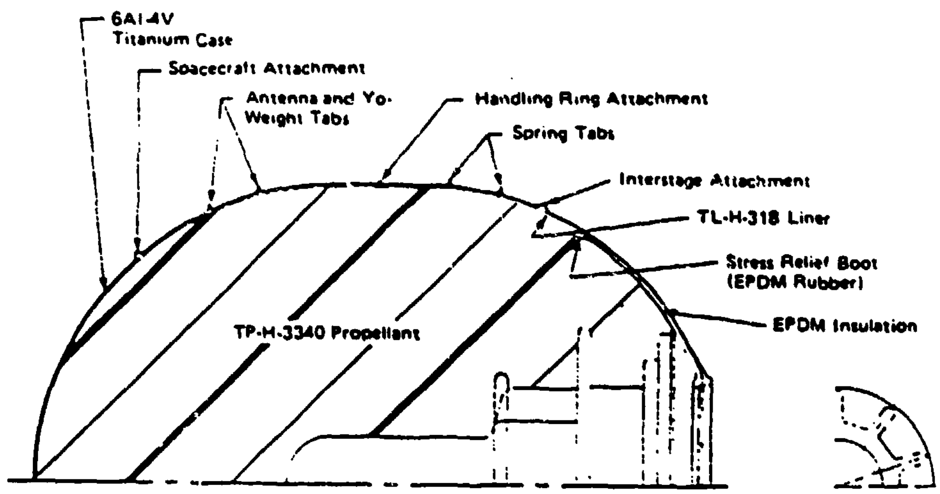

Figure C13-8 STAR-48 Loaded Case Assembly

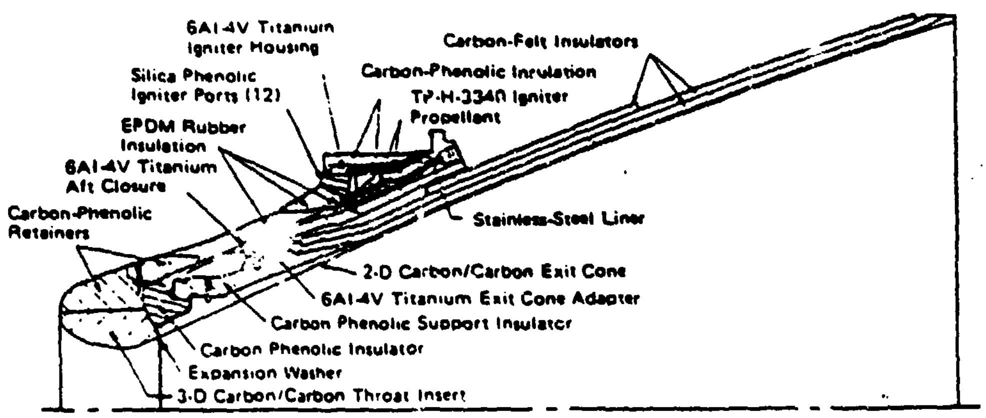

Figure C13- STAR-48 Nozzle Assembly

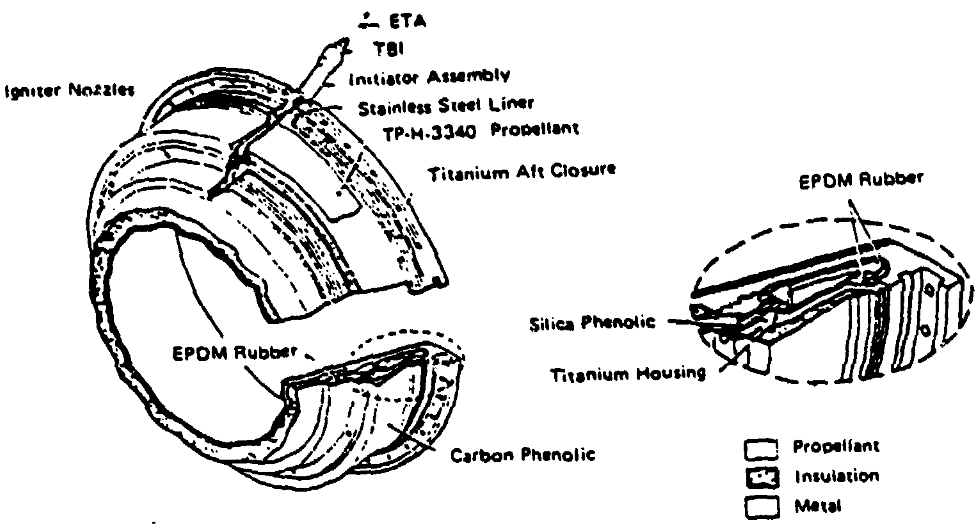

Figure C13-10 Toroidal Igniter

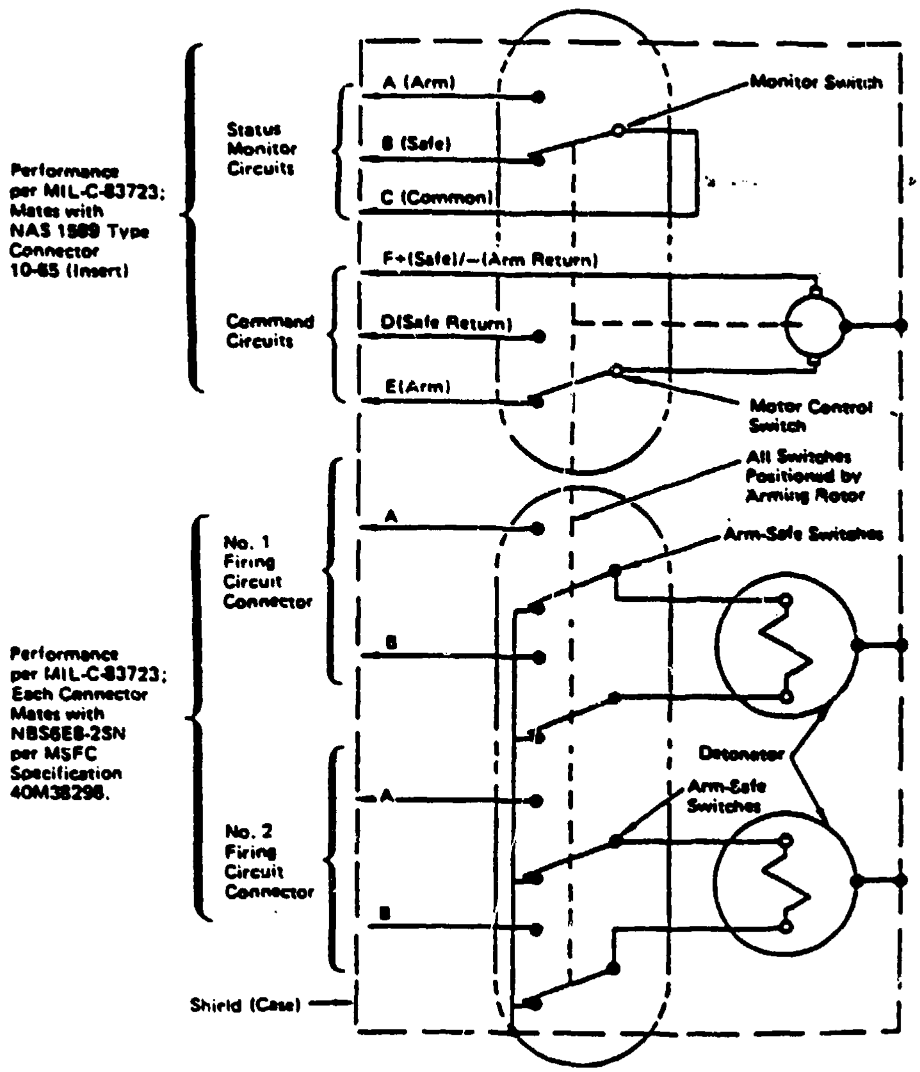

Figure C13-11 Electrical Schematic, S & A Device

Appendix C14
Transtage

APPENDIX C14
TITAN 34D TRANSTAGE

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APPENDIX C14
TITAN 34D TRANSTAGE

C14.0 INTRODUCTION

The T34D/TS system is a combination of the Titan IIIC Transtage and the Titan 34D/IUS programs. The T34D/TS launch vehicle uses the T34D core, previously safety assessed, and the flight qualified Titan IIIC third stage or "transtage."

C14.1 GENERAL DESCRIPTION

The T34D Common Core (T34K Standard Airborne Vehicle, CI No. T10D34K) is designed for maximum commonality between Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base (VAFB). The Common Core is configured with provisions for installing the unique airborne equipment of CCAFS (CI T01D34D and CI T05D34D) or VAFB (CI T04D34D) and two solid rocket motors. The VAFB configuration is known as T34D/Radio Guidance System (RGS). Two configurations exist for CCAFS, the T34D/IUS and the T34D/Transtage.

In referencing various parts of the vehicle two reference systems are used: the vehicle compartment designators and the vehicle's three-axis reference system. Figure C14-1 shows the alphanumeric compartment designators. The compartment numbers indicate the associated stage, and the letters are assigned from the top of the stage down.

The transtage family for Titan, as well as STS, is depicted in Figure C14-2. STS transtage receipt-to-launch flow is depicted in Figure C14-3.

C14.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C14.2.1 Titan Stage III - Transtage

The Transtage rocket engine is a multiple-restart, pressure-fed liquid propellant engine using the storable propellants Aerozine-50 and nitrogen tetroxide to develop 8000-lb thrust in vacuum. Three versions of the Transtage engine are available. They are designated AJ10-138 standard, AJ10-138A, and AJ10-138 modified. The T34D uses the AJ10-138A engines, called Improved Transtage Injector Pattern (ITIP). The purpose of the new pattern is to improve engine performance. Two of these engines comprise the propulsion system for Stage III (Fig. C14-4); each engine is a separate entity being unified only through common propellant feed and electrical supply systems and a common mounting frame supplied as a part of the vehicle. The engine located toward the target side of the vehicle is Subassembly 4; the engine on the flight-up side of the vehicle is Subassembly 5. Two spheres contain helium at an initial pressure of 3280 psig to maintain tank pressure at approximately 160 psia to force propellants from the tanks into the engine. The engines consists of a thrust chamber assembly, propellant lines, control harness, and instrumentation.

The Transtage propulsion system may be used during the boost phase of vehicle flight to reach higher orbits, modify an established orbit, and to rendezvous with other space vehicles. The engine is designed for a minimum of three restarts. Before starting under zero-gravity conditions, the attitude control engines must be fired to provide forward thrust to settle the liquid propellant in the main engine tanks.

The engine start sequence is initiated by applying a 28-Vdc signal to the pilot valve solenoid. With the pilot valve energized, fuel is ported through the pilot valve to a cavity behind the bipropellant valve power piston. The force resulting from the fuel pressure on the power piston is sufficient to overcome the spring and friction forces holding the valve in the closed position. This force moves the common fuel poppet and oxidizer piston stem assembly to the open position, causing propellants to flow through the bipropellant valve to the injector. Opening time is controlled by an orifice located between the pilot valve and bipropellant valve. Propellants from the bipropellant valve flow into the fuel and oxidizer manifolds of the injector and are injected into the thrust chamber where hypergolic ignition occurs. The valve design provides an oxidizer lead. Propellant flow rates to maintain the design mixture ratio for the engine system are controlled by balance orifices located at the bipropellant valve inlets. Ninety percent of rated thrust is achieved within 0.4 second after receipt of the electrical signal at the pilot valve. The pilot valve solenoid draws approximately 1.6 amps to sustain engine operation.

Termination of the 28-Vdc signal to the pilot valve allows the bipropellant valve power cavity to be vented through the pilot valve overboard vent tube. The bipropellant valve spring then forces the fuel poppet and oxidizer stem assembly to the closed position, terminating fuel and oxidizer flow to the thrust chamber. Bipropellant valve closing time is controlled by an orifice in the pilot valve overboard vent tube.

A summary of AJ10-138A Rocket Engine configuration is provided in Figure C14-5.

C14.2.2 Propellant Tanks

The Transtage propellant tanks are constructed of titanium. The forward domes are elliptical and the aft domes conical. Incorporated into each of the tank barrels are two longerons to facilitate support of the tanks on the truss (Fig. C14-6).

Transtage pressurization and vent system is shown in Figure C14-7. The pressurization figures are shown in Table C14-1. The transtage propellant feed system is illustrated in Figure C14-8, and the figures for the propellant feed system are shown in Table C14-2.

The oxidizer tank is 140.7 inches long with a diameter of 63.1 inches. Although the basic thickness of the tank is 0.057 inches, there is one area, just below the intersection of the barrel and the core (in the radius area), where the thickness is only 0.029 inches.

Table C14-1 T34D/Transtage Pressurization and Vent System

CODE	QTY	DRAWING NO.	NOMENCLATURE	MATERIAL	MAXIMUM FLIGHT PRESSURE	MAXIMUM GROUND (LOCK-UP)	PROOF PRESSURE (PSIG)	BURST PRESSURE (PSIG)	SAFETY FACTOR FLIGHT PROOF	SAFETY FACTOR FLIGHT BURST	SAFETY FACTOR GROUND LOCK-UP PROOF	SAFETY FACTOR GROUND LOCK-UP BURST
A	2	PD47S0154-049	Check Valve		207.0 psig 221.7 psia	87.3 psig 102.0 psia	306	518	1.48	2.46	3.51	5.84
B	2	PD47S0154-040	Check Valve		207.0 psig 221.7 psia	87.3 psig 102.0 psia	306	510	1.48	2.46	3.51	5.84
C	2	PD48S0114-029	Filter		3280.0 psig 3294.7 psia	3280.0 psig 3294.7 psia	5550	9250	1.69	2.82	1.69	2.82
D	1	PD48S0113-419	Coupling (O)	6061-T6-AL	207.0 psig 221.7 psia	87.3 psig 102.0 psia	600	900	2.90	4.35	6.0	9.0
E	4	PD47S0153-069	Solenoid Valve		3280.0 psig 3294.7 psia	3280.0 psig 3294.7 psia	5550	9250	1.69	2.82	1.69	2.82
F	1	PD71S0070-019	Ground Check		207.0 psig 221.7 psia	87.3 psig 102.0 psia	321	535	1.55	2.59	3.68	6.13
G	1	PD48S0113-099	Cap (O)	6061-T6-AL	0 psig 14.7 psia	0 psig 14.7 psia	600	900	-	-	-	-
H	1	PD71S0070-119	Sol. Valve Pressure Switch		207.0 psig 221.7 psia	87.3 psig 102.0 psia	321	535	1.55	2.59	3.68	6.13
J	1	PD47S0163-029	Valve, Hand Shutoff		3280.0 psia 3294.7 psia	3280.0 psig 3294.7 psia	5500	9250	1.69	2.82	1.69	2.82
K	1	PD48S0112-001	Seal	Teflon	0 psig 14.7 psia	3280.0 psig 3294.7 psia	7200	14000	-	-	-	-
L	1	PD48S0112-069	Coupling Half (Ha)	1704 PH CRES	0 psig 14.7 psia	3280.0 psig 3294.7 psia	7200	14000	-	-	-	-
M	1	PD48S0112-079	Cap (Ha)	1704 PH CRES	0 psig 14.7 psia	0 psig 14.7 psia	6000	12000	-	-	-	-
N	2	PD48S0115-001 PD48S0115-009	Storage Sphere (Ha)		3280.0 psig 3294.7 psia	3280.0 psig 3294.7 psia	4936	6600	1.50	2.01	1.50	2.01
P	2	PD48S0113-001	Gasket	KYNAR	0 psig 14.7 psia	0 psig 14.7 psia	800	900	-	-	-	-
R	1	PD48S0113-089	Cap (F)	17-4 PH CRES	0 psig 14.7 psia	0 psig 14.7 psia	600	900	-	-	-	-

(F) - FUEL (O) - OXIDIZER (Ha) - HELIUM

Table C14-2 T34D/Transtage Propellant Feed System

CODE	QTY	DRAWING NO.	NOMENCLATURE	MATERIAL	MAXIMUM FLIGHT PRESSURE	MAXIMUM GROUND (LOCK-UP)	PROOF PRESSURE (PSIG)	BURST PRESSURE (PSIG)	SAFETY FACTOR FLIGHT PROOF	SAFETY FACTOR FLIGHT BURST	SAFETY FACTOR GROUND LOCK-UP PROOF	SAFETY FACTOR GROUND LOCK-UP BURST
A	1	PD41S0087-012	Tube Assembly Fuel		192.0 psig 206.7 psia	92.3 psig 107.0 psia	359	449	1.87	2.34	3.89	4.86
B	1	PD41S0088-012	Tube Assembly Oxidizer		214.0 psig 228.7 psia	94.3 psig 109.0 psia	3.41	427	1.59	2.00	3.62	4.53
C	1	PD48S0113-89	Pressure Cap Fuel	6061-T6-AL	0 psig 14.7 psia	0 psig 14.7 psia	600	900	----	---	----	----
D	1	PD48S0113-099	Pressure Cap Oxidizer	6061-T6-AL	0 psig 14.7 psig	0 psig 14.7 psig	600	900	----	----	-----	----
E	2	PD83S0115-519	Sensing Element (F) (O		214.0 psig 228.7 psia	94.3 psig 109.0 psia	375	470	1.75	2.20	3.98	4.98
F	1	PD48S0113-001	Seal	Teflon	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
G	2	PD47S0155-039 80801831070-019	Valve, Check		3 PSID	0 psig 14.7 psia	15 PSID	25 PSID	5.0	8.3	----	----
H	1	PD48S0113-379	Coupling Half (F)	17-4 PH CRES	192.0 psig 206.7 psia	92.3 psig 107.0 psia	600	900	3.13	4.69	6.50	9.75
J	1	PD48S0113-409	Coupling Half (O)	17-4 PH CRES	214.0 psig 228.7 psia	94.3 psig 109.0 psia	680	960	2.83	4.21	6.16	9.54
K	2	PD47S0115-029 80801831070-009	Valve, Check		3 PSID	0 psig 14.7 psia	15 PSID	25 PSID	5.0	8.3	----	----

Table C14-2 (continued)

CODE	QTY	DRAWING NO.	NOMENCLATURE	MATERIAL	MAXIMUM FLIGHT PRESSURE	MAXIMUM GROUND (LOCK-UP)	PROOF PRESSURE (PSIG)	BURST PRESSURE (PSIG)	SAFETY FACTOR FLIGHT PROOF	SAFETY FACTOR FLIGHT BURST	SAFETY FACTOR GROUND LOCK-UP PROOF	SAFETY FACTOR GROUND LOCK-UP BURST
1	1	80801B31023-049	Tee (F)	321 CRES SH	192.0 psig 206.7 psig	92.3 psig 107.0 psig	377 +0/-18		1.96		4.09	
2	1	80801B31023-039	Tee (O)	321 CRES SH	214.0 psig 228.7 psia	92.3 psig 107.0 psia	377 +0/-18		1.76		4.09	
*3	2	80801B31071-008	Blanked off Spool Pieces	321 CRES SH	N/A	N/A	336 +0/-16		N/A		N/A	
4	1	80801B31024-049	Deflector (F)		N/A	N/A	N/A		N/A		N/A	
5	2	80801B31025-009	Baffle		N/A	N/A	N/A		N/A		N/A	
6	1	80801B31024-039	Deflector (O)		N/A	N/A	N/A		N/A		N/A	
7	2	80801005400-001	Restricter		180 psig 194.7 psia	129 psig 143.7 psia	359 +5/-5		2.0		2.8	4.0
8		80801B35022-009	Screen (F)		N/A	N/A	N/A		N/A		N/A	
9		80801B35023-009	Screen (O)		N/A	N/A	N/A		N/A		N/A	
10	1	80801A58022-370, 379, 380, 389, 390, 419, 450	Manhole Covers (O)	Aluminum	207.0 psig 221.7 psia	87.3 psig 102.0 psia	230.2		1.11		2.64	
11	1	80801A57022-370	Manhole Covers	Aluminum	187.0 psig 201.7 psia	87.3 psig 102.0 psia	206.5		1.10		2.37	
12	1	80801A58000	Tank (O)	Titanium	Fwd 207.0 psig 221.7 psia	87.3 psig 102.0 psia	230.2		1.11		2.64	
					Aft 214.0 psig 228.7 psia	94.3 psig 109.0 psia	235.5		1.10		3.50	
13	1	80801A57000	Tank (F)	Titanium	Fwd 187.0 psig 201.7 psia	87.3 psig 102.0 psia	206.5		1.10		2.37	
					Aft 192.0 psig 206.7 psia	92.3 psig 107.0 psia	212.6		1.11		2.30	

The fuel tank is 165.9 inches long, with a diameter of 46.5 inches. Its average thickness is 0.046 inches, the thinnest section being 0.021 inches. It is located in the same area as the thinnest section of the oxidizer tank.

The oxidizer tank has a volume of 176 cubic feet. It is pressurized to a nominal pressure of 92 psia on the ground. In flight, the operating pressure of the tank is increased to 161 psia. The fuel tank operates at the same pressures, but has a volume of about 141 cubic feet.

C14.2.3 Attitude Control Systems (ACS)

The Transtage multipurpose ACS (Fig. C14-9) contained in the control module is a blowdown, storable, liquid-monopropellant variable-pulsewidth rocket engine control system. The propellant is anhydrous hydrazine (N_2H_4), and the pressurant is gaseous nitrogen, which are both stored in a single tank separated by a rubber diaphragm. The system has no single nonstructural mode of failure and performs a wide range of propulsion functions, including vehicle pitch, yaw, and roll control; main tank propellant settling; orbit adjust, maneuvering, and vernier control; and multipayload deployment and controlled dispersion. Total impulse of 50,000 lb-sec is provided. There are six engine modules, each having two thruster assemblies. Each thruster assembly contains series-redundant propellant valves. A Shell 405 catalyst bed is provided in each thruster. The two thruster assemblies provide redundancy; two thruster malfunctions are required before engine module failure occurs. Thrust chamber firing is initiated by a command signal from the guidance system through the digital flight control system.

Components for the ACS system are shown in Figure C14-10 and Table C14-3.

Attitude Control Pressurization System - Gaseous nitrogen (GN_2) is stored at a pressure of 355 $\pm$ 2 psig in a spherical pressurant tank. The tank is made of titanium and has a nominal capacity of 3.55 lb of GN_2. Included in the pneumatic module is a manual fill valve.

C14.2.4 Inertial Guidance System

The Inertial Guidance System (IGS) guides the various stages of the Titan vehicle on a trajectory and controls Stage III velocity so that the payload arrives at the desired location in space traveling at the desired velocity.

To guide the vehicle on this trajectory, the IGS computer sends analog DC steering signals to the digital flight control system (DFCS) actuators and the SRM thrust vector control system. During flight the computer also sends discrete commands to the airborne electrical system to control such things as engine cutoff and engine start signals.

The guidance operation (Fig. C14-11) is accomplished only during powered flight and assumes that the payload will follow a predictable path upon separation from the airframe. In brief, the guidance equipment measures vehicle acceleration, from which it computes present velocity and position. Measurements are made from an inertial reference, making unnecessary a ground link for guidance. The computer determines steering signals necessary to attain the proper vehicle attitude and delivers engine cutoff signals when the proper velocity has been reached. Because the orbital portion of flight is predictable, the guidance system need only bring the payload to a point in space which, at the proper attitude and velocity of the vehicle, will result in insertion into the correct orbit.

C14.2.5 Airborne Hydraulic Systems

The Stage I, II, and III hydraulic systems provide the engine actuators with hydraulic fluid under pressure in flight and during PACE ground checkout of the flight control system. Electrical signals from the DFCC-VDAs on the flight control computer are sent to the servo valves in the actuators. These electrical inputs control the flow of hydraulic fluid to the actuator pistons. The pistons are connected to the engine thrust chambers and gimbal the thrust chambers to change the attitude of the vehicle.

C14.2.6 Transtage Hydraulic System

The major component of the Transtage hydraulic system is an integrated motor pump/reservoir unit (Fig. C14-12). The other components are the hydraulic disconnect and four linear actuators.

The integral motor pump/reservoir is a variable delivery pressure compensated hydraulic pump enclosed by a reservoir and immersed in fluid. The pump is driven by a 28 volt compound wound DC motor with an integral RF filter. Operating pressure is 2250 to 2400 psi with a 2.0 gpm delivery capability. The motor pump/reservoir assembly also has an integral check valve, filter, pressure transducer and fill and bleed disconnects. The entire unit has a 24 in^3 oil capacity, weights 19.0 lbs.

C14.2.7 Flight Termination System

The Flight Termination System (FTS) provides the Range Safety Officer (RSO) with the capability to shutdown the core engines, or shutdown the core engines and destroy the Titan vehicle should it become necessary. The FTS will also automatically destroy stages of the vehicle if they should inadvertently separate.

Table C14-3 T34D/Transtage Attitude Control System

CODE	QTY	DRAWING NO.	NOMENCLATURE	MATERIAL	MAXIMUM FLIGHT PRESSURE	MAXIMUM GROUND (LOCK-UP)	PROOF PRESSURE (PSIG)	BURST PRESSURE (PSIG)	SAFETY FACTOR FLIGHT PROOF	FLIGHT BURST	GROUND PROOF	LOCK-UP BURST
A	1	PD475S0163-029	Hand Valve (N2)	17-4 PH CRES	355.0 psig 369.7 psia	355.0 psig 369.7 psia	5500	9250	15.49	26.06	15.49	26.06
B	1	PD485O112-089	Coupling Half (N2)	17-4 PH CRES	355.0 psig 369.7 psia	355.0 psig 369.7 psia	6000	12000	16.90	33.80	16.90	33.80
C	1	PD485O112-079	Pressure Cap	17-4 PH CRES	0 psig 14.7 psia	0 psig 14.7 psia	6000	12000	----	----	----	----
D	1	PD485O113-469	Coupling Half (F)	17-4 PH CRES	355.0 psig 369.7 psia	355.0 psig 369.7 psia	790	885	2.23	2.49	2.23	2.49
E	1	PD485O113-479	Pressure Cap	17-4 PH CRES	0 psig 14.7 psia	0 psig 14.7 psia	600	885	----	----	----	----
F	1	PD4700174-001 PF4700174-002	Start Valve (F)	17-4 PH CRES	355.0 psig 369.7 psia	355.0 psig 369.7 psia	600	1000	1.6	2.82	1.69	2.82
G	4	PD6000179-010 PD6000179-020	Rocket Engine Module		N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
H	2	PD6000179-009 PD6000179-019	Rocket Engine Module		N/A	N/A	N/A	N/A	N/A		N/A	N/A
J	6	PD4800143-001	Filter Element	17-4 PH CRES	355.0 psig 369.7 psia	355.0 psig 369.7 psia	666	877	1.88	2.47	1.88	2.47
K	1	PD485O113-001	Seal	KYNAR	0 psig 14.7 psia	0 psig 14.7 psia	N/A	N/A	N/A	N/A	N/A	N/A
L	1	PD485O112-001	Seal	AMS-3651B Teflon	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
1	1	80801836220-069	Tank	6AL-4V Titanium	355.0 psig 369.7 psia	355.0 psig 369.7 psia	600	800	1.69	2.25	1.69	2.25
2	1	80801836237-089	Tube Assembly	21-6-9 CRES	355.0 psig 369.7 psia	355.0 psig 369.7 psia	2000 206.5	N/A	5.63	----	5.63	----

(F) - FUEL

C14.2.8 Inadvertent Separation Destruct System (ISDS)

An ISDS is incorporated in each SRM, in Stage I and in Stage II, to automatically destroy these stages should they inadvertently separate from the Space Launch Vehicle. No ISDS is required for the Transtage because the RSO can use the command transmitter to send the destruct signal to the command receivers.

C14.2.9 Command Shutdown and Destruct System

The CSDS receives and responds to ground originated radio commands for either engine shutdown only or for engine shutdown together with vehicle destruct. The command system consists of an omnidirectional antenna system, two command receivers, engine shutdown circuitry, destruct circuitry, and associated power systems. This equipment provides two types of signals. The first shuts down all core engines, and the second destroys the propellant tanks of all stages that are electrically mated to the Transtage.

Engine shutdown is in response to engine shutdown command from the RSO. Each command receiver issues a nominal 28-volt output signal that energizes the thrust chamber valve overrides in both Stages I and II and deenergizes the propellant solenoid valves in the Transtage.

C14.2.10 Hazardous Material Data

Stage I, II, and Transtage all use the same storable, hypergolic, liquid rocket propellants. The fuel is Aerozine-50, an approximately 50/50 mixture of hydrazine and unsymmetrical dimethylydrazine (UDMH). The oxidizer is nitrogen tetroxide. The use of propellants storable at ambient temperature and pressure eliminates holds and delays. The hypergolic property of spontaneous ignition upon contact with one another eliminates the need for an ignition system and related checkout and support equipment.

Transtage propellants should be considered as fire, explosion, toxic, and corrosion hazard producers, and it is extremely important that personnel involved in the handling of or working near the propellants have a sound knowledge of their characteristics. Correct handling and storage procedures must be used to minimize the hazards to personnel and equipment.

C14.2.11 LOCATION OF PYROTECHNICS

The Titan 34D/Transtage uses many types of ordnance items. Figure C14-13 indicates the approximate location of vehicle ordnance. Table C14-4 corresponds to the figure and identifies the type of Transtage EED.

Table 14-4
Transtage Ordnance Items

1. Transtage Destruct System - One destruct safe-arm initiator, PD64S0336-515. Three strands of primacord MMS-N170 Type I, Form B. Six boosters, 60#7-1 (one on each end of primacord strands). Two bidirectional destruct charges, PD60S0135-503.

2. Stage II to Transtage Attachment and Separation System. Sixteen separation nut pressure cartridges, PD60S0129-507.

3. Attitude Control System Valve Actuation - Two ACS Start Valve Cartridges, PD60S0129-507.

4. Transtage Pressure System Valve - Two crossover valve pressure cartridges, PD60S0129-507.

C14.2.12 Pressure Cartridge

(1) Description - The cartridge is an electrically initiated, hermetically sealed powerpack consisting of a dual bridgewire embedded in an explosive primer mix adjacent to a booster charge for ignition of a base propellant. It produces a hot gas under high pressure of short duration.

(2) Ordnance Item Number and Manufacturer PD60S0129-507 - McCormick-Selph Associates

(3) Quantity Per Flight - 64

(4) Purpose of the device - Actuation of the following: Forty-eight separation nuts, PD33S0007-005, and -003, for attachment and release of Stage I to Stage II, Stage II to Transtage, and SRM forward outrigger to core. Six P/F release cable cutters (two cartridges per cutter). One transtage pressure system crossover valve for venting helium system (two cartridges). One ACS valve (two cartridges).

(5) Type of Explosive and Weight

Primer	90-mg zirconium/ammonium perchlorate and borium chromate
Loose Booster	346-mg boron/potassium nitrate
Booster Pellet	84- to 89-mg boron/potassium nitrate
Grain Sustainer	186- to 200-mg ammonium nitrate composite

(6) Testing Specifications - Bridge circuits (pin numbers) - Pins A-D and B-C Circuit resistance - Each circuit 0.22 to 0.32 ohms; maximum test current - 0.02 amperes; maximum no-fire current - 1.0 amp for five minutes; minimum sure-fire current - 2.2 amps; firing voltage - 25.0 vdc minimum; minimum pin-to-case resistance - 2 megohms.

(7) Grounding Provisions - Modified connectors 81D28P8-4S will be used for shorting the pressure cartridge bridgewires.

(8) Storage Specification - Normal storage for Class C explosive.

C14.3 Mission Scenario

A typical mission(s) scenario is described in Figures C14 thru C17.

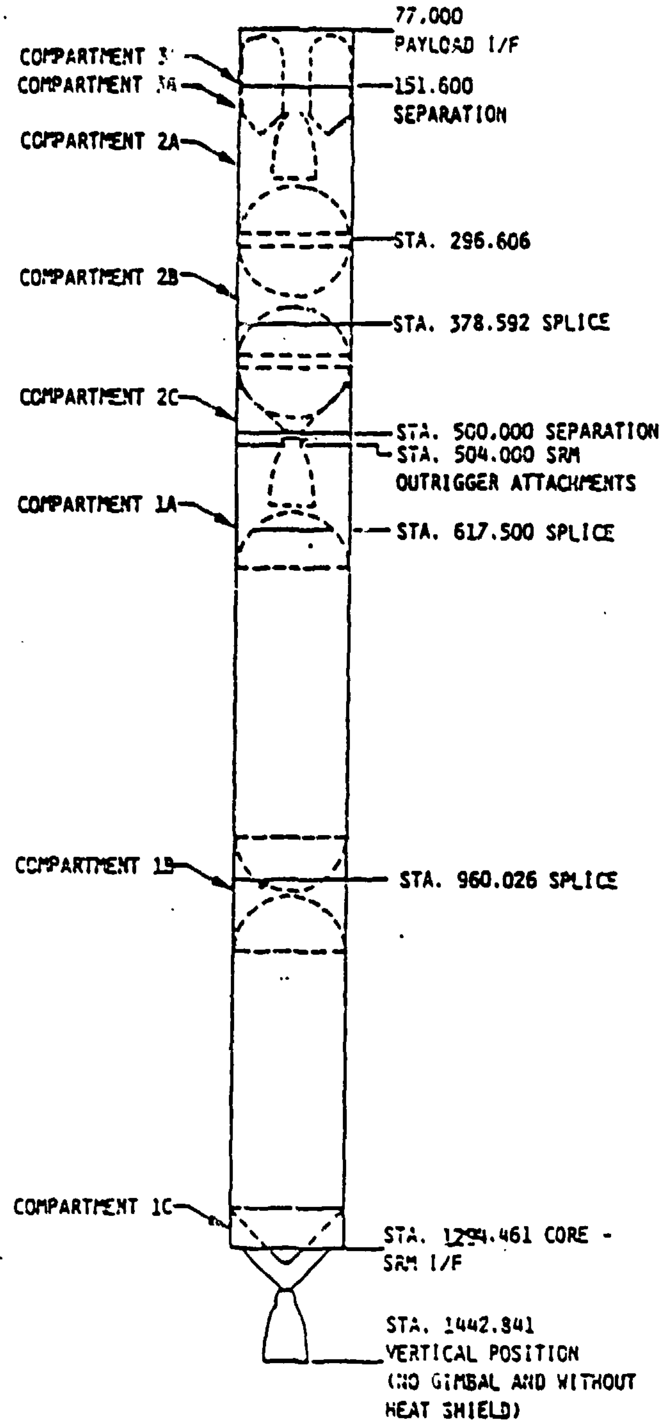


Figure C14-1 Compartment/Station Designation

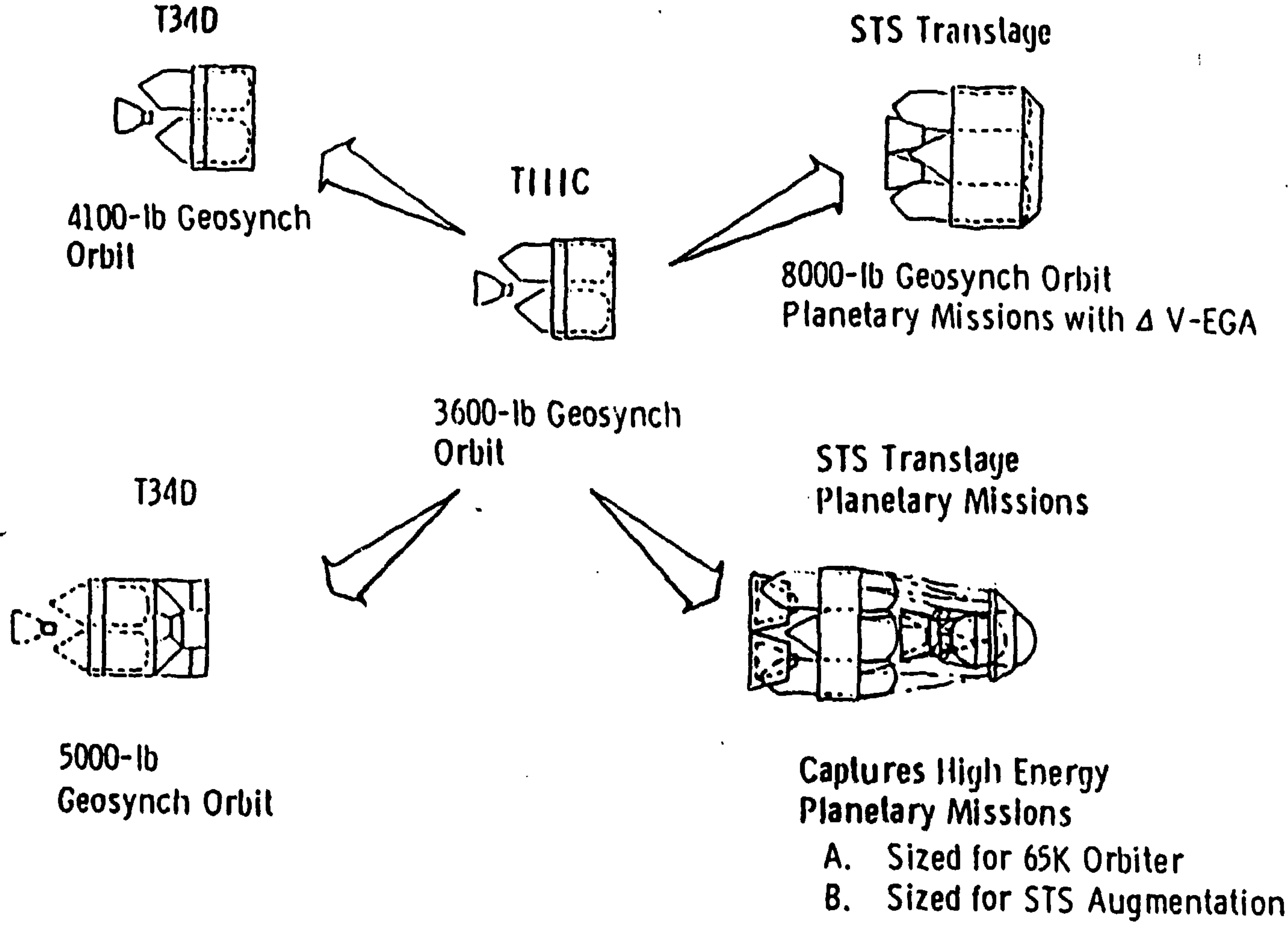


Figure C14-2 Transtage Family

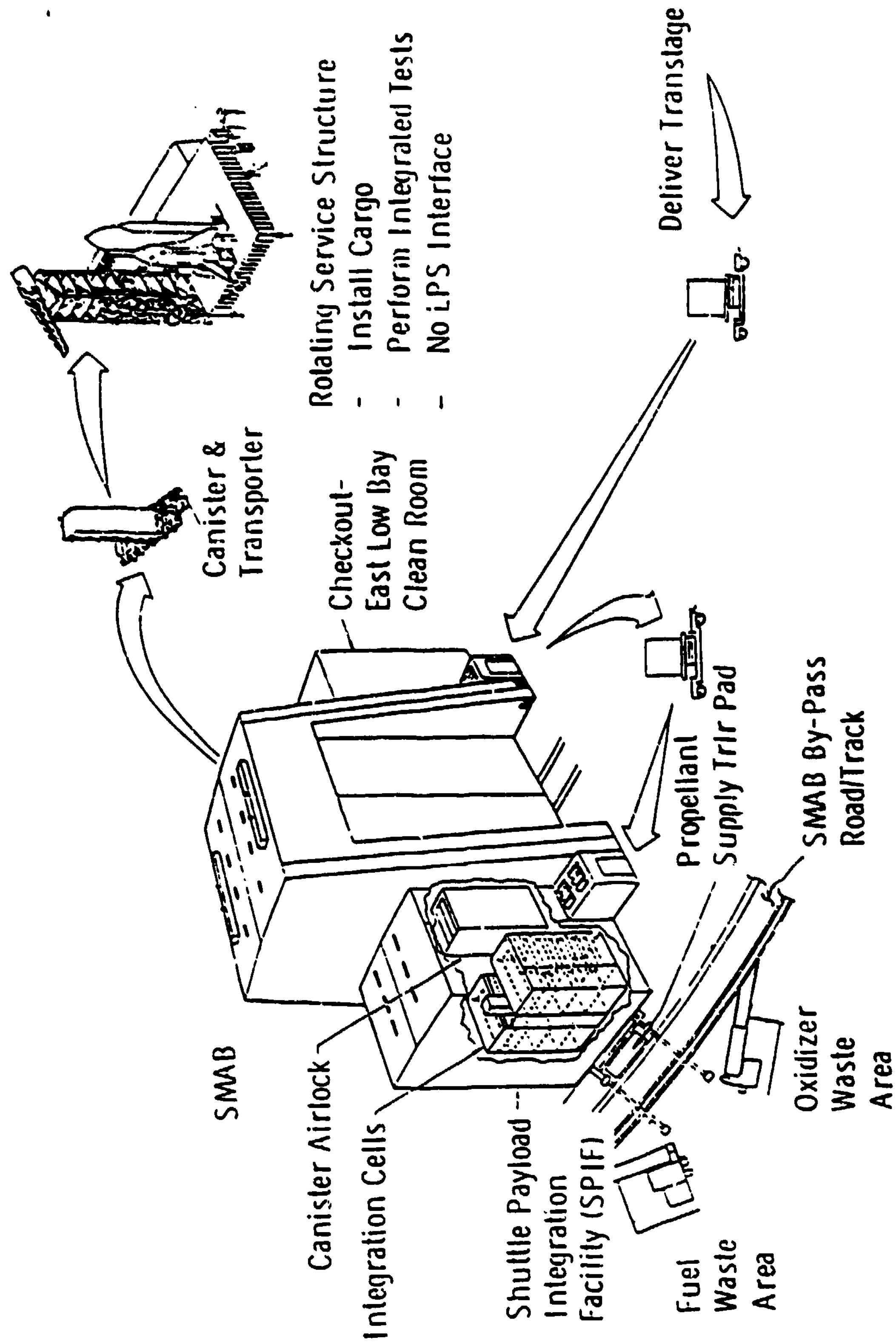


Figure C14-3 STS Transtage Receipt-to-Launch Flow

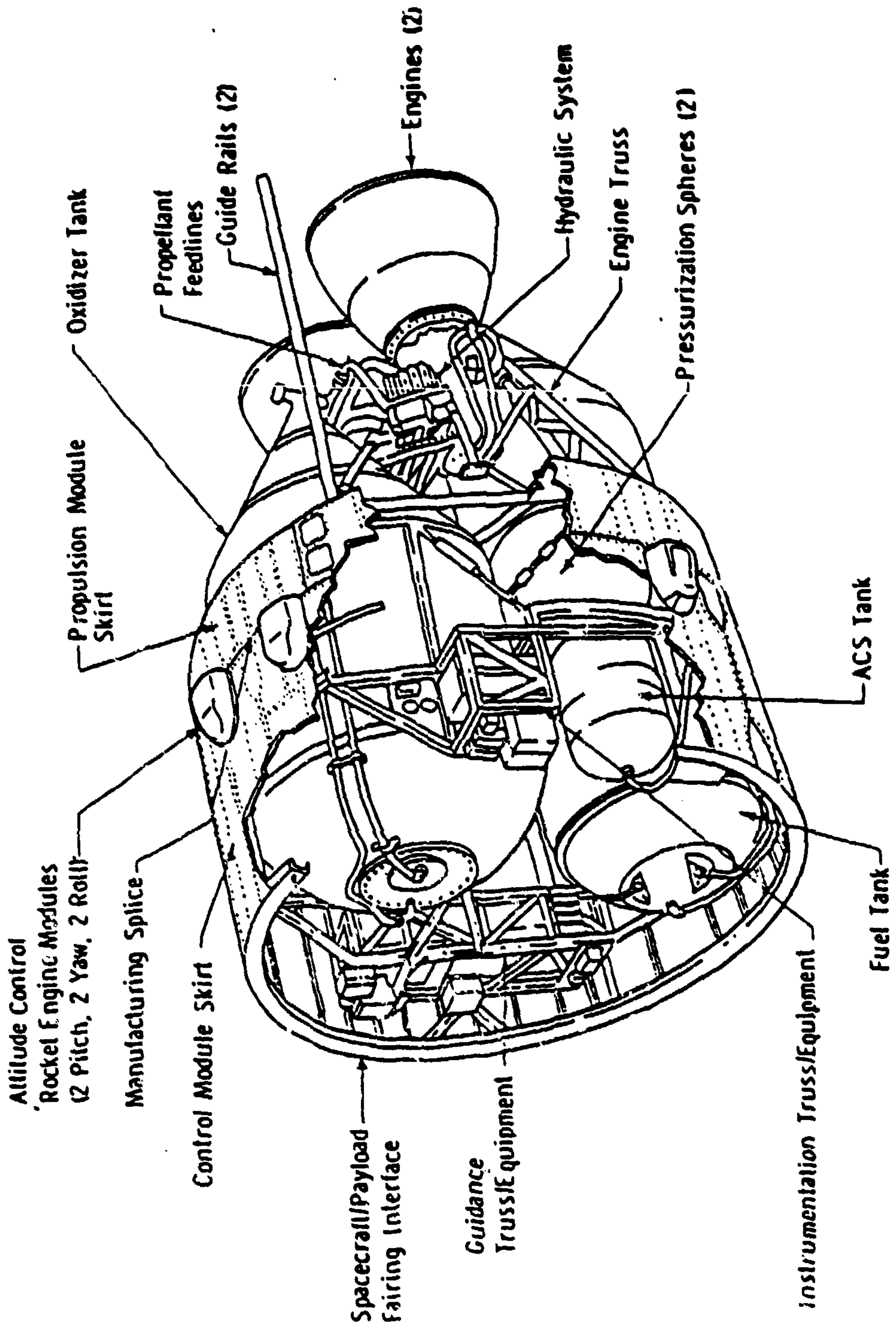
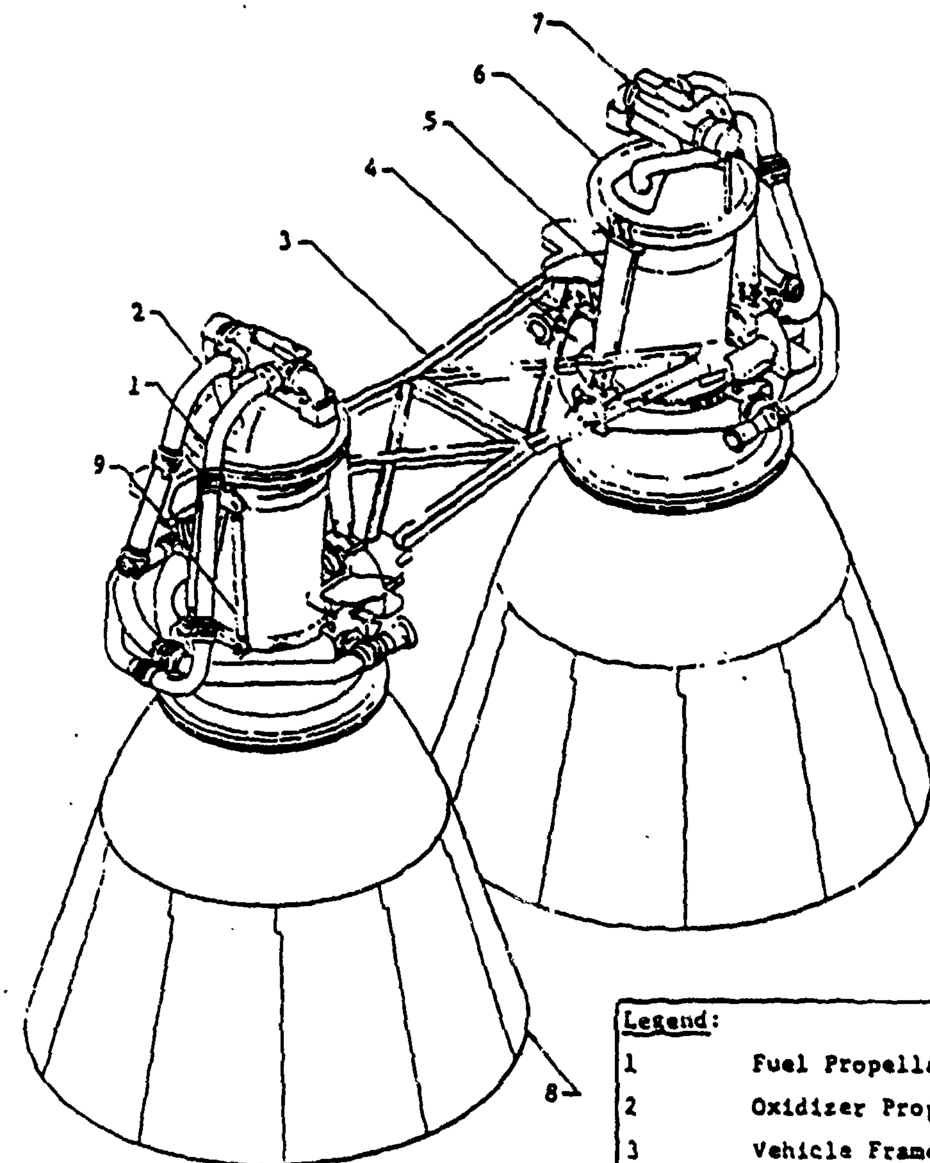


Figure C14-4 Stage III-Transtage



Legend:

1	Fuel Propellant Feed Line
2	Oxidizer Propellant Feed Line
3	Vehicle Frame (Reference)
4	Thrust-Mount Assembly
5	Thrust Chamber
6	Injector
7	Bipropellant Valve
8	Nozzle Extension
9	Shipping Arm

Figure C14-5 Engine Components

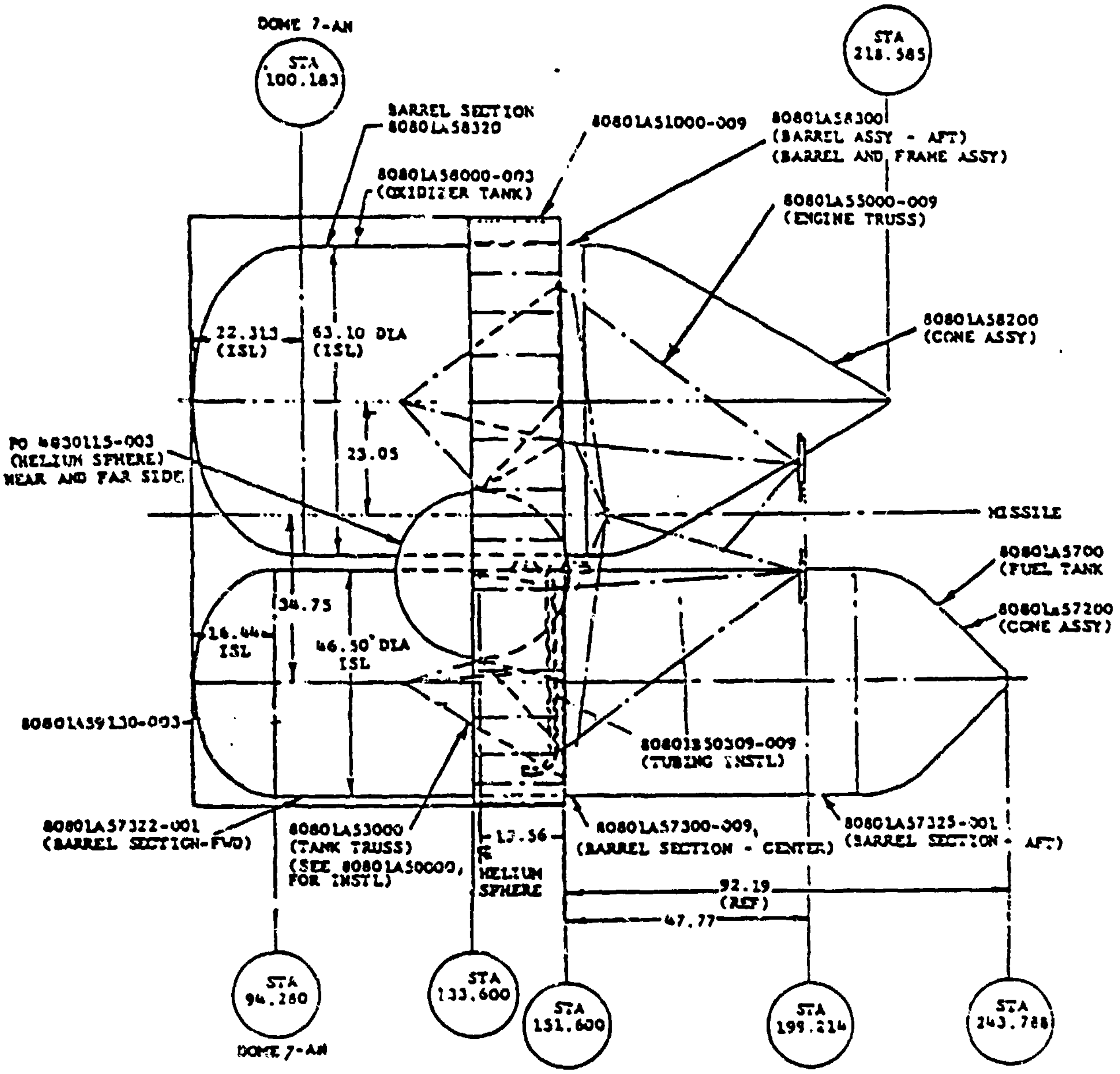


Figure C14-6 Transtage Propellant Tanks

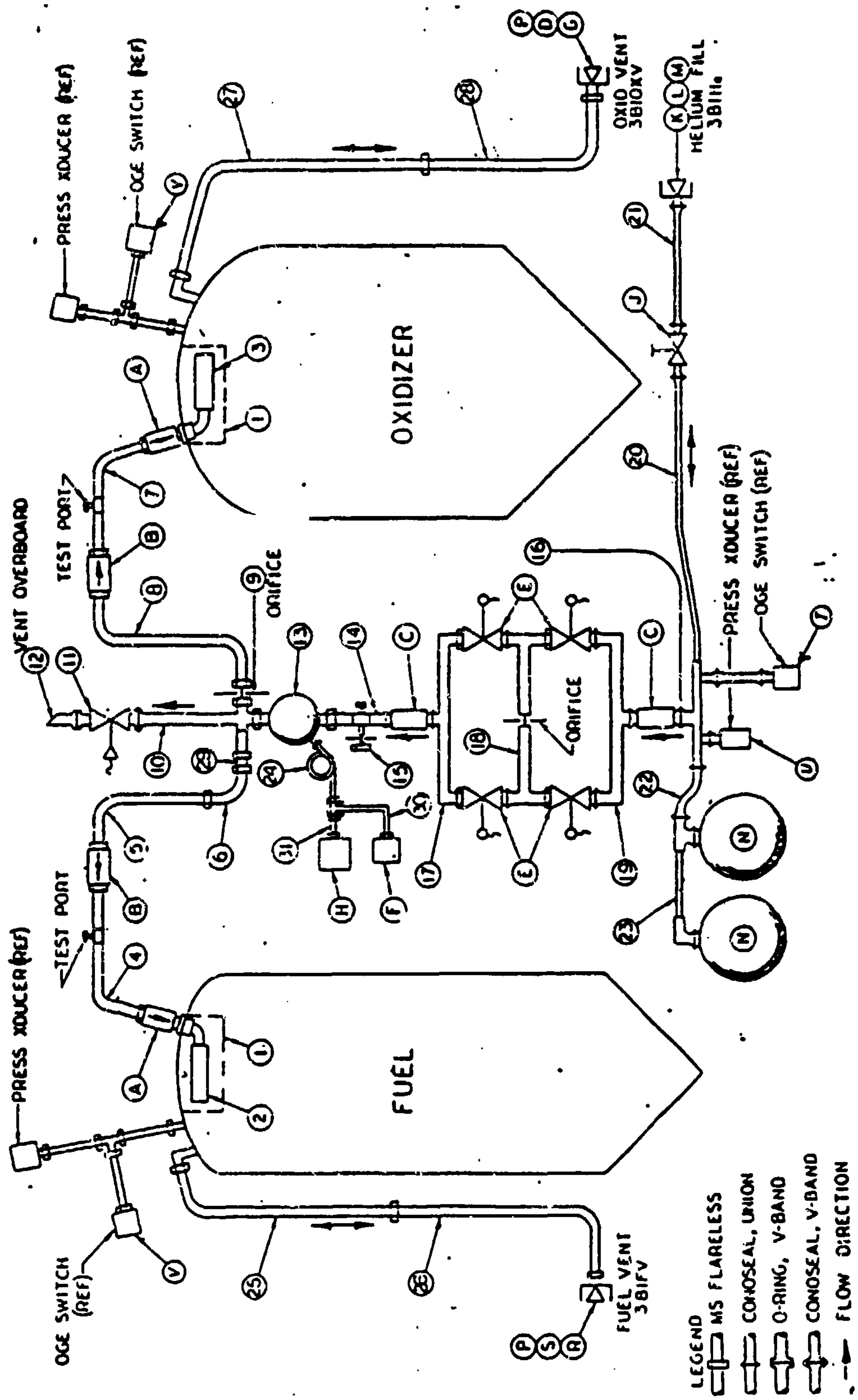


Figure C14-7 Transtage Pressurization and Vent System Components

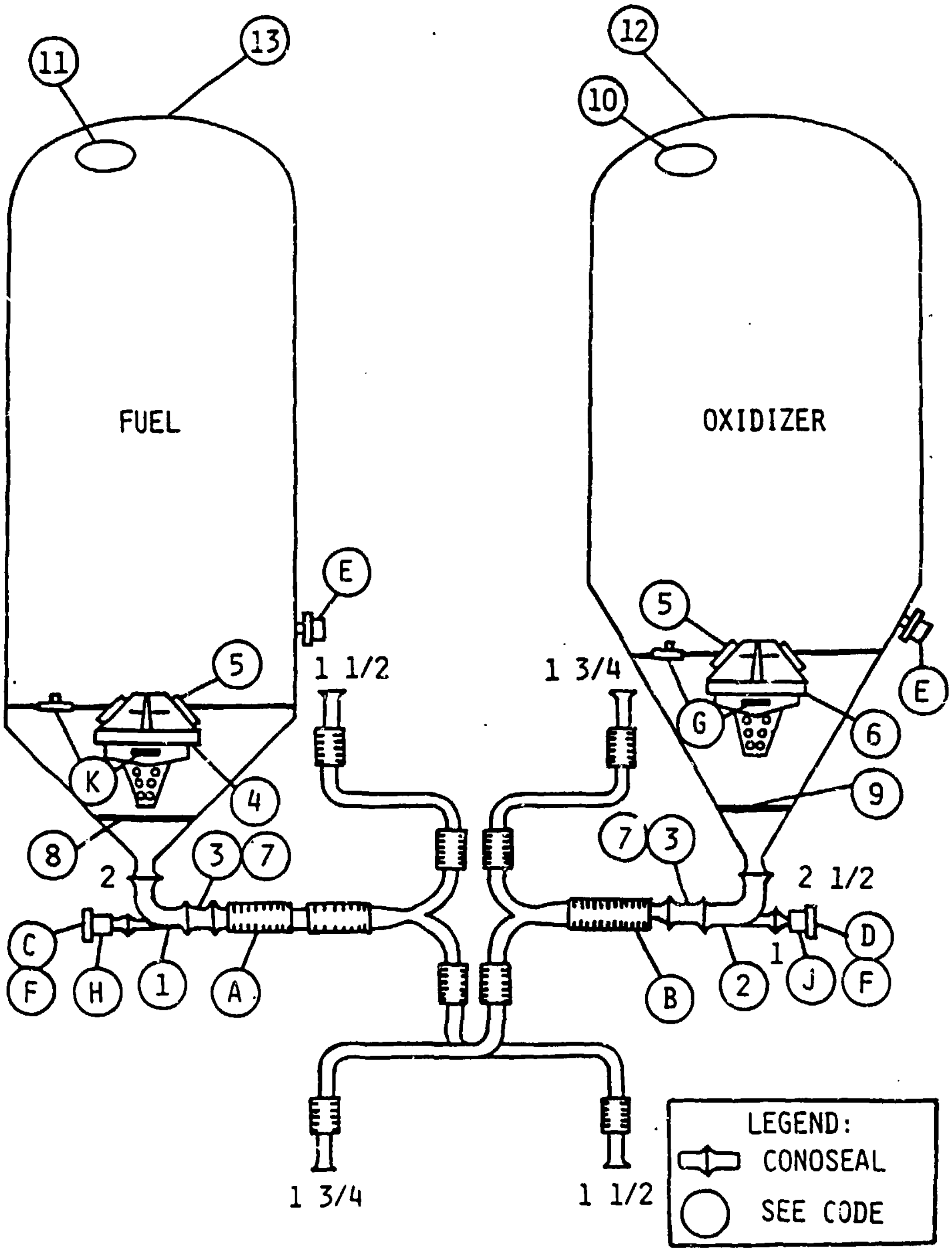


Figure C14-8 Transtage Propellant Feed System

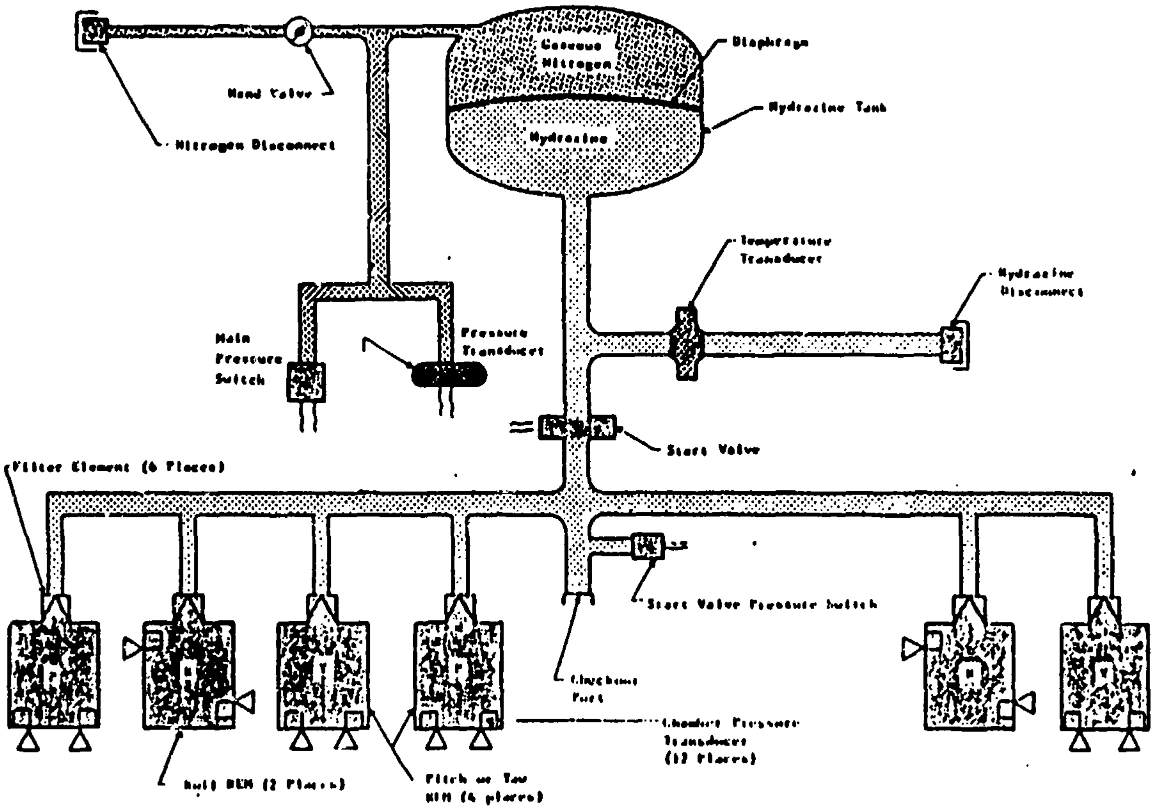


Figure C14-9 Attitude Control System Schematic

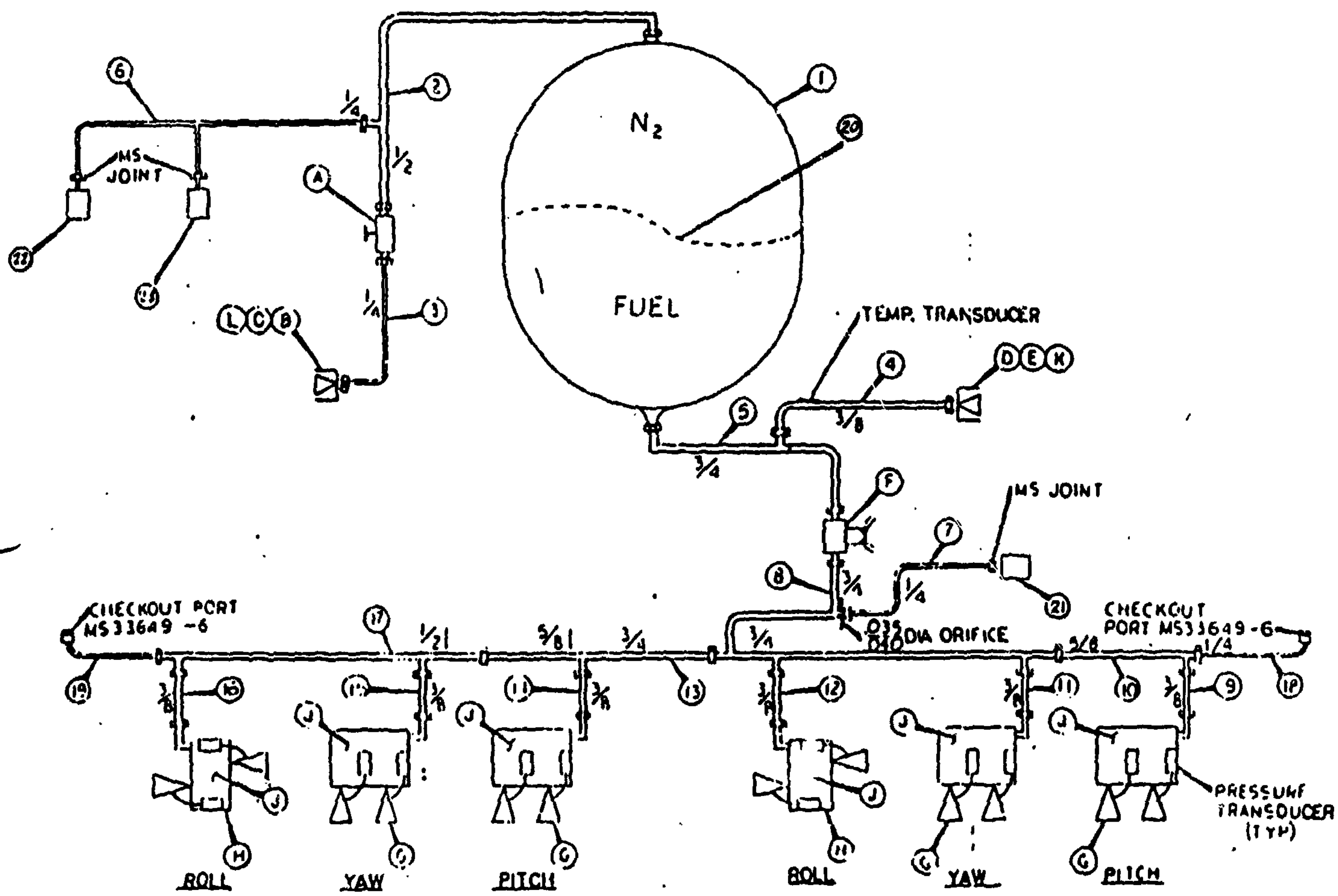


Figure C14-10 T34D/Transtage Attitude Control System Components

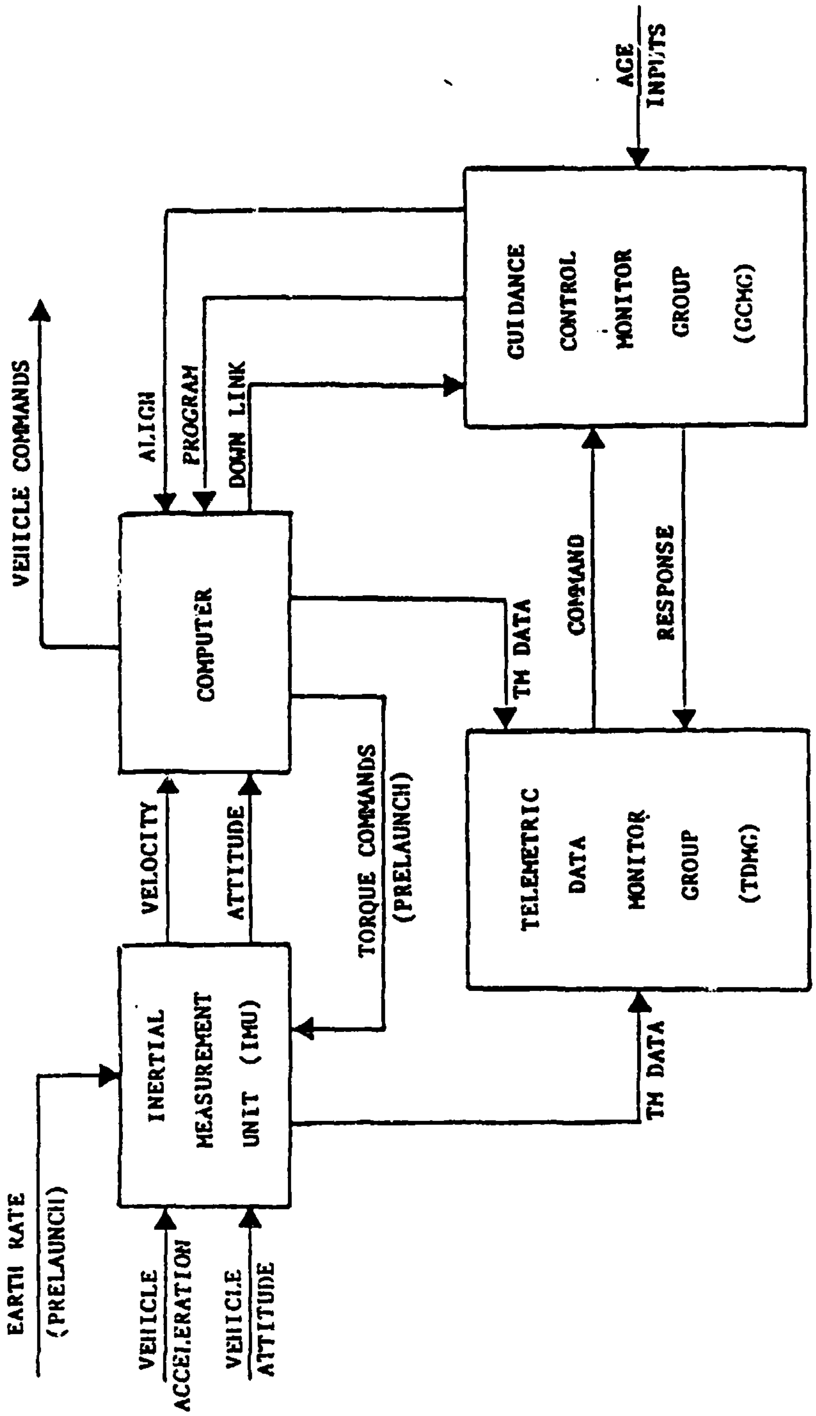


Figure C14-11 T34D/TS IGS System

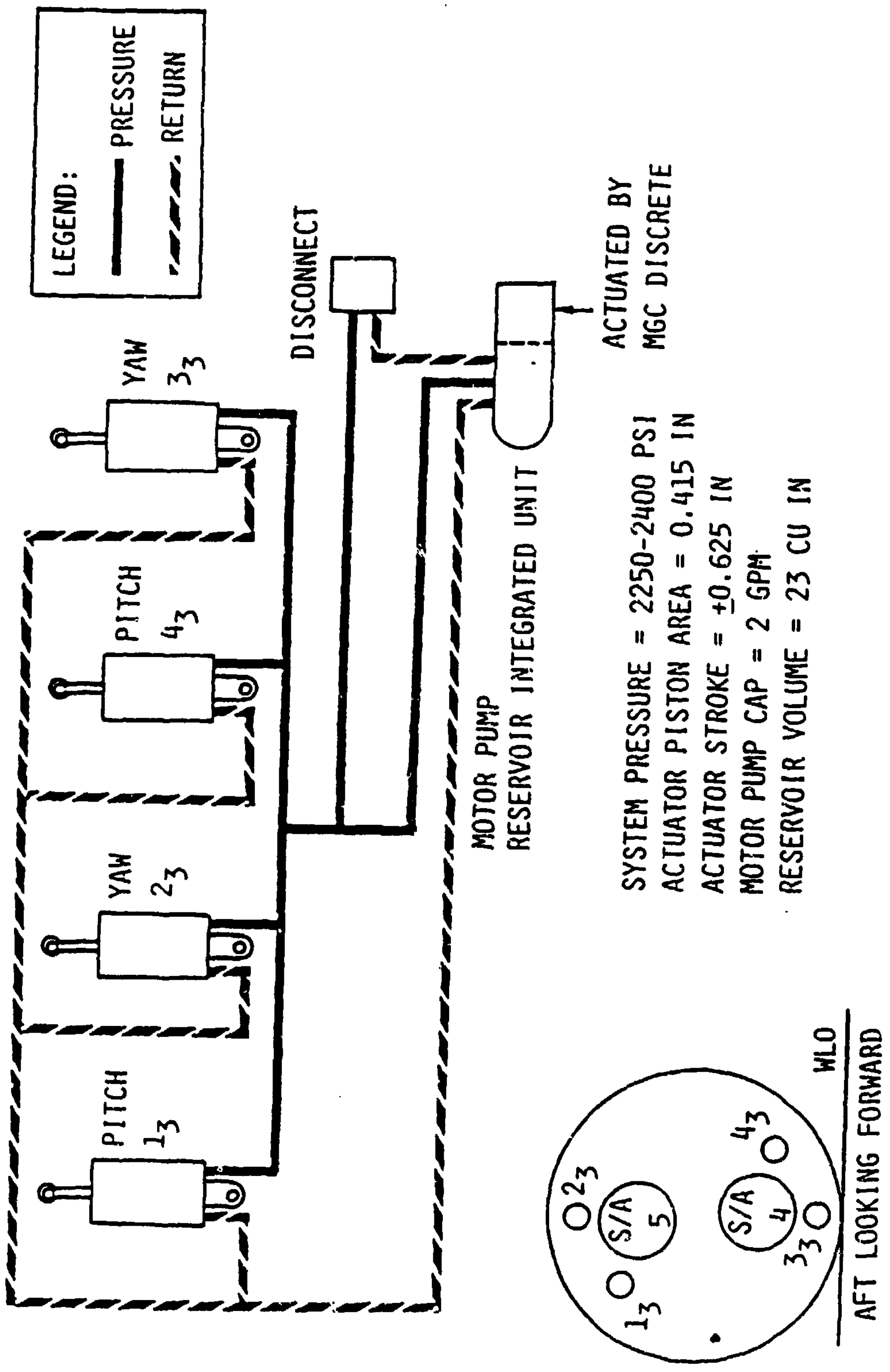


Figure C14-12 Transtage Hydraulic System

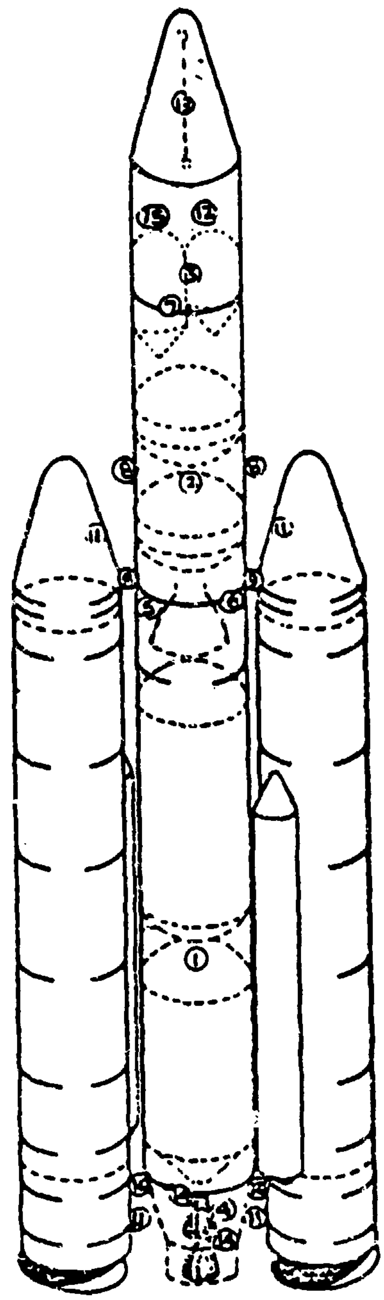


Figure C14-13 Vehicle Location of Ordnance

Fourth Stage of Titan IIIC, Being Integrated on T34D

Performs
- Orbital Transfer from Low-Earth Orbit to Higher Energy Orbits
- Spacecraft Thermal Control and Telemetry Maneuvers During Transfer
- Orbit Circularization and Plane Change Maneuvers at Synchronous Altitude
- Spacecraft Separation and Postseparation Maneuvers

25 Out of 26 Successful Missions Since 1966

Up to Eight Payloads Deployed on Single Flight

Mature, Flight-Proven Payload Delivery System

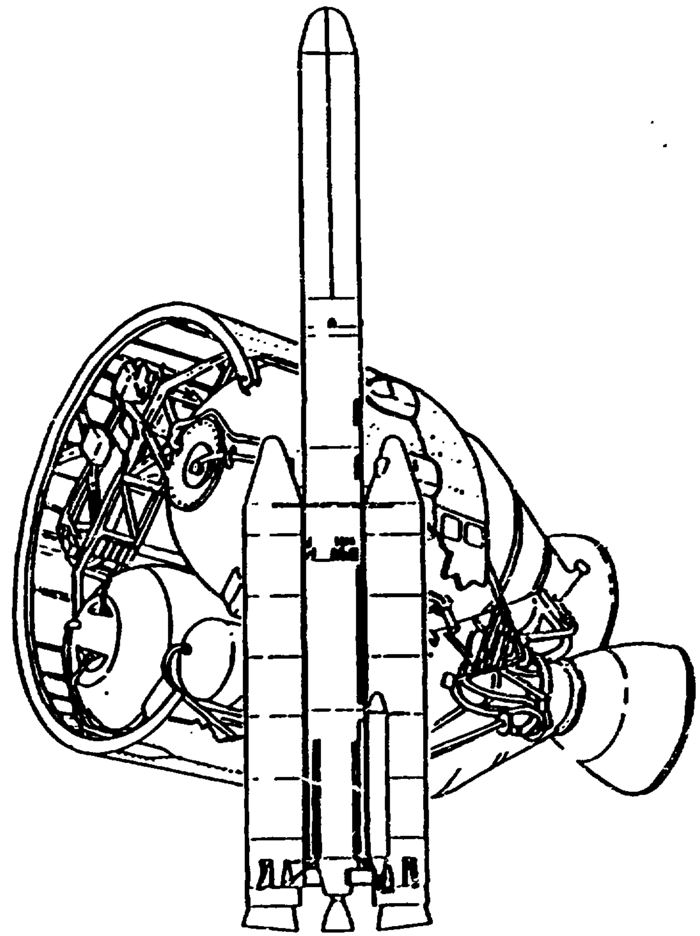


Figure C14-4 Transtage Performance

- Remote Manipulator System (RMS) Is Baseline
- Dual-Fault Latches Open on Cradle to Release Transtage
- RMS Reach Is Adequate for the Most Aft Location in Cargo Bay and Aft Cargo C.G. Location
- Separation Provided by Orbiter RCS
  - ACS Turned on at 300 ft Clearance
  - Translage Ignition at 3000 ft Clearance

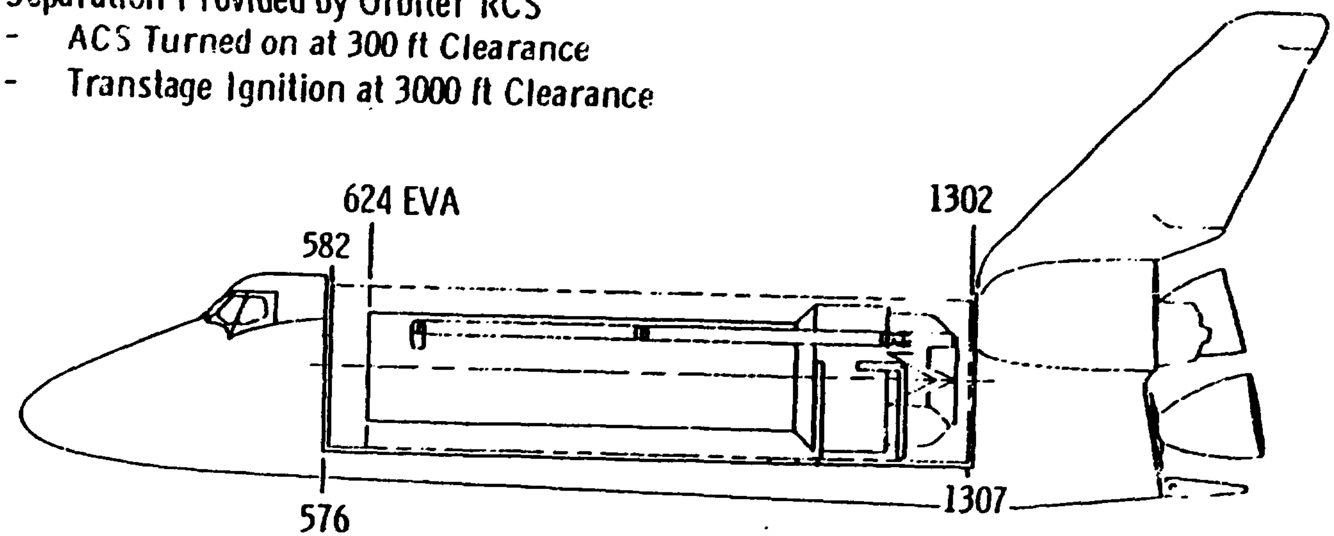


Figure C14-15 Transtage Deployment from Shuttle

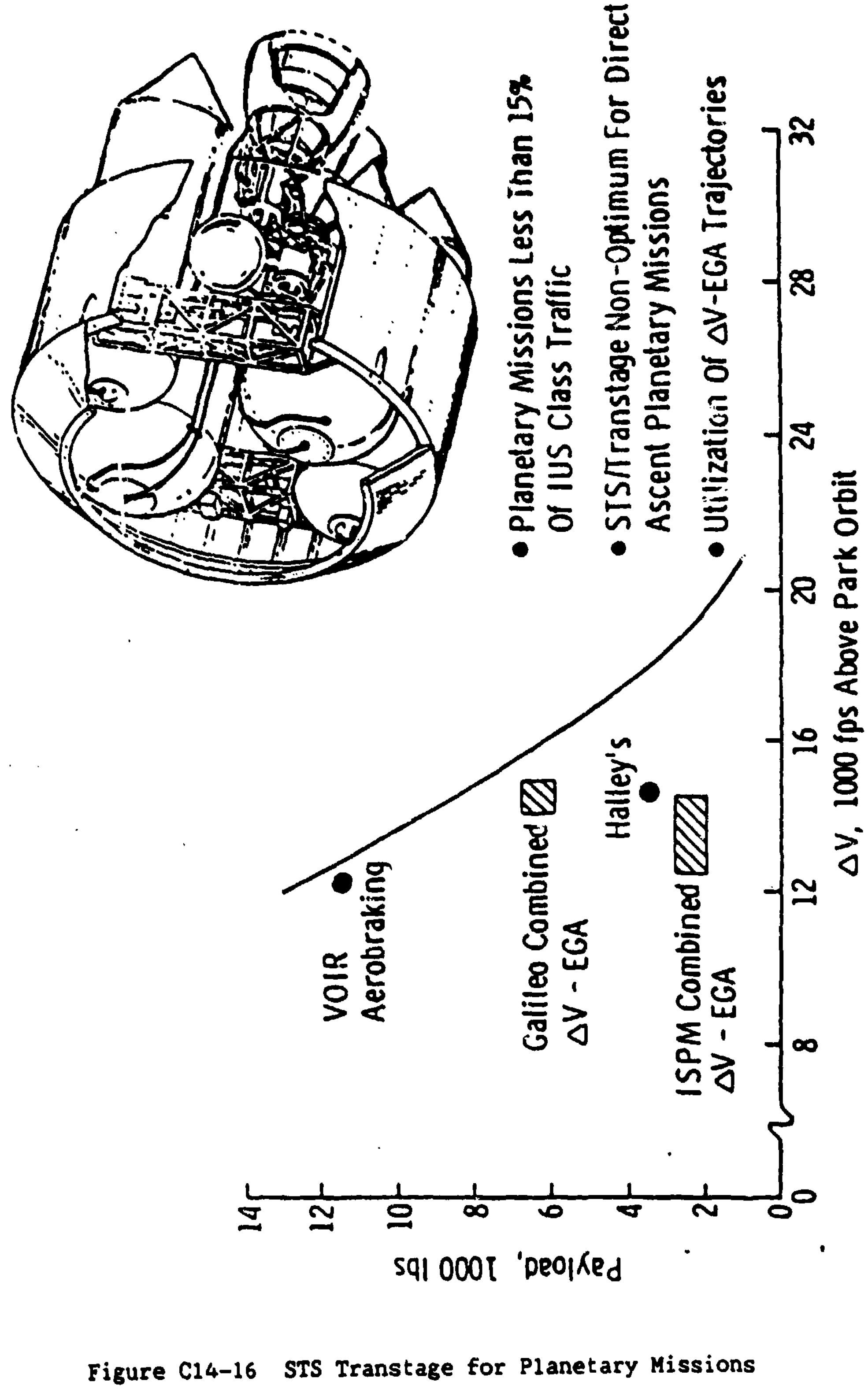


Figure C14-16 STS Transtage for Planetary Missions

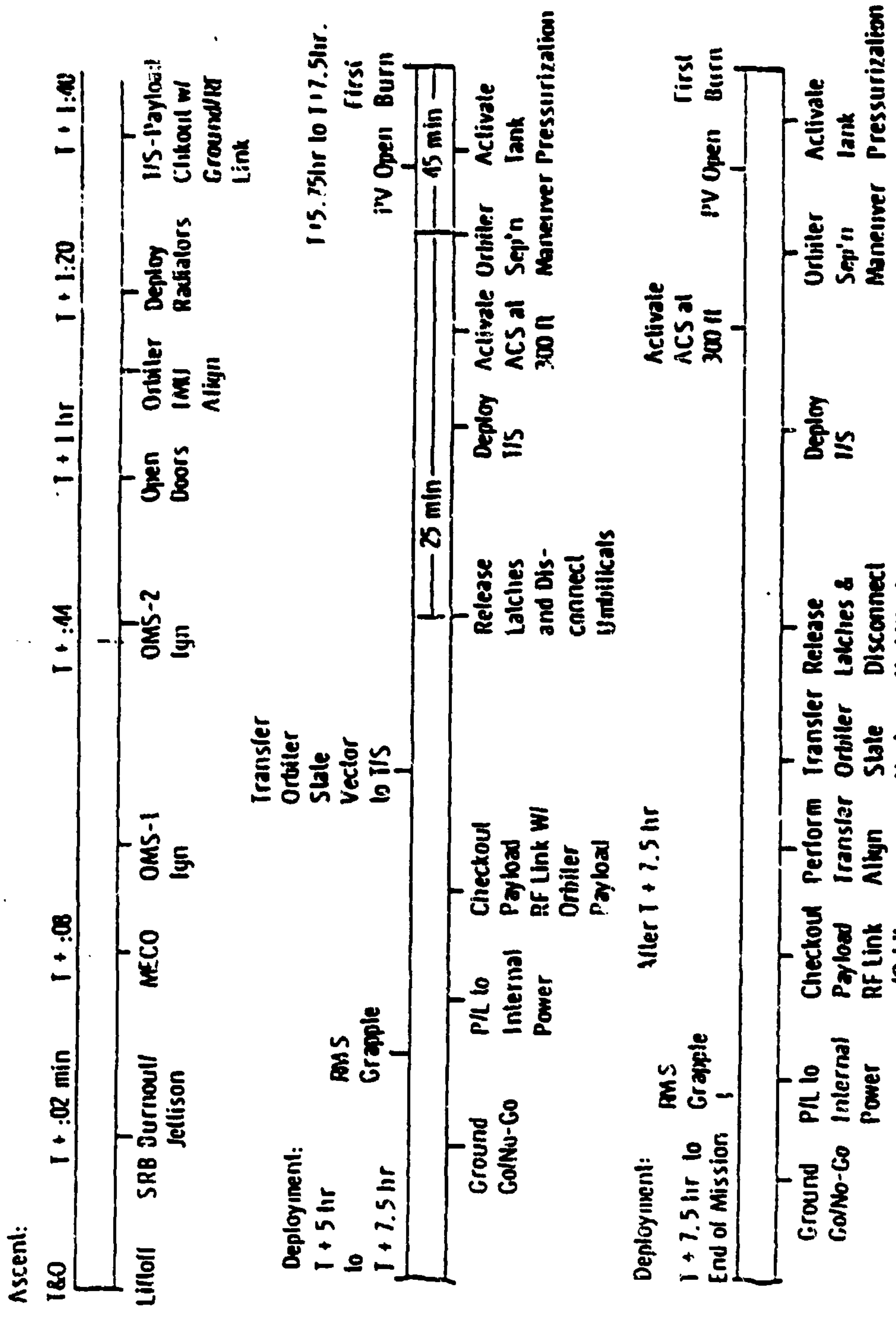


Figure C14-17 Transtage Flight Operations

Appendix C15
Defense Satellite Communication System

APPENDIX C15

DEFENSE SATELLITE COMMUNICATION SYSTEMS (DSCS)

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APPENDIX C15

DEFENSE SATELLITE COMMUNICATION SYSTEMS (DSCS)

C15.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref. 016, Section 7.1 of the ARAA, Volume II, DSCS II/III Payload Annex; DSCS ARA?, General Electric Space Division, Valley Forge Space Center, Philadelphia, PA, SVP1027B, April 1983; and DSCS Launch Base Test Plan, General Electric Space Division, Valley Forge Space Center, Philadelphia, PA, SVP3021, 15 Mar 1983.

C15.1 GENERAL DESCRIPTION

The goal of the DSCS III program is to provide an operational network of long-life military communications satellites.

The satellite can be commanded both at SHF (X-band) and S-band to allow routine housekeeping, orbit maintenance, and communication subsystem reconfigurations by the Air Force Satellite Control Facility and DSCS ground stations. It can also autonomously receive multiple distributed SHF user inputs and provide rejection of jammer interference.

The Satellite is designed for launch from the Air Force Eastern Launch Site (ELS) on a Titan 34D as a dual launch in combination with a DSCS II or as a single or dual DSCS III launch on the Space Shuttle.

The DSCS Payload consists of a DSCS II (II-E15) and a DSCS III (III-A2) Satellite (including the bipod adapter assembly), launched in tandem, using a TITAN 34D/TS launch vehicle with a MDAC Payload Fairing as illustrated in Figure C15-1. The DSCS II Satellite is the "Forward" satellite of the stacked configuration, with the DSCS III Satellite being the "Aft" satellite. The launch will take place from Cape Canaveral Air Force Station using the Integrate-Transfer-Launch Complex-40.

5.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C15.2.1 Structure Subsystem

The structure subsystem provides the framework to support and house all the DSCS satellite equipments. It also includes the mechanism for:

1. Joining the stacked satellite assembly to the Launch Vehicle;
2. Retention and separation of the forward and aft satellites;
3. Retention, release, and deployment of the solar array and Gimballed Antenna;
4. Rotation of the deployed solar arrays to follow the sun.

The structural function of the subsystem is achieved by the following assemblies that house or support the spacecraft equipments:

1. Equipment Bay Structure
2. North/South Panels
3. Antenna Supports
4. Solar Array Substrates
5. Launch Vehicle Adapter
6. Propulsion Module Adapter

Other structural elements include the graphite epoxy tube trusses and fittings to support the propulsion subsystem, the solar array assembly yoke, and the aluminum-honeycomb-sandwich substrates for the solar arrays.

Separation of each satellite is achieved by the pyrotechnic release of four preloaded captivated separating nuts and matched separation springs to achieve a separation velocity between 1 and 2 fps with tip-off rate of less than 1 degree er second.

Release and deployment of the solar arrays are obtained by the pyrotechnic actuation of four preloaded nuts and hinge torsion springs with synchronization mechanisms to control unfolding of the two panels and yoke.

C15.2.2 DSCS II

The DSCS II satellite is made up of a spinning section that provides gyroscopic stabilization and a despun section upon which the Earth-oriented antennas are mounted (see Fig. C15-2). The satellite is nine feet in diameter and approximately six feet high. It contains the following subsystems:

1. Communications Subsystem
2. Telemetry, Tracking and Command Subsystem
3. Attitude Control Subsystem
4. Electrical Power Subsystem
5. Electrical Distribution Subsystem
6. Thermal Control Subsystem
7. Propulsion Subsystem

The entire communications payload equipment and the narrow beam antenna gimbal drives, a deployable mass boom, primary power isolation modules, a PCM telemetry multiplexer, and the switching logic assembly are located on the despun platform.

Four pressure-fed propellant tanks are mounted one in each quadrant so that their combined center of gravity is along the spin-axis of the satellite. Two solid-propellant spin rockets, attached to the primary structure members below the lower solar array support ring, provide initial spinup after separation.

C15.2.3 DSCS III

The DSCS III satellite is an integrated design configured to support the high-performance communications payload. The satellite has 3-axis attitude control and stabilization, a fixed payload antenna platform, a gimballed antenna, large north/south viewing panels for passive heat rejection, and an oriented solar array for efficient power operation. The structurally integrated design is completely modular for ease of assembly, test, and maintenance. Communication subsystem components are mounted on the north panel with the housekeeping subsystem components and SCT components mounted on the south panel. This grouping arrangement minimizes cable weight and waveguide runs, takes advantage of structural shielding for hardening, and provides an equivalent Faraday Cage around the component electronics for satellite charging and system-generated electromagnetic pulse protection. See Figures C15-3 and C15-4 for further configuration description.

The DSCS III satellite consists of the following subsystems:

1. Structure
2. Communications
3. Telemetry, Tracking, and Command
4. Single-Channel Transponder
5. Propulsion
6. Attitude Control
7. Electrical Power and Distribution
8. Thermal Control

Table C15-1 lists the various performance parameters and design attributes of the DSCS III satellite and its subsystems.

Table C15-1 DSCS III Satellite Description

SYSTEM DEFINITION	
o Type Satellite	6 channel SHF Comm. Relay with anti-jam protection and nuclear hardening
o Launch Vehicle	TITIC, T34D/IUS, T34D/TS
o Launch Configuration	I DSCS II + DSCS III
o Launch Site	Eastern Test Range - LC-40
o Operational Scenario	
Nominal Oper. Longitudes	56°E, 175°E, 16°W 135°W
Ground Control	AFSCF and DSCS Control Stations
o DSCS III Satellite	III-A1, II-A2
o Ground Storage	2 years
o Launch Window	Daily (Minimum one hour per day)
THERMAL CONTROL	
o Type	Passive with heaters
o Heater, Number	114
o Heater Power	157 Watts (24 hour average)
PROPULSION	
o Type	Monopropellant Hydrazine
o Tanks, Number	4
o Tanks, Type	22 inch Spherical
o Pressurant	He Unregulated Blowdown
o Diaphragm	Elastomeric
o Engines, Thrust Level	1 lb. each of 16 engines at beginning of mission, 12 for Pitch, Roll, Yaw and 4 for N,S station keeping
o Propellant Tank Capacity	608.6 lbs
ELECTRICAL POWER	
o Batteries	
Number/Amp Hours/Cells	3/32 each/16 each
Depth of Discharge-No Failure /1 Batt. Failed	65%/80%
o Power Controllers	1 North Panel, 1 South Panel, 1 SCT
o Ordnance Controller-Controls	13 Primary EED's 13 Backup EED's
o Harness Shielding	2 Mil Mylar + 2 Mil Aluminum
o Grounding	Multi-point
o Solar Array	980 Watts End of Mission

C15.2.4 Communications Subsystem

The Communications Subsystem (COMM) is composed of a transponder panel containing six independent RF channels, a multibeam receiving antenna, two Earth-coverage horn receiving antennas, two multibeam and two Earth coverage horn transmitting antennas, a gimballed dish transmitting antenna, and supporting interfaces.

Two channels are dedicated to Earth coverage reception and transmission; the remaining four channels have the commandable option of choosing either Earth coverage or multibeam antenna inputs. However, for transmission, each high-power channel is connected to a transmitting multibeam antenna, and each low-power channel has the option of Earth coverage or of sharing a multibeam antenna with a high-power channel. In addition, both of the high-power channels and one low-power channel are switchable to a high-gain gimballed antenna for purposes of enhanced effective isotropic radiated power (EIRP) performance.

C15.2.5 Propulsion Subsystem

The DSCS III Propulsion Subsystem is designed to deliver the impulses required to perform the following orbit adjust and attitude control functions:

1. Orbit Adjust Functions
   a. Removal of Injection Velocity Errors
   b. Initial Orbital Circularization and Station Attainment
   c. East-West and North-South Stationkeeping
   d. Satellite Longitudinal Repositioning

2. Attitude Control Functions
   a. Initial Stabilization of the Satellite
   b. Momentum Wheel Unloading
   c. Attitude Correction during Delta Velocity Maneuvers

The Propulsion Subsystem is a monopropellant hydrazine system consisting of a propellant storage and expulsion section, a propellant distribution section, and a rocket engine section. The propulsion subsystem provides the following:

1. Storage of Propulsion Subsystem propellant;

2. Distribution and metering of propellant to the Propulsion Subsystem engines.

3. Thrusting of sufficient magnitude and duration to satisfy all satellite maneuvering and momentum exchange functions.

4. Telemetry signals sufficient to monitor the status of propulsion subsystem equipment and propellant and to detect propulsion subsystem failure.

5. Execution of commands received from the TT&C.

The Propulsion Subsystem module structure assembly supports the propellant tanks, the thruster modules, and the propellant distribution component modules. The structure assembly and handling and support equipment shown in Figure C15-5 are manufactured by GE and sent to Hamilton Standard for assembly of the propulsion components, plumbing, and test harness attachment directly on to the structure. After completion of assembly and test of the propulsion subsystem, the completed propulsion module assembly is returned to GE, where it is then ready for installation into the satellite. Installation is performed by lowering the PS module onto the satellite center body structure.

The subsystem, shown schematically in Figure C15-6, is a mass expulsion hydrazine propulsion system using helium pressurant operating in a nonregulated blowdown mode. Two component sets, each capable of performing all mission functions, in a block redundant manner, are maintained as separate protected entities by means of normally closed latching isolating valves located in the propellant transfer lines.

The propellant and pressurant are stored in four spherical positive expulsion propellant/pressurant tanks. Each tank has a pressurant fill and vent valve and a propellant fill and drain valve, which permits preferential selection of the tanks, thereby consuming unequal quantities of propellant to maintain satellite balance. Utilization of individual pressurant fill and vent valves allows each tank to be capped individually after pressurization to prevent transfer of propellant between tanks.

Propellant is normally fed from either of two diagonally opposed tanks or all four tanks. The propellant and diaphragm are oriented to expel the fluid toward the center of the satellite. All these considerations were necessary to minimize the center of mass transfer during any mode of propellant use.

Propellant from each tank is filtered through a high-capacity, low-micron rating etched disc filter. The system filters are located upstream of the isolation valves and thrusters to provide contamination protection for these components. Latching isolation valves are used to isolate propellant flow from each tank to each set of thrusters. The latching valve arrangement permits isolation of any tank or either thruster group so that no single-point failure results in loss of more than approximately one-fourth the mission propellant. Inlet filters are also included in each latching valve and thruster valve to provide protection from contamination. Co-extruded titanium-to-stainless steel transition joints are installed between the system filters and the latching isolation valves. This allows the use of low-weight titanium tanks and propellant lines with stainless steel valves and pressure transducers.

The propellant feed system components, consisting of the latching valves, filters, and pressure transducers, are packaged into two modules. Each module is supported by an aluminum bracket that also provides satellite mounting interfaces. Locating these propellant feed system components into a modular group permits mounting the assemblies at only two satellite interfaces, simplifying the support structure, minimizing heater power requirements, and providing common access regions for assembly, checkout, and maintenance.

Thrusters are positioned on the satellite in packages called Thruster Modules (TM). A TM consists of thruster and propellant valve assemblies (see Fig. C15-7), thermal control features, mounting hardware, and supporting structure. The TM support structure is a precision-machined bracket whose configuration satisfies the specific satellite mounting and thruster assembly thrust vector orientation requirements. By using TMs, the mechanical interface with the satellite is reduced to a simple bolted flange at each of the TM locations. The thruster arrangement requires the use of only two types of TMs. The roll thrusters are installed in two two-thruster TMs, while the pitch, yaw, and north-south thrusters are each packaged into two six-thruster TMs. Each of the TM arrangements is designed to interface with the satellite's bulkhead structure.

Each thruster is controlled with a pair of normally closed series-redundant solenoid-operated propellant valves. The thrusters have redundant-thrust chamber heaters to provide an elevated chamber temperature. Temperature sensors are mounted on all the thrusters and on each of the propellant tanks.

There are eight thrusters in each component set, each operating at an initial thrust level of 1.0-pound force. These eight thrusters provide the satellite with north-south and east-west orbit adjustment and three-axis attitude control.

In operation, the Propulsion Subsystem is controlled by means of the attitude control subsystem (ACE), which contains the valve driver electronics. Ground control is used to energize the ACE into either the "Operational" or the "Orbit Adjust" thruster control modes, depending on whether the satellite function of initial stabilization, wheel desaturation, or orbit adjustment is required. When operating in the "orbit adjust" mode, in-track and cross-track velocity corrections are accomplished by operating pairs of thrusters designated North, South, East or West. Thruster catalyst bed heaters are ground-command activated "on" 100 minutes prior to thruster rise and are turned "off" after the use is completed.

During the "Operational" mode of ACS control, where wheel unloading is required, six thrusters and their six respective thrust chamber heaters are energized automatically in readiness for momentum wheel desaturation. The appropriate pitch, roll, or yaw axes thrusters are automatically selected by the ACS for accomplishing wheel unloading.

Table C15-2 provides the Propulsion Subsystem fill and drain requirements for flight. Hydrazine is a toxic fluid; therefore, personnel restrictions and safety precautions are imposed during filling or draining operations.

Table C15-2 Propulsion Subsystem Fill and Drain Requirements

Fluid	Hydrazine (N_2H_4)	Helium MIL-P-27401, Supp. I
Pressure Load	As Required	350 ± 10 psia
Mass Load	600 lb.	1.0 lb.
When: Fill	T-11/12 days	T-11/12 days
Drain	Only for Abort	Only for Abort
Pressure	T-11/12 days	T-11/12 days
Temp Loaded	Ambient	Ambient
System Oper, Press.	350 psia	350 psia
Where: Fill	PAD	PAD
Drain	PAD	PAD

Table C15-3 lists the materials selected for the DSCS III propulsion subsystem. The sealed components, including the latching valve torque motors, the valve coils, and the electrical portion of the pressure transducer, have not been tabulated because they are protected from the hydrazine.

Table C15-3 Propulsion Subsystem Materials Summary

1. Materials in Contact with Liquid N_2H_4

   304 Stainless Steel (SS)
   304L SS
   17-7PH
   430 SS
   AF-E-411 Ethylene Propylene Terpolymer (EPT)
   302 SS
   6AL-4V-Ti; titanium alloy
   AF-E-322

2. Materials in Contact with Two-Phase (Gas and Liquid) N_2H_4

   Inconel 600
   Haynes 25 (AMS 5759)
   304 SS

3. Elements in Contact with Hot Decomposition Gases

   Inconel 625
   Haynes 25 (AMS 5759)
   304 SS

C15.2.6 Thermal Control Subsystem (TCS)

The Thermal Control Subsystem (TCS) is designed to maintain equipment temperatures during the launch and orbital phases of the mission. To perform this task within specified limits, the TCS uses a passive control design that employs thermal blankets, thermal coatings, insulation spacers, RF transparent thermal shrouds, heater assemblies, and increased panel thickness.

The batteries, installed on the South Panel, are independently temperature controlled by conductively isolating them from the panel with insulating spacers and radiation, isolating them from the satellite with insulation blankets. The batteries radiate directly to space from their baseplate, which has a sized insulation window and OSR thermal coating. Thermostatically activated electrical redundant heaters maintain minimum temperature levels.

C15.2.7 Attitude Control Subsystem (ACS)

The attitude control subsystem is a three-axis active zero momentum stabilization system using an Earth sensor and sun sensor for active attitude sensing and reaction wheels (with propulsion subsystem assist) to provide satellite control torques. All electronic processing to provide control of the subsystem is performed by the attitude control electronics (ACE).

The ACS orients and stabilizes the satellite, provides attitude control during initial stabilization and inclination correction, maintains pointing during payload operations, and controls satellite attitude during orbit adjust operations. The subsystem consists of the following:

1. Redundant Earth Sensors
2. Redundant Sun Sensor Assemblies
3. Four Reaction Wheels
4. Yaw Rate Gyro
5. Attitude Control Electronics (ACE)

The ACE interfaces with all ACS components and provides subsystem electrical interface functions with other satellite subsystems. The ACE controls the positioning of the Solar Array Drive and the Gimballed Antenna. The ACE provides the interface control between the telemetry, tracking, and command (TT&C) and the Beam Forming Network. All ACS functions, from booster separation through on-orbit operation, are controlled by a Firmware program stored in the ACE. By using combinations of ground-issued discrete and message commands, functional modes required for each mission phase are selected. Mode changes and selections of functional blocks are implemented by discrete commands. Selected program constants are changed by serial messages. The ACS can operate for a minimum of 30 days without ground commanding after initial stabilization. Inflight monitoring of the ACS performance is provided by analog, bilevel, and serial digital telemetry data.

After separation, the solar arrays are deployed in the "noon" position; i.e., the solar cells face along the minus yaw axis, away from the multibeam antennas (MBAs). The sun sensors mounted on the solar array yokes provide pitch and roll data to the ACE, which activates the Propulsion Subsystem engines to rotate the satellite until the negative yaw axis faces the sun within the ACS error bands. After noon, stabilization to Earth is initiated. The two-axis sensor provides pitch and roll data to the ACE, which activates the proper engines to remove the pitch and roll errors. The gyro senses rates around the yaw axis. The gyro signals are processed by the ACE to drive the yaw axis thrusters to reduce the rates. The solar array is advanced to the proper angle corresponding to the satellite position in orbit and is clock synchronized to orbit rate. Yaw acquisition is initiated when proper vehicle orbit/sun geometry is attained. The array-mounted sun sensors now provide yaw information. Reaction wheel control is then initiated. Four wheels are nominally employed to fulfill the momentum exchange function, although any three can provide the required capability. Unloading of the accumulated secular angular momentum is accomplished once a day, permitting the propulsion subsystem to be disabled most of each orbit.

C15.2.8 Electrical Power and Distribution Subsystem (EPDS)

The EPDS is a direct-energy-transfer design consisting of a sun-oriented solar array, three 32-amphour batteries and a +28 Vdc boost regulator. The power subsystem shall provide the following:

1. Generation and storage of electrical power (energy);
2. Distribution of conditioned power to users from internal or external sources;
3. Distribution of electrical signals (excluding RD signals) between equipments and protection of such interconnecting cabling from effects of electromagnetic interference;
4. Protection of the power bus from detrimental transient loads, shorts, and effects of load sequencing and removal;
5. Detection of electrostatic discharge (ESD) phenomena and provision of ESD detector signals to the TT&C Subsystem;
6. Telemetry signals sufficient to monitor the EPDS status and to diagnose EPDS faults;
7. Execution of commands received from the TT&C Subsystem.

The EPDS provides for the conversion of solar energy to electrical power, the regulation and distribution of power to the satellite subsystems (load), and the battery storage of a portion of the electrical energy for subsequent use during periods when the solar array capability is inadequate. The EPDS provides the electrical power for all subsystem modes of operation from prelaunch checkout, through launch, and through complete mission life.

Batteries provide power during prelaunch, launch, and ascent for on-orbit eclipse operations and provide fuse blow current to clear faults. Three 32 ampere-hour sealed nickel cadmium 16-cell batteries satisfy the total energy storage requirements with conservative depths of discharge to provide the ten-year design life. The batteries are safely charged using four selectable voltage limit functions that are modified by sensed battery temperature. Each battery has dedicated charge and reconditioning circuitry. A reconditioning cycle can be initiated by ground command. The discharge can proceed to a battery terminal voltage of 1 Vdc.

The redundant electronic power conditioning circuits are inherently on-line for use without relay or contactor transfer, and the transition from battery discharge to battery charge to shunt occurs without a disturbance to the regulated bus.

C15.2.9 DSCS II/DSCS III (A-2) on Titan 34D/Transtage

The following is a description of the DSCS II/DSCS III/T34D configuration and its interfaces:

The satellite is configured to take advantage of the Titan T34D maximum payload envelope and to provide efficient and straightforward launch vehicle interfaces and satellite-to-satellite interfaces (DSCS II/DSCS III). The major external interfaces of the structural subsystem are with the T34D launch vehicle and between satellites. The T34D launch vehicle interfaces are defined in the Interface Specification. The DSCS II to DSCS III interface has been established and is defined in the DSCS II/III Interface Specification and the DSCS II/III Mechanical Interface Drawing. In addition to resolution of physical interface, the analytical interfaces between the DSCS II Satellite Contractor and Launch Vehicle Contractor have been established and verified by the DSCS II/III - T34D Verification Loads Cycle Analysis.

The electrical interfaces between the launch vehicle and the satellite consists of three categories:

1. The forward-satellite (DSCS II) separation signals to DSCS III Ordnance Controller.

2. The aft satellite (DSCS III) separation EED power.

3. Forward and aft satellite separation indication.

The launch vehicle provides signals to the DSCS III ordnance controller to initiate separation of DSCS II from DSCS III. The launch vehicle also provides power to the DSCS-III-supplied EEDs that release the DSCS III satellite from the bi-pod adapter on the launch vehicle. The payload, comprised of DSCS II, DSCS III, and the bipod adapter, provides signals to the launch vehicle that indicate separation of the forward and aft satellites.

Upon receipt of discrete signals from the launch vehicle, the DSCS III ordnance controller provides a current pulse from each of two batteries to each of eight EEDs for DSCS II satellite separation. See Figure C15-8, satellite separation signals block diagram of the interfaces between launch vehicle and DSCS III and between DSCS III and DSCS II.

The ordnance controller contains the relays to switch power from the batteries to the electro-explosive devices (EEDs) used for separation of the forward satellite from the aft satellite and for deployment of the solar array wings and gimballed antenna. Redundant EEDs are used for each of the separation events, which results in the following number of EEDs associated with each function.

Function	EEDs Fired	Explosive Nuts
Satellite-to-Satellite Separation	8	4
Deploy North Solar Array	8	4
Deploy South Solar Array	8	4
Deploy Gimballed Antenna	2	1

C15.3 MISSION SCENARIO

Approximately 5-1/2 hours after lift-off, the DSCS II (forward) satellite is separated from the DSCS III (aft) satellite using ordnance power supplied by the aft satellite upon receipt of discretes from the transtage. Satellite separation activates separation switches, which in turn start the ordnance sequences within the EIQ. These sequences time the arming and firing of the following ordnance functions: omni antenna release at separation plus 11.06 seconds; despin platform release at separation plus 13.01 seconds; and spin-up rocket operations at separation plus 22.76 seconds. Arming and firing of the narrow-coverage-antenna release ordnance and the mass-boom deployment ordnance are accomplished through ground command.

C15.3.1 DSCS II Satellite System

The DSCS II satellite consists of two sections: a cylindrical spinning section and a despun platform. The satellite spin axis is oriented perpendicular to the near-equatorial orbit plane such that sun impingement is within ±27.5° of being orthogonal to the cylindrical solar array. Spinning provides a gyroscopically stabilized satellite and tends to distribute heat input from the sun such that a benign thermal environment is produced for internal components. The nominal satellite weight in orbit after spinup and antenna deployment is about 1360 lb. The satellite has a favorable moment of inertial ratio, being about 1.2 at beginning of life with antennas deployed, decreasing to about 1.1 at end of life. The despun platform, containing the communication equipment, is servoed about the spin axis to direct its Earth-coverage receive-and-transmit antennas continuously toward the Earth. In addition, two gimballed parabolic narrow-beam antennas also mounted on the despun platform are continuously pointed to any selected visible ground area.

C15.3.2 DSCS III Satellite System

The DSCS III satellite provides a 6-channel SHF communications relay satellite continuously operable in a synchronous equatorial orbit at any longitudinal station.

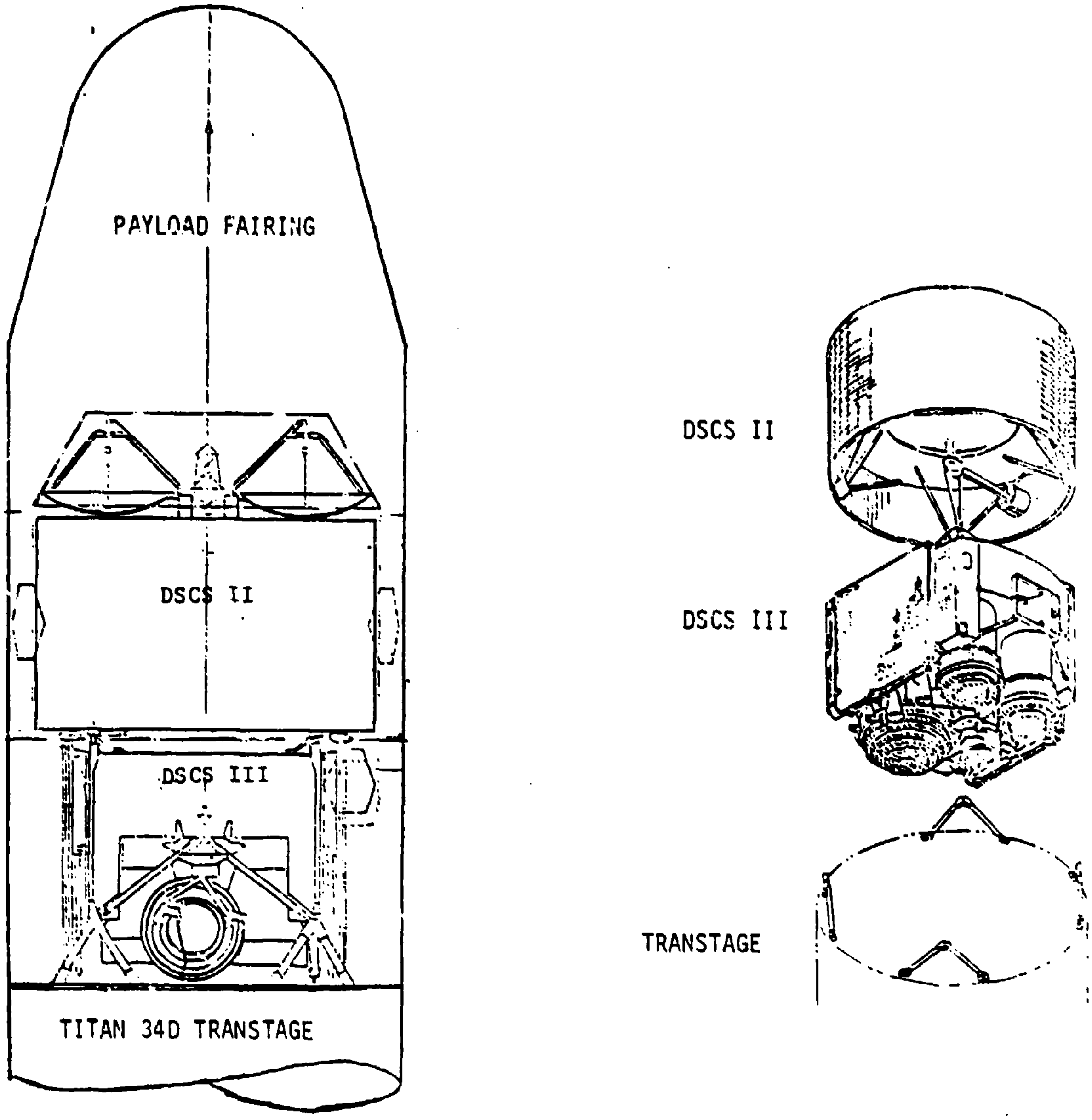

Figure C15-1 DSCS Payload Launch Configuration

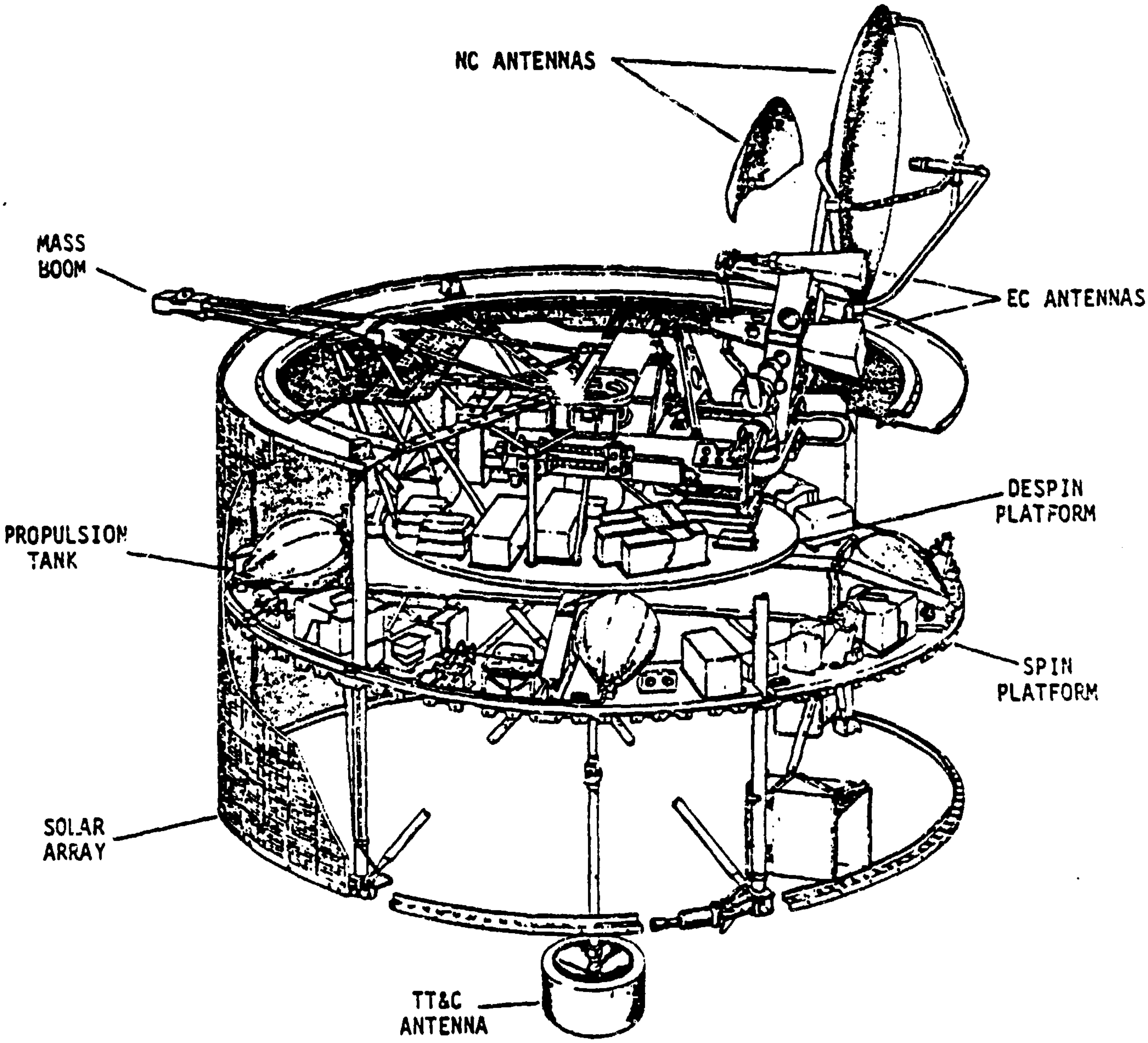

Figure C15-2 DSCS II Satellite Configuration

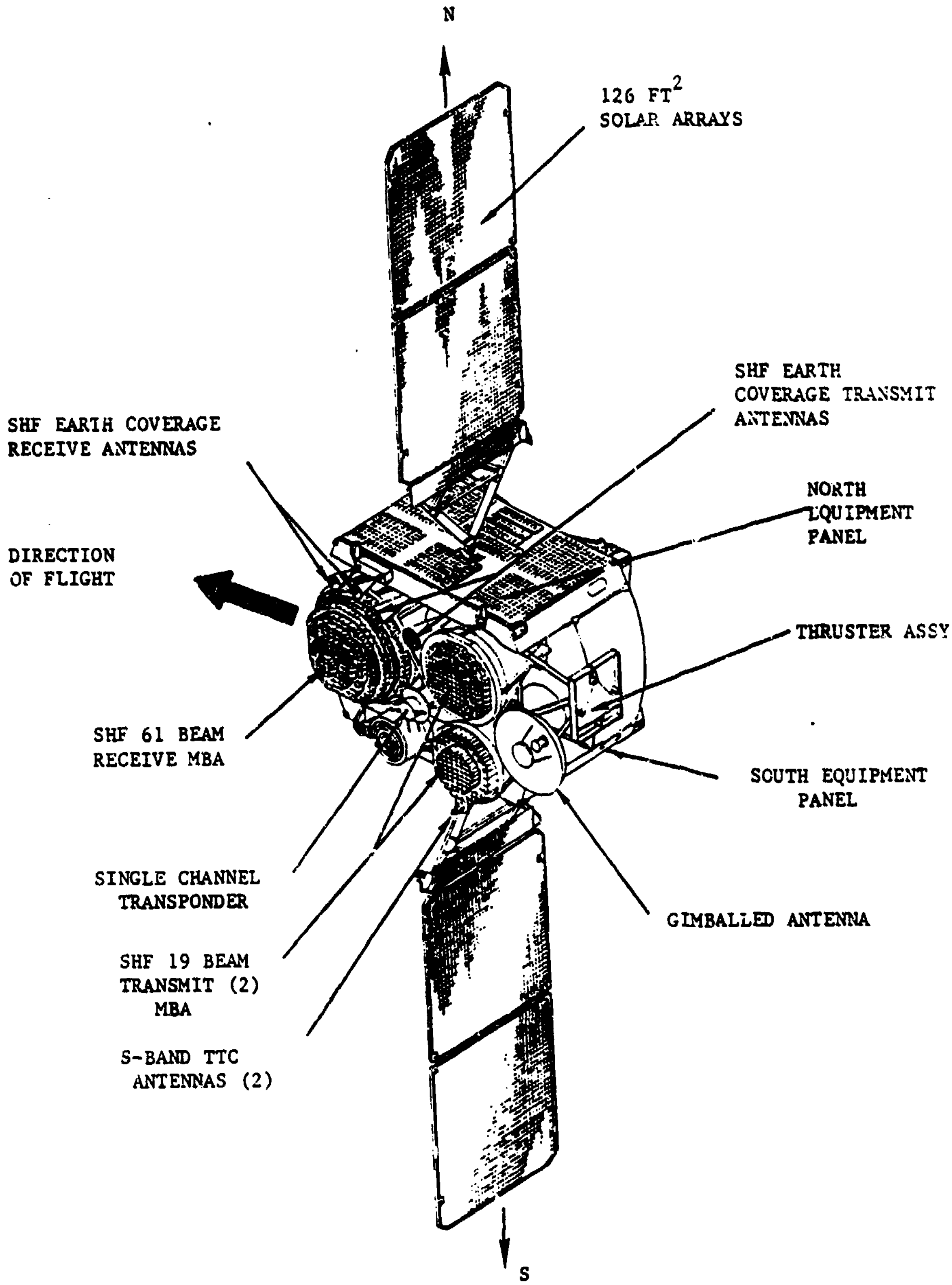

Figure C15-3 DSCS III Satellite Orbital Configuration

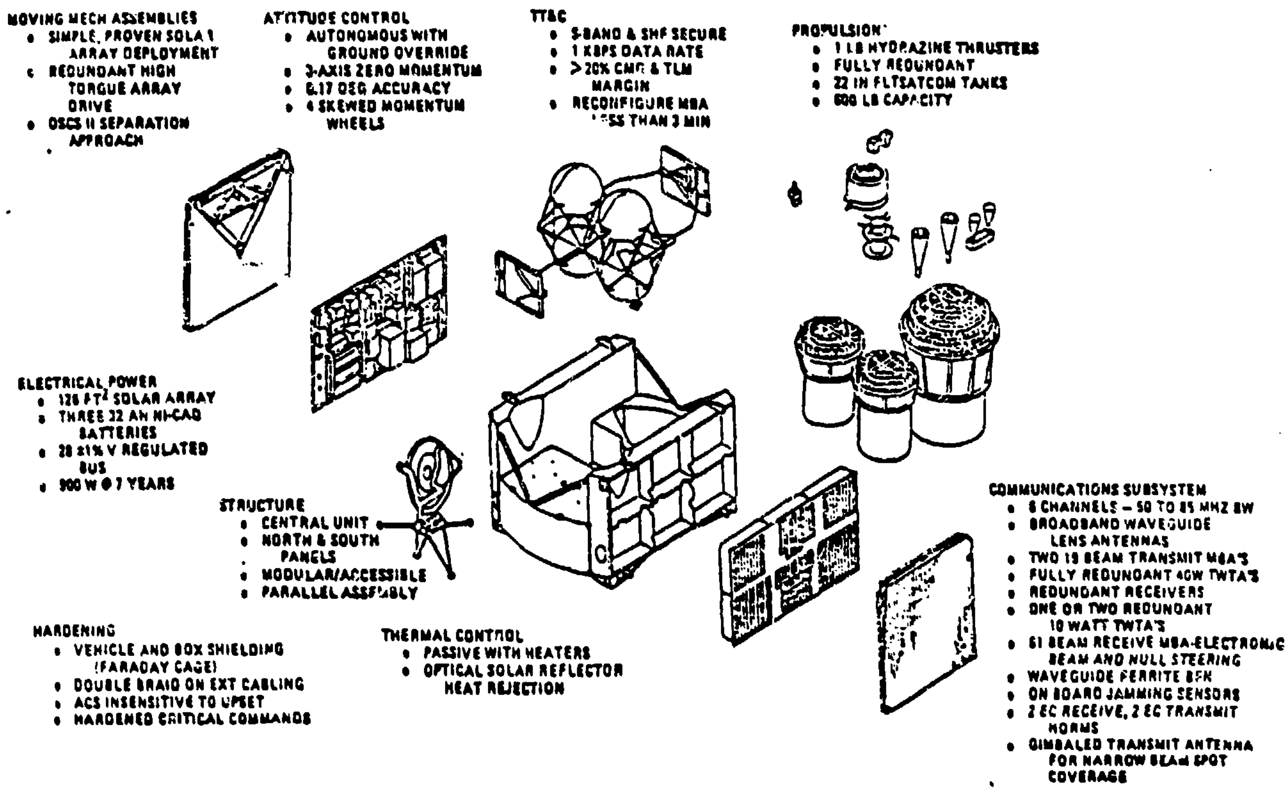

Figure C15-4 DSCS III Assemblies

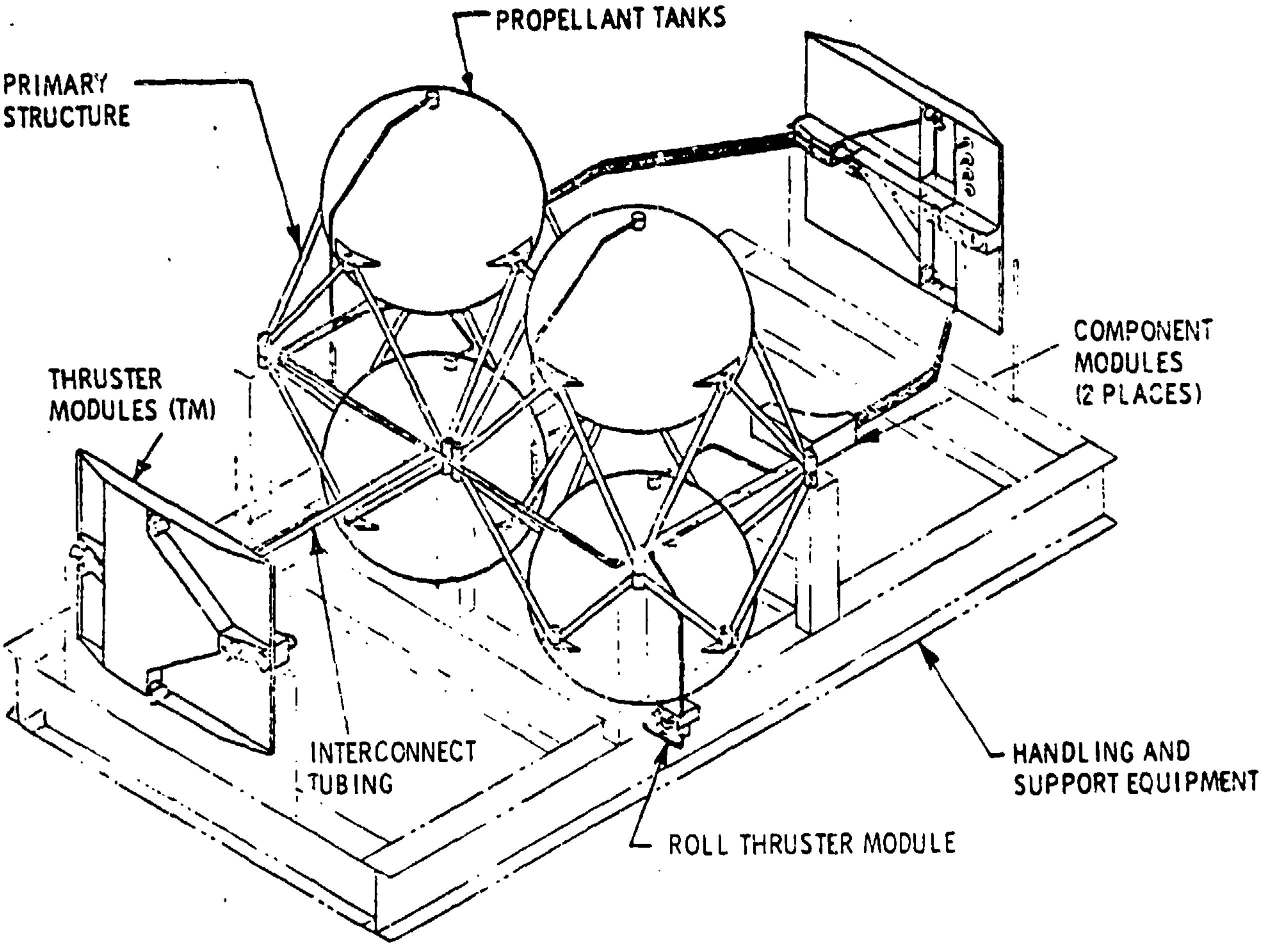

Figure C15-5 Propulsion Modular Design

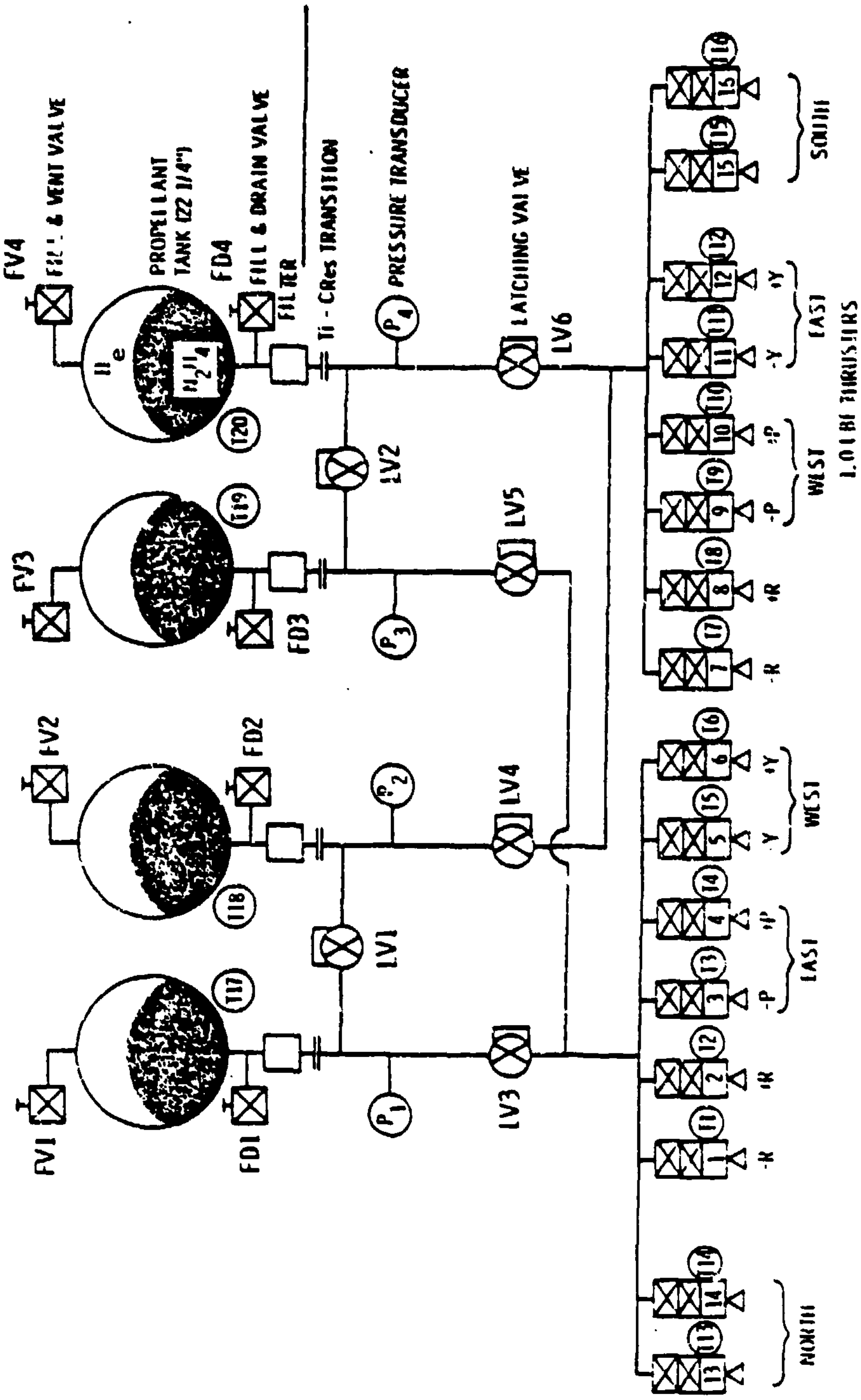

Figure C15-6 Propulsion Subsystem Block Diagram

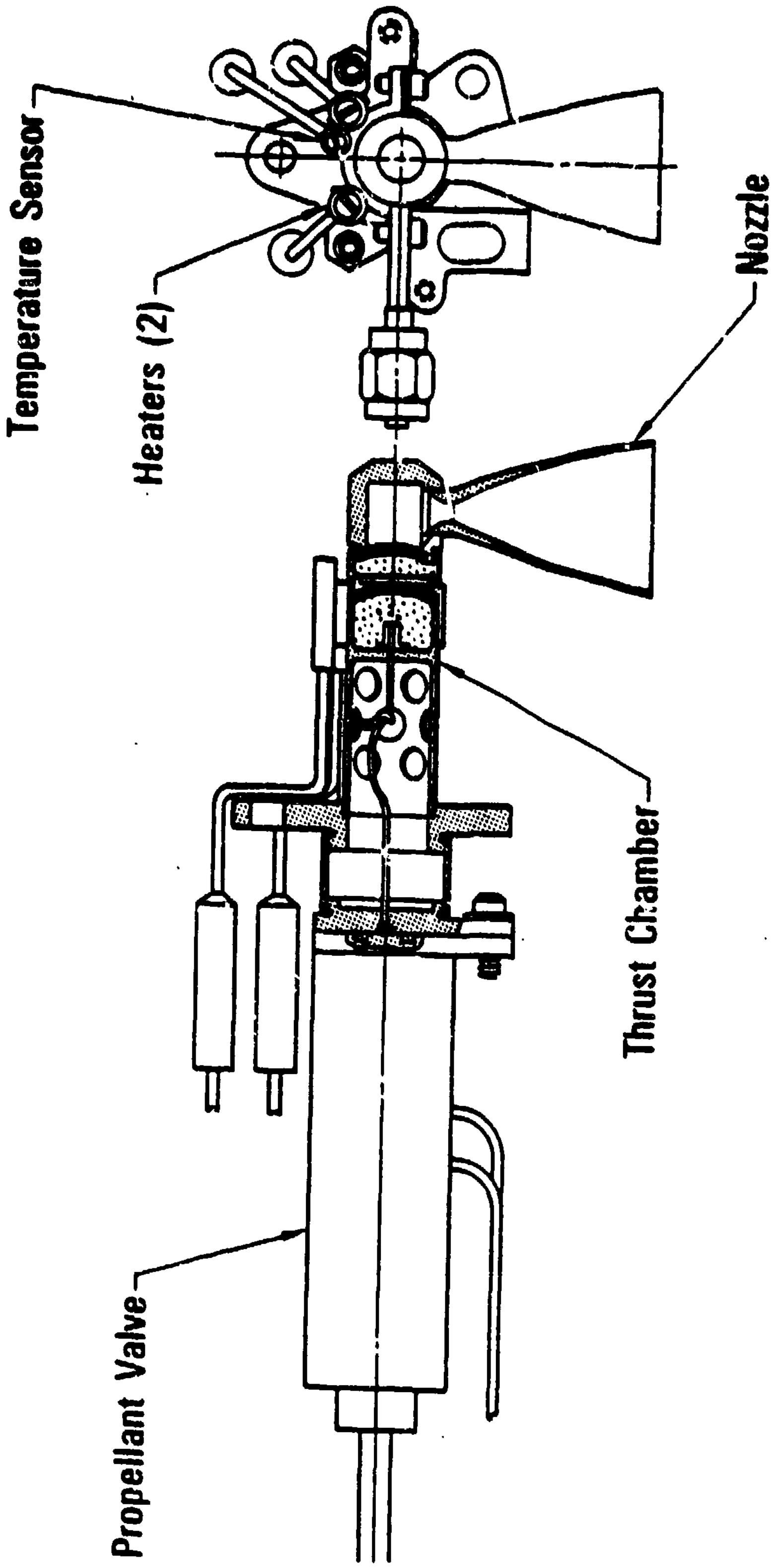

Figure C15-7 Thruster and Propellant Valve

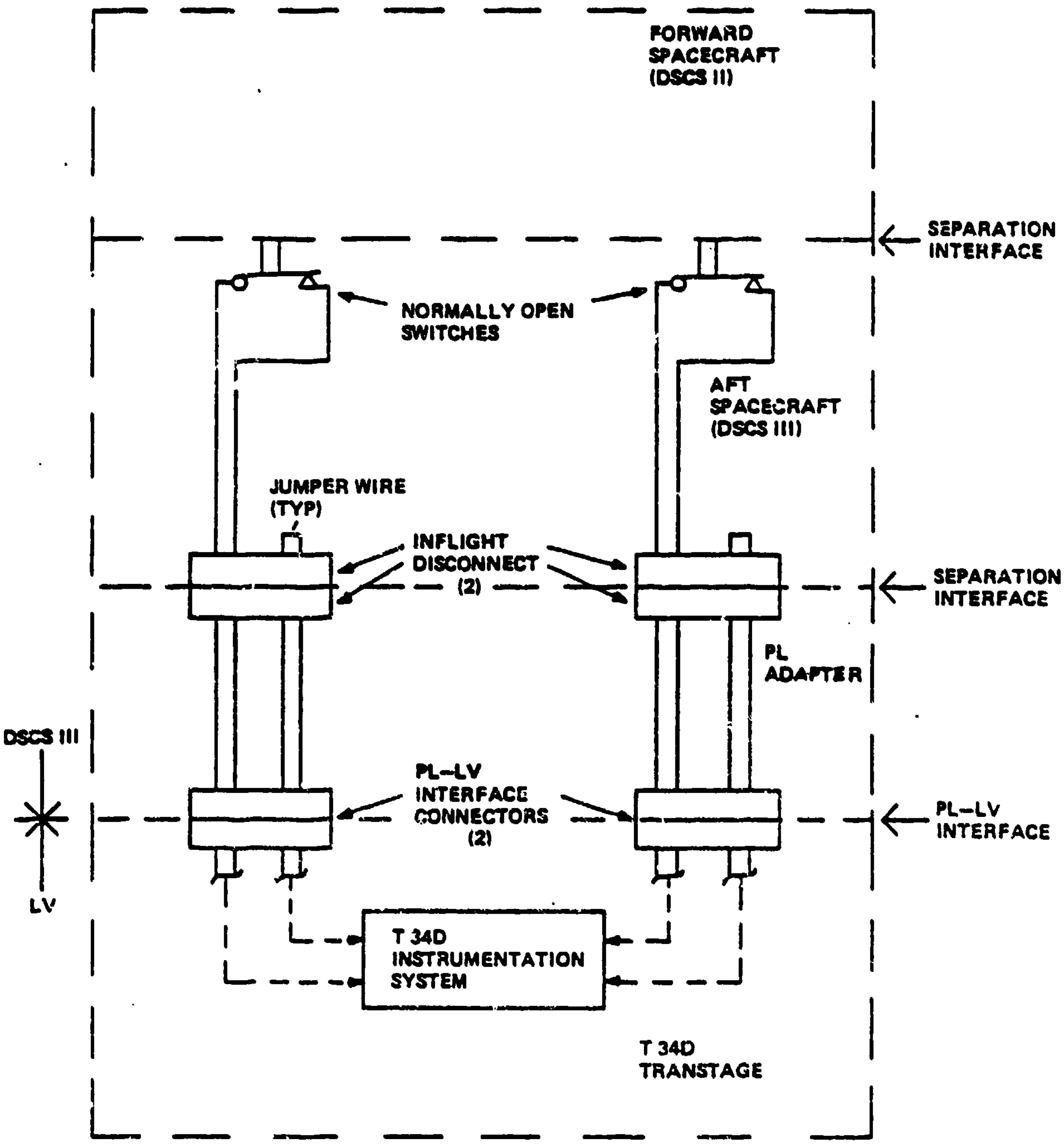

Figure C15-8 Satellite Separation Signals

Appendix C16
Defense System Program

APPENDIX C16
DEFENSE SYSTEMS PROGRAM (DSP)

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APPENDIX C16
DEFENSE SYSTEMS PROGRAM (DSP)

C16.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Document, Ref. 312.

This section addresses the DSP 5R/6R payload being integrated to the T34D/TS Launch Vehicle at Launch Complex 40, Cape Canaveral Air Force Station (CCAFS), Florida.

C16.1 GENERAL DESCRIPTION

Figure C16-1 shows the relative size and shape of the satellite with solar paddles stowed in the launch mode.

Key requirements driving Satellites 0005R and 0006R are as follows:

o - Accommodation of Sensor Evolutionary Development (SED) sensor

o - Five-year lifetime design goal

o - T34D/IUS or T34D/Transtage launch vehicle compatibility

o - High-power option of Link 1.

Incorporation of the SED sensor with its new capability leads to the requirement for improved pointing accuracy, reduced jitter, and an additional IPAD orthogonal to the previously existing one. The SED sensor also leads to the requirement for more power, and because of its increased mass it contributes to the requirement for a larger reaction wheel. More power results in larger solar array paddles.

The five year lifetime design goal, in combination with the primary mission, means that the orbit inclination must be controlled after about three years. This results in the requirement for larger propellant tanks on these satellites than on previous DSP spacecraft.

Figure C16-2 depicts Satellites 005 and 006 retrofit configuration.

C16.2 DESCRIPTION OF SAFETY CRITICAL SUBSYSTEMS, HAZARDOUS MATERIALS, SCHEMATICS

C16.2.1 Spacecraft Structure Subsystem

The spacecraft structure subsystem is the basic framework of the satellite providing physical space and attachment points for mounting all other spacecraft subsystems and equipment (see Fig. C16-3). In addition, the structure provides a rigid reference for installation and alignment of nozzles, jets, sensors, deployable antennas, and the reaction wheel. At its upper end the structure interfaces with the AESC sensor. The lower end interfaces with the TS through the booster adapter. The subsystem also provides mechanisms for separating the satellite from the TS, deploying the solar paddles and antennas, and adjusting satellite inertia properties.

The structure is designed to have sufficient strength and rigidity to withstand the maximum (limit) loads and environments induced during the prelaunch, launch/ascent, and on-orbit phases of the mission. The most critical loads on the structure occur as responses to dynamic loading conditions. The ultimate factor of safety for all mission phases is 1.25.

The stiffness characteristics of the spacecraft structure are primarily predicated on the requirement to avoid excessive loads. This is accomplished by designing the satellite to natural frequencies that minimize dynamic coupling with the booster and between major satellite components. Acceptability of the selected resonate frequencies were verified by dynamic loads analyses, which combine analytic models of the satellite and booster, and by a modal survey on a full-scale satellite. Other stiffness considerations in structural design are to prevent satellite deflections from violating the fairing dynamic envelope and to minimize sensor pointing errors and jitter.

The primary structure consists of an eight-strut tubular truss and a compartment assembly (See Fig. C16-4). The octagonal upper section of the compartment contains a horizontal platform and eight vertical panels for mounting electronic equipment. The lower section is circular in shape and provides support for a separable propulsion module and the reaction wheel truss. Support from four sides of the propulsion section are platforms containing the RADEC II and other spacecraft equipment.

C16.2.2 Propulsion Subsystem

This section contains a brief description of the 5R/6R propulsion subsystem (PSS) and certain PSS components.

The 5R/6R PSS, shown schematically in Fig. C16-5 and shown in Fig. C16-6, is a monopropellant hydrazine system that provides the satellite with orbit and attitude control and spin rate adjustment capability. The PSS consists of two propellant tanks, four 3.5-lbf monopropellant (catalytic) engine assemblies, one gas generator assembly, eight gas thrusters, four latching isolation valves, four fill and drain solenoid isolation valves, two liquid and two gas pressure transducers, two liquid filters, one gas filter, thermal control equipment, propellant feedlines, and mounting brackets.

The delta V and high-level attitude control thrusters are identical, nominally rated at 3.5 pounds of thrust, but capable of producing 4.1 pounds of thrust at 600-psi propellant tank pressure. The delta V monopropellant hydrazine rocket engine assemblies provide velocity increments to the satellite for orbital velocity changes. Each thruster, when commanded, fires one pulse per satellite revolution. Two high-level attitude control thrusters are positioned on the satellite to provide high torque. High-level thrusters can be operated at command pulse widths of 0.025 to 1 second. Delta V thrusters can be operated at command pulse widths of 0.2 and 1.6 seconds.

The eight low-level jets are contained in two assemblies, four in each, and operate in pairs to form a low-torque couple for attitude control and spin-speed correction.

The gas generator is a monopropellant hydrazine thruster that does not have a throat and expansion nozzle. It is used to pressurize a spherical phenum tank with warm gas. A 3-watt gas generator chamber heater is hard-wired to maintain catalyst-bed temperature close to 70°F. In addition, a pre-pulse circuit is employed to operate the gas generator for 0.05 second, followed by a 1.0-second delay each time the gas generator valve receives a command to open. This results in a pre-warming of the catalyst bed and provides for smooth starts. The gas generator temperature is monitored via telemetry, and the thermistor is mounted to the generator-to-plenum interface flange.

The propulsion subsystem also includes two spherical propellant tanks containing 352 pounds of hydrazine with GN_2 as pressurant. The tanks are interconnected and operate in a blow-down mode. Pressure tranducers, electrical harness, system plumbing, isolation valves, and thermal control equipment are also part of the subsystem. All the propulsion components are mounted to the propulsion platform.

Vernier velocity maneuvers are performed using radially oriented hydrazine thrusters that are pulsed at the appropriate point in the satellite rotation relative to sun-line crossings to build up the desired delta V. The ACS provides the timing and pulse command functions. Because of possible misalignments of the delta V thrusters from the satellite center of mass, larger than normal pointing and spin-speed disturbances can arise when they are pulsed. During this time, the spin jets are activated to constraint momentum buildup, and the high-level hydrazine thrusters are substituted for the low-level control jets in order to maintain stronger control. ACS pointing requirements are relaxed during the infrequent vernier-velocity maneuvers.

C16.2.3 Attitude Control Subsystem

The attitude control subsystem (ACS) operates in conjunction with the propulsion subsystem to execute the pointing, payload scanning, and vernier velocity functions of satellite operation. The subsystem uses the power, telemetry, and command provisions of the power and electrical distribution subsystem and the communication and command subsystem. The ACS relies upon the structural subsystem for critical alignment of assemblies; provides attitude determination signals to the payload and sequencing signals to the electrical distribution subsystem; and operates in a test mode in connection with the electrical operating ground equipment. The attitude control subsystem is designed to perform for a period of at least three years on-orbit with a goal of five years.

The attitude control subsystem consists of the following units:

- Sun Sensors
- Earth Sensor Assemblies (ESA)
- Control Electronics Assemblies (CEA)
- Rate Gyro Assembly (RGA)
- Attitude Control Converter (ACC)
- Reaction Wheel Assembly (RWA)
- Reaction Wheel Electronics Assembly (RWEA)

The attitude control subsystem functional block diagram is shown in Figure C16-7.

The ACS consists of non-redundant reaction wheel and rate gyro assemblies, and redundant sun- and earth-sensor assemblies. The assemblies are managed by means of the dual attitude converter. Redundant control electronics assemblies process the sensor data to produce thruster commands in the various control modes. The thruster commands are amplified by redundant valve-drive electronics for application to the thruster solenoids. Satellite spin is effected by the reaction wheel, which has two redundant motors run by one of the redundant reaction wheel electronic assemblies. The capability exists to use one or both motors for reaction wheel run-up.

C16.2.4 Electrical Power and Distribution Subsystem

The primary objective of the power portion of the subsystem is to provide adequate electrical power to all satellite electrical equipment during all operational phases.

The power subsystem is fully self-contained in that it supplies electrical energy from solar array and battery sources, is thermally integrated, monitors its own operation, and provides signals for telemetry. It provides onboard automatic control and command capability for override or backup control. It also provides protection of the subsystem and other selected spacecraft loads essential to mission success.

The configured items comprising the power portion of the subsystem as shown in Figures C16-8 and C16-9 are a solar cell array, three batteries, a power control unit (PCU), five shunt element assemblies (SEA), the solar array transient suppression assembly (SATSA), the array switching unit (ASU), and 12 battery reconditioning discharge resistors and associated circuitry. The same design characteristics for these components are summarized in Table C16-1.

The solar cell array converts incident solar energy during periods of illumination to electrical energy that is used to supply the satellite loads and to recharge the batteries. During periods of launch, eclipse, or peak power demand in excess of solar array capability, the batteries supply all or part of the load, as required. While on the launch pad, power is supplied by the Power and Ordnance Test Set until T-220, at which time the inflight jumpers are installed and the spacecraft is on internal power. The SATSA provides limiting of the solar array bus voltage to 42V and filters high frequency oscillations in the solar array power, shunt, and return lines. The ASU connects (or disconnects) strings of solar cells on the solar paddles and base panels to (or from) the primary bus and return bus.

Since the solar array power capability at the required operating voltage can exceed the demand, dissipative SEAs, which are controlled by the PCU, are used to limit the array voltage. The maximum SEA dissipation occurs under conditions of minimum spacecraft load and battery trickle charging and maximum solar array output.

C16.2.5 Thermal Control Subsystem (TCS)

The spacecraft is exposed sequentially to four periods of differing thermal environments: (1) on-stand, where fairing air conditioning is use to maintain satisfactory component temperatures, (2) launch to satellite separation, (3) from separation to earth acquisition, and (4) normal orbit. The function of the thermal control system is to maintain all spacecraft equipment within acceptable temperature limits throughout the post-launch phases of the mission. This is accomplished primarily by passive means using combinations of thermal hardware, including super insulation, second surface mirrors, and thermal coatings. Thermostatically controlled heaters are employed in a few locations to achieve optimum operating temperatures or to prevent equipment damage or propellant freezing during worst-case cold conditions. The worst-case on-orbit cold condition occurs at equinox (430-Btu/hr/ft^2 solar constant) with a maximum eclipse of 1.2 hours. The worst-case hot condition occurs during winter solstice (445 Btu/hr/ft^2). A description of the thermal environments following launch is continued in the environmental specification, SY1-72.

Table C16-1 Electrical Power Configuration for Phase II Upgrade

Equipment	
o Solar Array	
Number of Panels	20
Number of Cells	90,288
Type of Cells	N-on-P; 2 x 2 cm
Cover Material	0.0006-inch fused silica or ceria doped glass
Area	517 sq. ft
Weight	343 pounds (maximum)
o Batteries	
Number of Batteries	3
Type	Nickel-cadmium
Cells per Battery	22
Capacity (each)	24 amp-hr
Size (each)	850 in.3 (11.9" x 8.7" x 8.2")
Weight (each)	53.0 pounds
Heater Control	45°F (thermal switch)
o Power Control Unit	
Primary Bus Control	Low voltage limit at 31.4 ± 0.2 Vdc High voltage limit at 31.8 ± 0.2 Vdc
Battery Charge Control	Set to temperature limits for tickle or full charge
Load Control	Undervoltage at 24.0 ± 0.5 Vdc
Size	570 in.3 (9.0" x 7.0" x 6.4")
Weight	10.9 pounds
o Shunt Element Assembly	
Number of Units	6
Maximum Total Shunt Power	432 Watts
Area (each)	1.0 ft2
Weight (each)	4.2 pounds
o Solar Array Transient Suppressor Assembly	
Number of Units	1
Size	12"L x 7"W x 2.25"H
Weight	3.4 pounds
o Array Switching Unit	
Number of Units	1
Size	9.55"L x 7.5"W x 7.5"W x 8.0"H
Weight	5.0 pounds

Almost all spacecraft heat dissipating components are located in either the main equipment compartment or the two RADEC II compartments. During an orbit, the spinning satellite causes the compartments to assume all attitudes with respect to the solar vector. To minimize the thermal effects of such varying levels of solar irradiation, the external surfaces of the compartments are completely covered with multilayer insulation blankets, except for local radiating areas. These areas are covered with second surface mirrors having high reflectivity. By appropriately sizing the radiating areas, satisfactory temperature levels are established for the compartments.

C16.2.6 Communication and Command Subsystem

The communication and command subsystem (CCS) transmits sensor mission data and satellite housekeeping data to the ground stations, receives and processes commands, and provides ranging signals for orbit determination.

C16.2.7 Ordnance Subsystem

Ordnance devices provide the following functions:

Separation of the satellite from the booster system;
Deployment of the two high-gain antenna assemblies;
Solar paddle retention/release;
Deployment of the four solar paddles;

The separation function uses a separation nut (128381). The two deployment functions are accomplished by pin pullers (117720). Both ordnance devices, i.e., the separation nut and the pin-pullers, are powered by the TRW pyrotechnic cartridge SP7210.

From a functional point of view, the ordnance devices are composed of pyrotechnic-powered cartridges which, when triggered by an appropriate electrical pulse, produce a self-sustaining chemical reaction whose products actuate a mechanical device.

The ordnance devices used in the structure subsystem consist of the separation nuts, the pin pullers, and their associated circuitry. The purpose of the ordnance component is to actuate a separation system to separate the satellite from the booster and a deployment system to deploy two-high gain antenna assemblies and four solar paddles. All of these functions take place in response to ordnance control circuitry.

The TRW ordnance control and firing circuits control the spacecraft post-separation sequence by automatically activating the deployment ordnance devices. As a backup, all ordnance functions may be activated by ground command.

C16.2.8 Separation Nut (Separation System)

This ordnance device is similar in design and function to the C117634 separation nut, which has performed successfully in the DSP, DSCS, P-74, HEAO, and TDRSS programs. It is based on an improved design that incorporates a sealed pressure cavity and provides reduced shock to nearby components.

C16.2.9 Shock Reduction

The shock imparted to the surroundings via the mounting structure of this separation nut is reduced by minimizing direct impact of moving parts on adjacent structures, by compensating motion of internal parts, and by using a relatively low pressure and a small free volume that results in low piston velocities.

C16.2.10 Pin Puller (Deployment System)

This ordnance device has been used on the DSP program on previous launches. The 117720 pin puller has performed successfully on all previous launches of the DSP and the DSCS, P-74, HEAO, and TDRS. It is based on a proven design that incorporates a dual cartridge (SP7210), either of which will activate a piston so as to remove a pin from a clevis or other mechanical fixture, thus releasing an assembly vital to performance of the satellite.

The satellite components that require deployment after satellite separation from the launch vehicle are the four restraining assemblies, four solar paddles, and two high-gain antenna assemblies. The deployment system consists of torsion springs and pin puller assemblies.

C16.3 MISSION SCENARIO

Satellites 0005R and 0006R are contract items (CIs) consisting of the SED sensor segment, which detects IR sources to accomplish Mission A; the RADEC segment, which accomplishes Mission B, and the spacecraft segment which provides an orbiting platform for the sensor segments. Advanced RADEC I is assembled with the SED sensor by AESC to form the SED Sensor CI. RADEC II is assembled with the spacecraft by TRW to form the spacecraft CI.

Satellites 0005R and 0006R are built to be launched on either the T34D/IUS or the T34D/Transtage. Satellites 0005R and 0006R are Phase II satellites retrofitted with some multi-orbit-satellite/build-to-print (MOS/PIM) features and modifications resulting from new requirements.

For further details see Transtage (TS) Upper Stage Vehicle Mission Scenario.

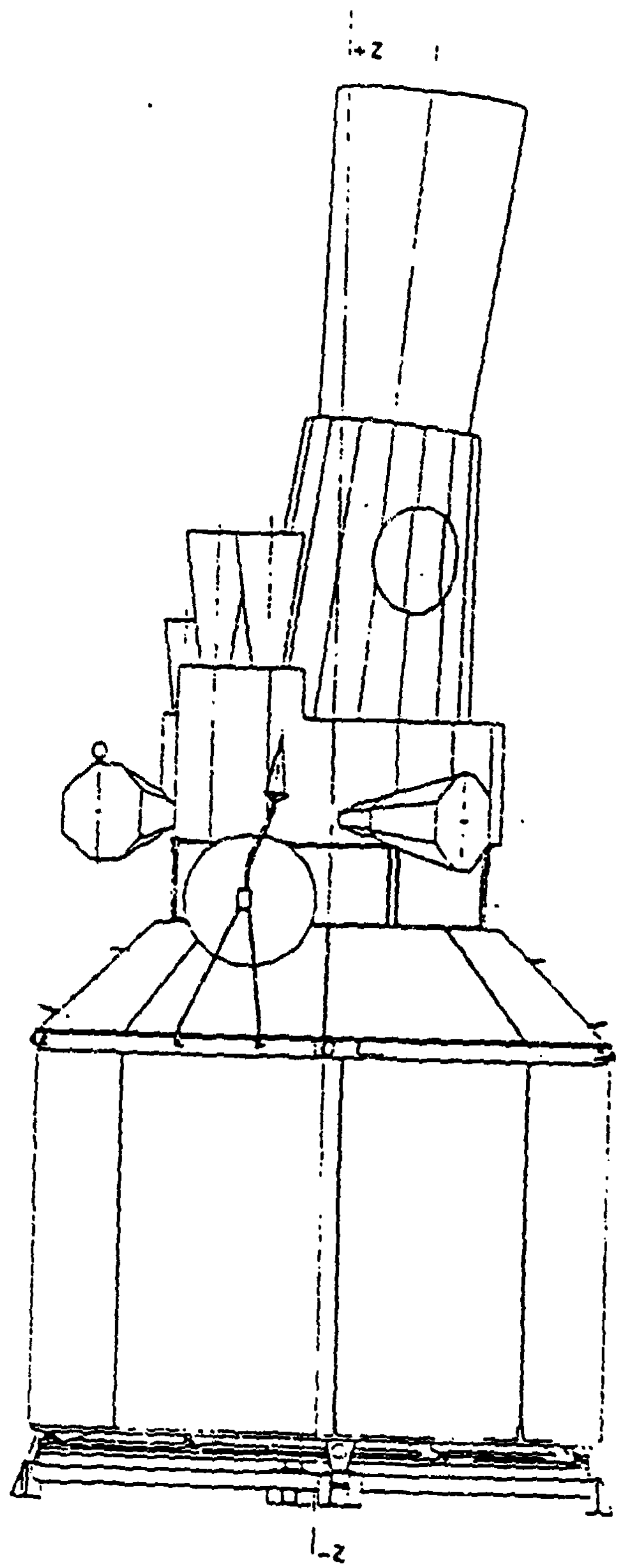

Figure C16-1 Satellites 5R/6R Typical

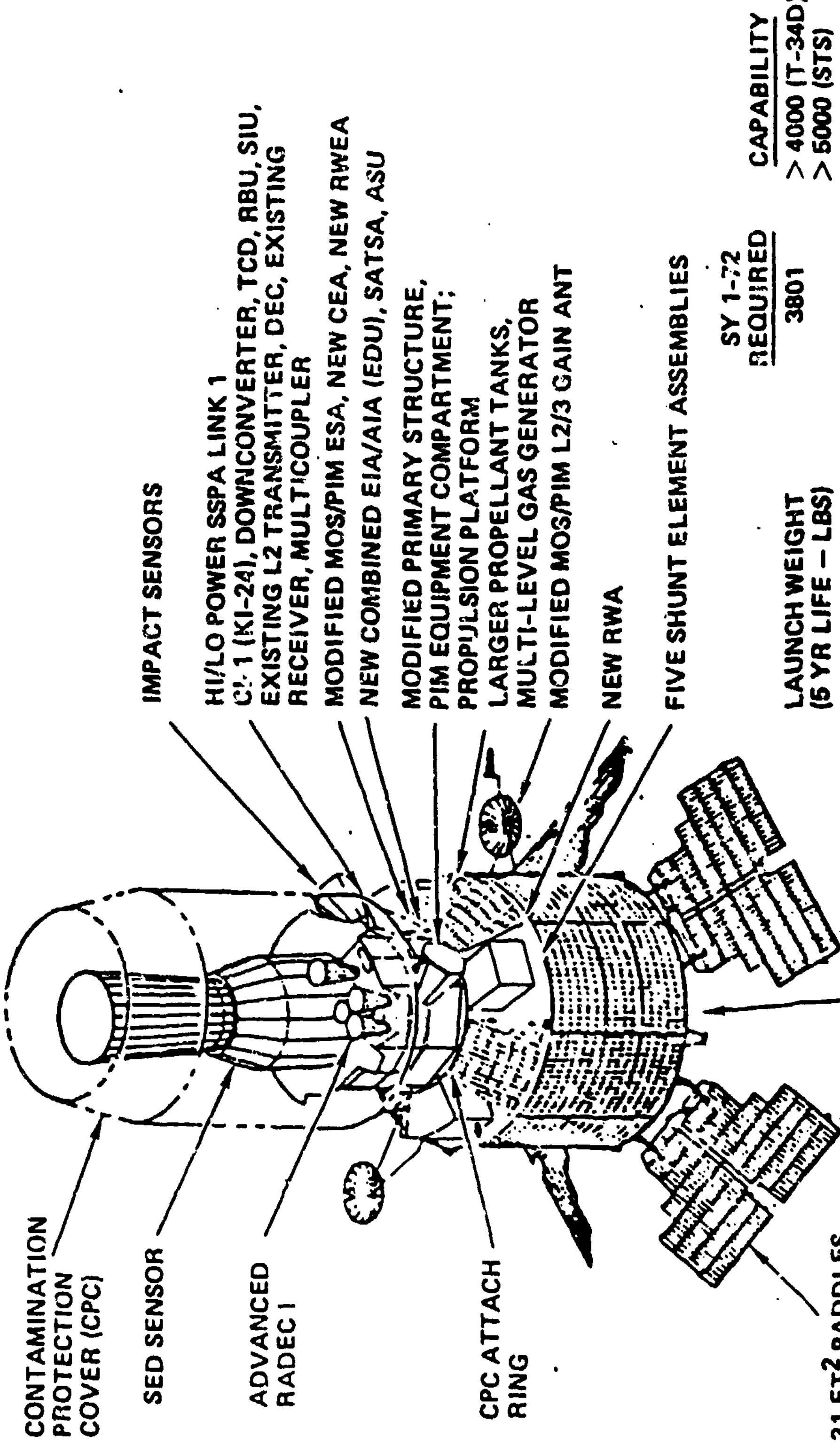

Figure C16-2 Satellites 005 and 006 Retrofit Configuration

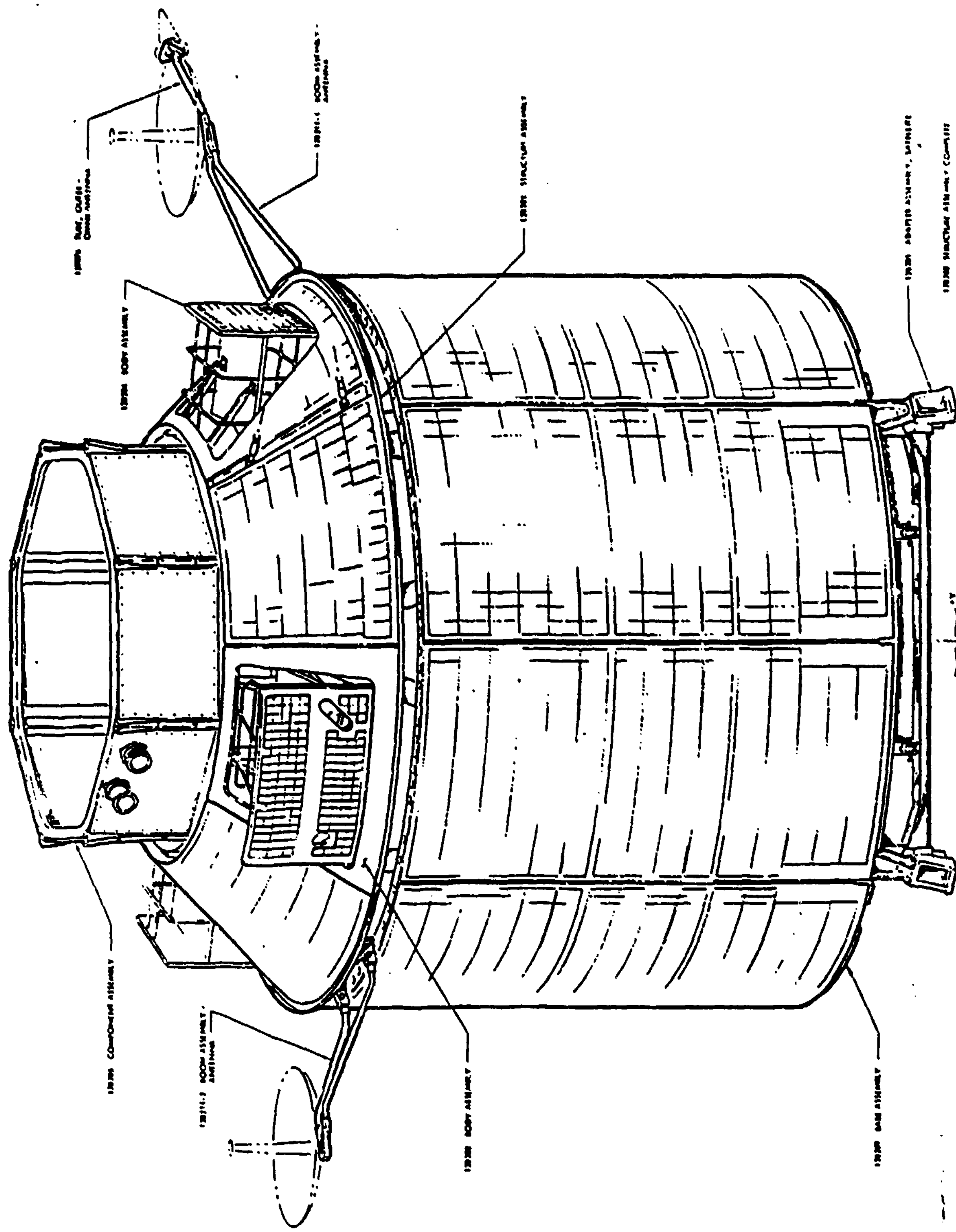

Figure C16-3 Spacecraft Structure Assembly

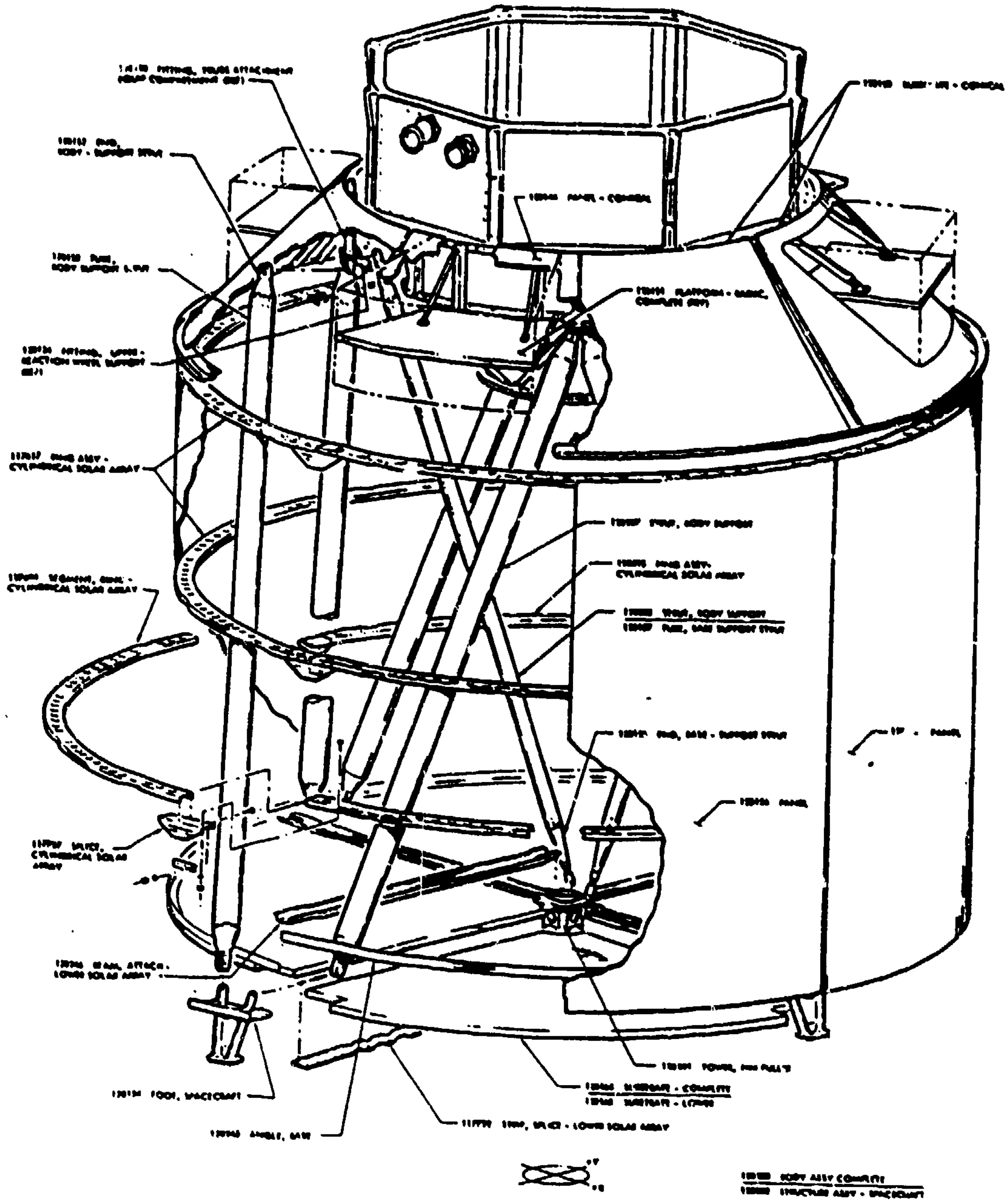

Figure C16-4 Structure Assembly

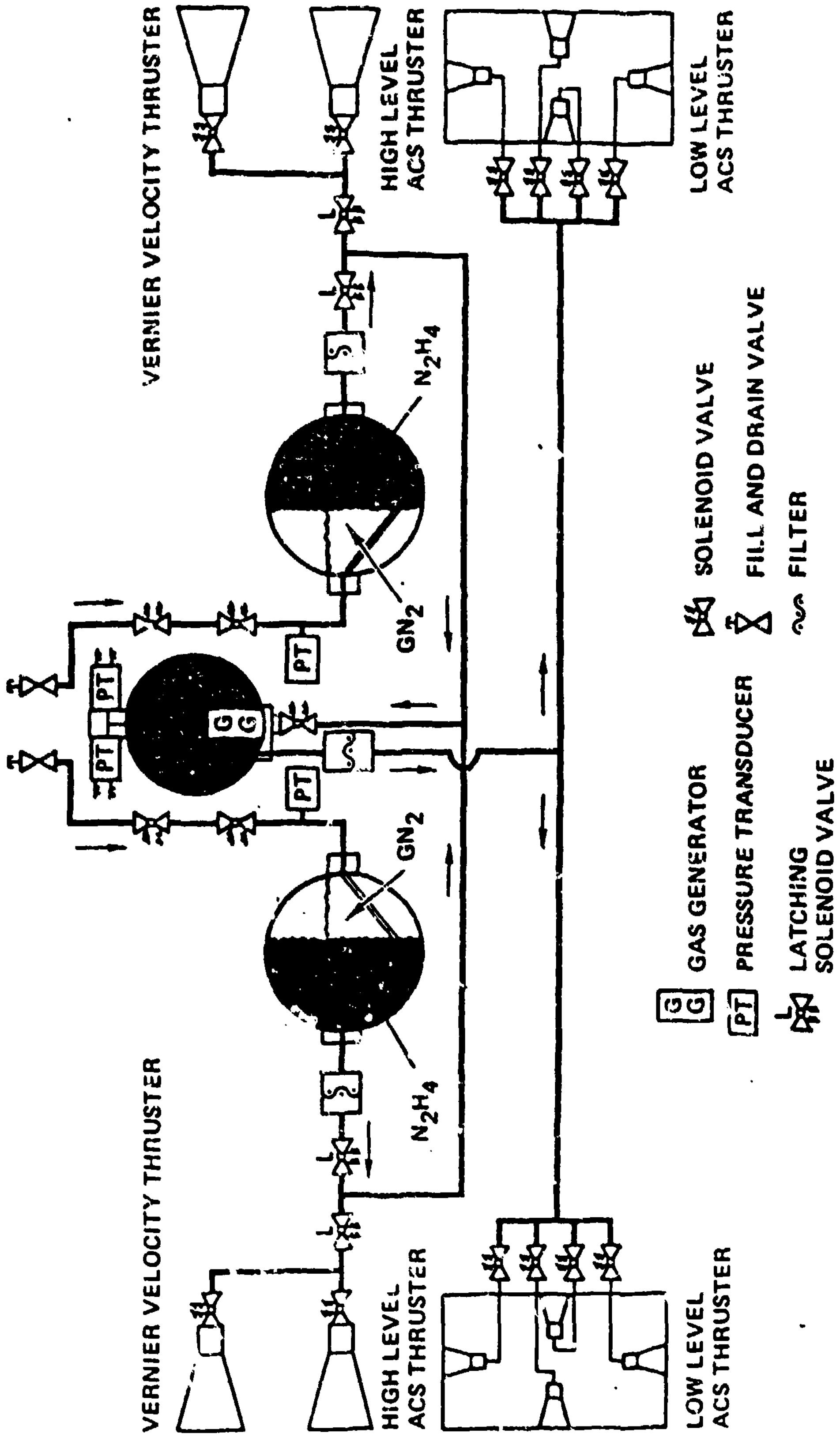

Figure C16-5 Propulsion Subsystem

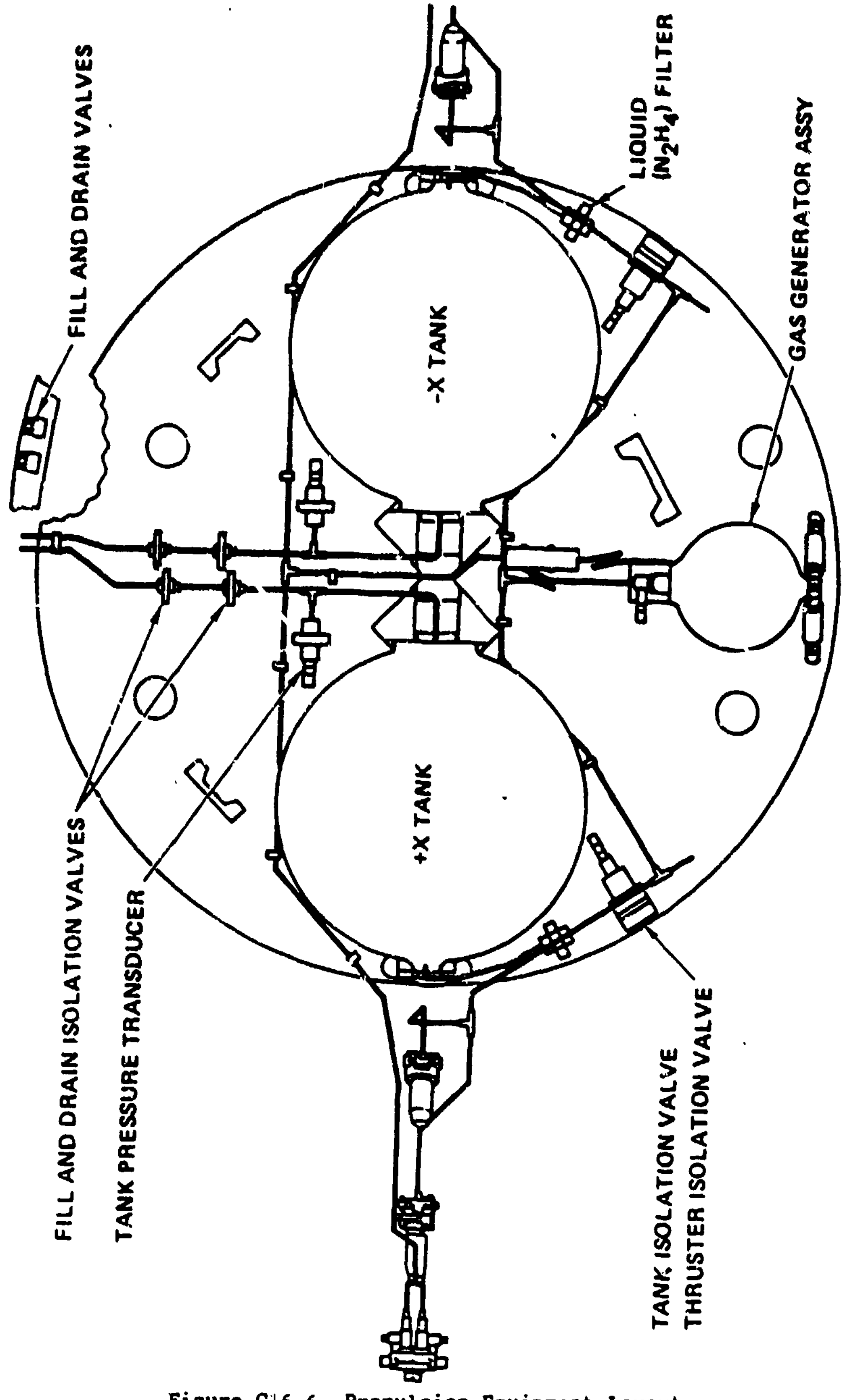

Figure C16-6 Propulsion Equipment Layout

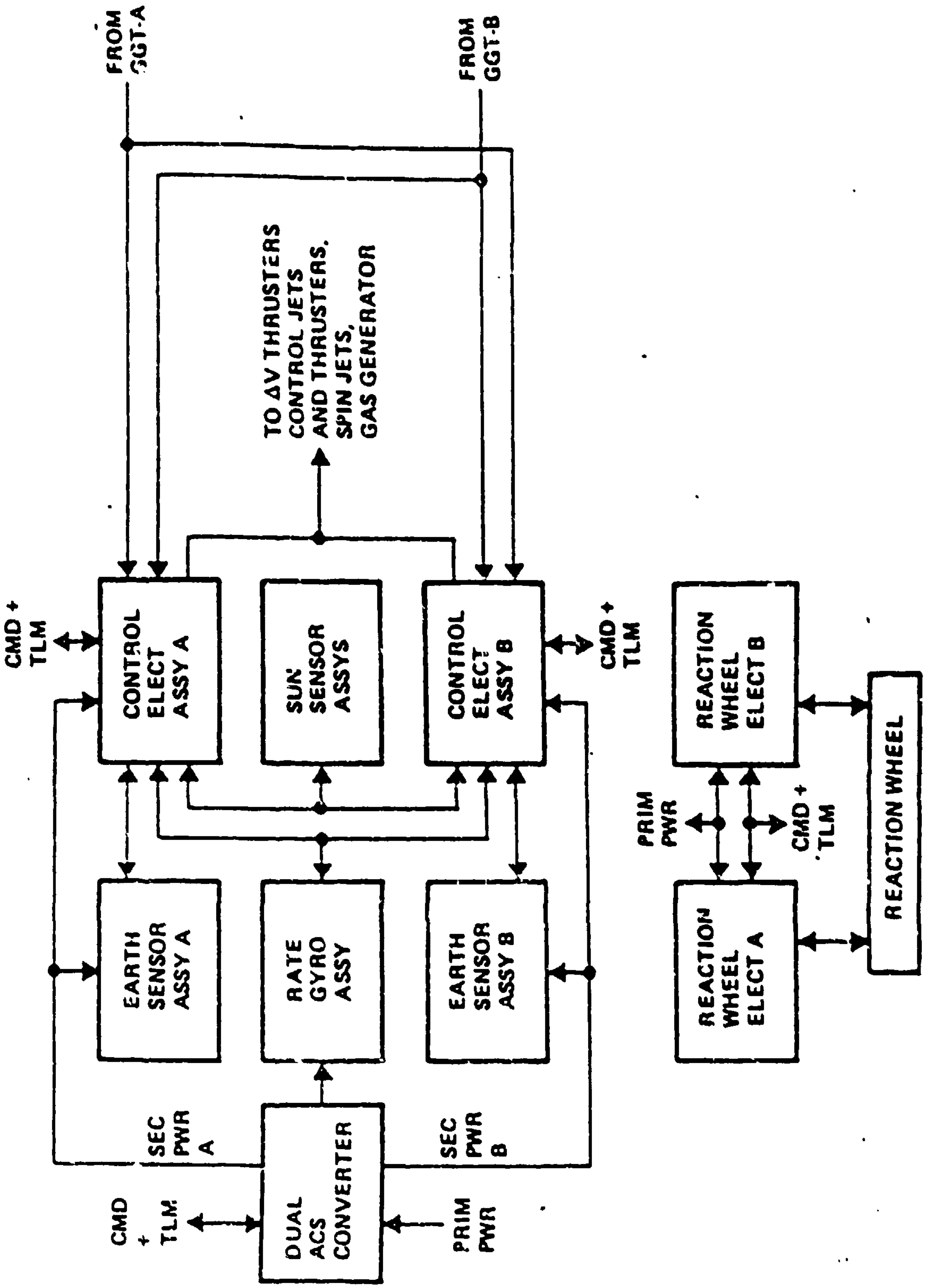

Figure C16-7 Attitude Control Subsystem (ACS)
Functional Block Diagram

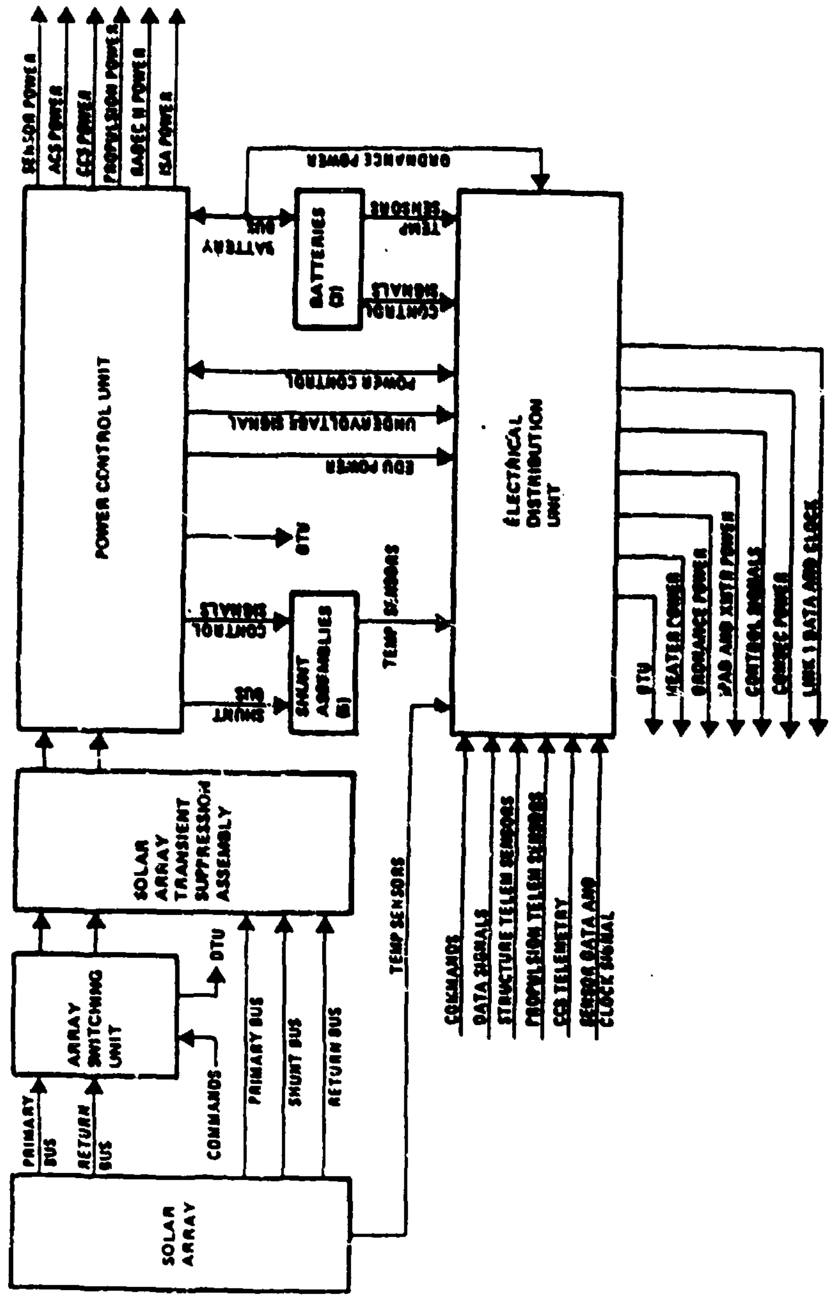

Figure C16-8 Electrical Power and Distribution Subsystem Block Diagram

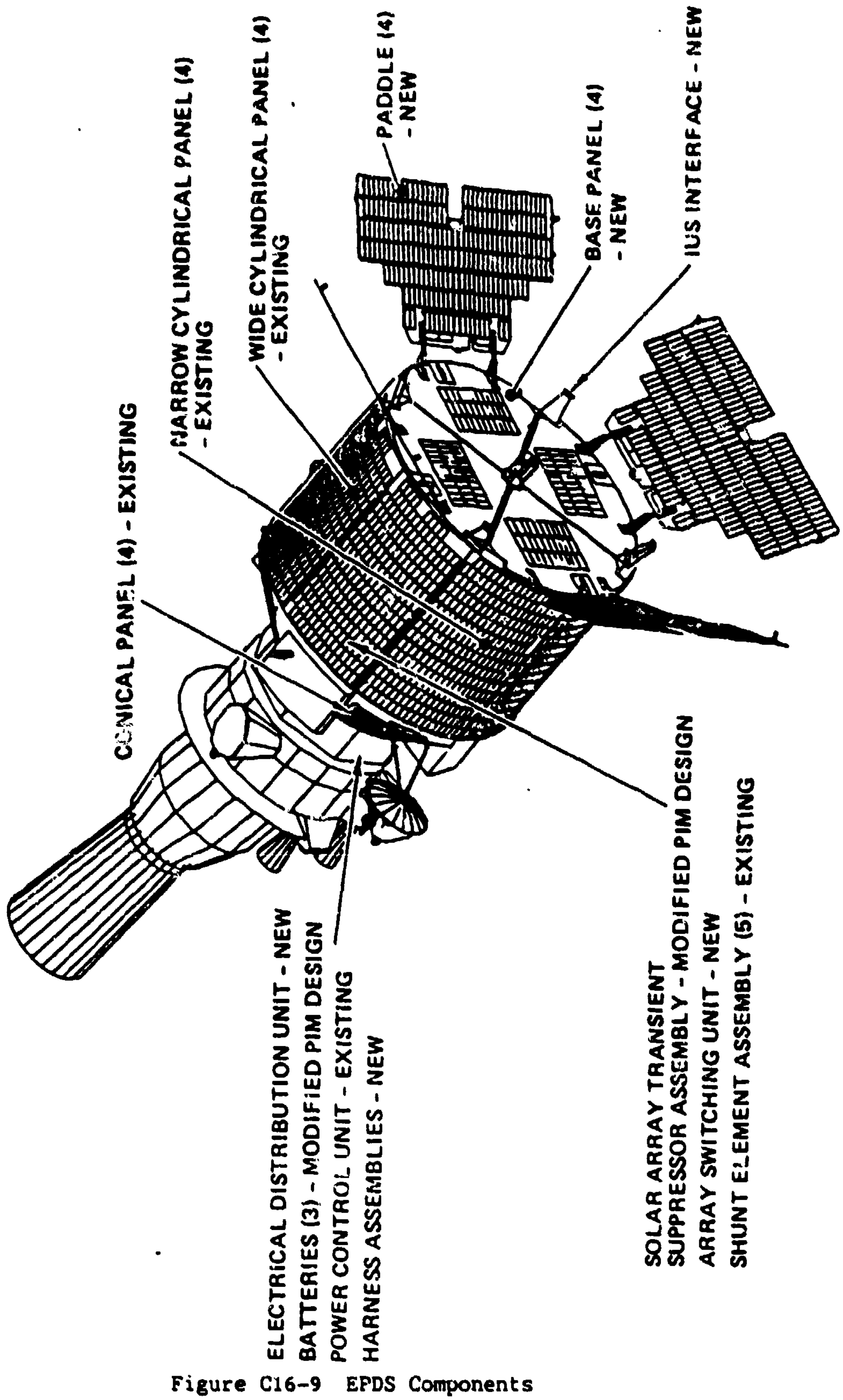

Figure C16-9 EPDS Components

Appendix C17
Global Positioning Satellite

APPENDIX C17
GLOBAL POSITIONING SYSTEM (GPS)

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APPENDIX C17
GLOBAL POSITIONING SYSTEM (GPS)

C17.0 INTRODUCTION

The following data were extracted from the "SSD 83-D152-1 Phase 2 Accident Risk Assessment Report for NAVSTAR Block II Production Satellite GPS-0013 and PAM-D II Upper Stage Cargo Element," Rockwell International, Satellite Systems Division, approved by Eyman and Inokuchi, September 1984.

C17.1 GENERAL DESCRIPTION

The Global Positioning System (GPS) provides navigational signals to the navigation user community for the purpose of determining position, velocity, and time. It also provides an orbiting platform and support for IONDS Global Segment (IGS) payload operations.

The Space Segment (SS) consists of the aggregate hardware, software, analyses, materials, services, and data required to design, develop, produce, and test a satellite constellation supported by the Control Segment (CS) and providing global navigation signals to the GPS User Segment (US) and signal data to the IGS user subsegment. The SS configuration includes the Space Vehicle (SV), Support Equipment (SE), and Flight Support Computer Programs (FSCP). The SS functional areas are identified in Figure C17-1. Two types of space vehicles may be on orbit in the GPS constellation: Block I Navigation Development Satellites (NDS) and Block II Satellites. This document describes the Block II Satellite only, since Block I satellites are launched from expendable boosters.

The space vehicle consists of two payloads and supporting subsystems.

The spacecraft (configuration shown in Figures C17-2, C17-3, and C17-4) is composed of the following subsystems:

a) Structures Subsystem
b) Thermal Control Subsystem (TCS)
c) Electrical Power Subsystem (EPS)
d) Attitude & Velocity Control Subsystem (AVCS)
e) Reaction Control Subsystem (RCS)
f) Orbital Insertion Subsystem (OIS)
g) Telemetry, Tracking and Command Subsystem (TT&C)
h) L-Band Subsystem (L-Band)
i) Integrated Transfer Subsystem (ITS)

The payloads to be flown on and supported by the GPS spacecraft are:

a) Navigation Payload (NAV)
b) Global Burst Detector (GBD)

C17.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C17.2.1 Telemetry, Tracking, and Command (TT&C) Subsystem

This subsystem provides telemetry, tracking, command, and navigation upload capabilities to support vehicle operations in conjunction with the Air Force Satellite Control Facility (AFSCF) and the Control Segment.

The TT&C S-Band antennas consist of three [illegible]nna elements connected to redundant transponders via the RF assembl[illegible] [illegible]conical spiral/bicone assembly is mounted on an articulated deployable mast that extends from the forward bulkhead. Another conical spiral element is mounted on an articulated deployable mast that extends from the aft bulkhead. Both antennas are functionally operable in their stowed positions with reduced pattern coverage. No space vehicle RF transmissions through these antennas will occur during Shuttle countdown, ascent, and descent. Prior to SV/PAM deployment, the TT&C transmitter will be enabled by Orbiter GPC commanding, via the PAM ASE, of the appropriate load control unit (LCU) relay.

C17.2.2 L-Band Subsystem (LBS)

The L-Band subsystem will generate, amplify, and transmit radio signals at the L_1, L_2, and L_3 carrier frequencies. The subsystem consists of three functional assemblies: (1) the navigation RF assembly (NFRA), (2) the L_3 transmitter assembly, and (3) the antenna assembly. This subsystem is not activated during Orbiter ascent and descent. Inadvertent operation is controlled by three independent inhibits, as are the TT&C and ITS subsystems.

C17.2.3 Integrated Transfer Subsystem (ITS)

The integrated transfer subsystem provides for the radio transfer of data between its equipped GPS Satellites. A single UHF antenna is used for the transmitting and receiving functions. Since simultaneous transmit and receive operations are not required, the antenna is switched between transmitter output and receiver input by transmit/receiver switch in the RF and Digital Unit (RFDU). The antenna is integrated into the navigation antenna and comprises a ring array of eight dipole elements and feed network to provide a RHCP split-beam antenna pattern. The subsystem never operates while the CE is in the Orbiter payload bay; therefore, it does not pose a radiation hazard to STS operations.

C17.2.4 Ionizing Radiation Data

This section provides ionizing radiation (radioactive) source data for identified reportable sources associated with operations to be conducted at ESMC and/or KSC.

Phase 0/1/2 The Radiological Safety Analysis Summary and Radiation Protection Plan for the Rubidium Frequency Standard CDRL 049A2 Contract F04701-78-C-0153 is provided as radioactive source data for Rubidium-87.

The cesium frequency standard (CFS) uses Cesium 133, which is a natural and stable (nonradioactive) source and does not emit radiation or thermal energy. In accordance with AFR 122-16, Nuclear Safety Review Procedures for Space of Missile Use of Radioactive Sources, A Radiological Safety Analysis Summary and Radiation Protection Plan are not required for Cesium 133.

The Department of the Air Force-Headquarters Air Force Systems Command Aerospace Safety Division has granted Nuclear Safety Approval for NAVSTAR Block II Satellites with Rubidium-87 physics packages to be placed in Earth orbit aboard the Space Transportation System.

Requirements for radioactive sources to verify compliance with Eastern Launch Site (ELS) data requirements will be provided at a later date. The necessary forms required by the Radiation Protection Handbook, KHB 1860.1, and by ESMC Regulation 160-1 will be provided during later submittals. JSC Form 44, ionizing radiation source data sheet for space flight hardware, will be prepared by the NAVSTAR Joint Program Office AFSC-SD/YE.

Each NAVSTAR Block II Satellite contains two Rubidium Frequency Standards (See Fig. C17-5) manufactured by Collins Communication Systems Division (CCSD), Rockwell International Corp, using a radioactive physics package supplied by Efratom Systems Corporation. This physics package includes two glass cells which contain approximately 600 micrograms of Rubidium metal vapor enriched in the isotope Rb-87. The lamp cell and its resonance cavity cell contain approximately 300 micrograms each. The thickness of glass used in the lamp cell and resonance cavity cell is 0.12 $\pm$ 0.03 cm. This is greater than the range of the beta particles emitted by Rubidium-87 and will not therefore give rise to a beta radiation dose outside the cell. The gamma ray dose at the face of the physics package is less than 10^{-3} millirem per hour.

The worst-case potential hazard that could occur would be the release of Rubidium-87 from both sets of the lamp and resonance cavity cells. The amount of Rubidium-87 released would be less than a milligram (0.12 nanocuries).

The composition, probable chemical form, and fraction of permissible body burden of the total activity that may be released is summarized below:

Radioactive Material	Probable Chemical Form	Total Activity	% Maximum Permissible Burden for Total Body
Rubidium-87	Soluble	0.12	1.5×10^{-4}

The maximum body burden is defined as that amount of an internally deposited radioisotope which imparts the maximum permissible dose, 5000 millirem per year. The added dose due to human ingestion of the total available Rubidium-87 activity would be:

$$\frac{8736 \text{ hrs/yr} \times 0.03 \text{ mr/hr}}{5000 \text{ mr/yr}} \times 100 = 5.24\%$$

In the event of a worst-case potential hazard, the rubidium would probably be released to the environment as a respirable oxide, and inhalation would be the route of ingestion with long term deposition in the lung. This could lead to proportionally higher exposure to lung tissues (maximum retention = 12%) and decreased exposure to other body tissues. Rubidium uptake mechanism in the body closely resembles that of potassium with a slightly slower transport. The skeletal and heart muscles form the chief store-house, followed by liver and gall bladder. Nonradioactive Rubidium is not known to be toxic to humans. Muscle tissue may accumulate Rubidium up to concentrations of 20-60 percent higher than those of potassium with very little adverse effects on the body. Rubidium is present in nearly all iron ores, in some mineral springs, and is fairly abundant in food elements, especially beans, barley, and soft tissue of most animals.

There is no conceivable event in which one person could receive an intake of Rubidium-87 which is a significant fraction of the annual limits if all the lamp or resonance cavity cells are broken and the released material is inhaled-ingested. The internal radiation dose equivalent from intake of Rubidium-87 as a consequence of damage to cells in the payload will be negligible.

C17.2.5 Radiological Safety Data

The lamp cell and resonance cavity cell of the Rubidium Frequency Standard contains a Group 1 silvery-white, soft, alkaline Earth Metal: rubidium, in which the radioactive Isotope Rubidium-87 has been enriched to about 60%. The Rubidium-87 decays to stable Strontium-87 with the emission of a beta particle. The following data summarize the pertinent parameters for the radioisotope contained in the vehicle:

Radioisotope	Emitted Radiation Decay Mode	Half Life (yrs)	Maximum Permissible Burden in Total Body* (u ci)	Maximum Allowable Occupational Concentrations** (u ci/ml)	
				In Air	In Water
Rubidium-87	B-	$4.7x10^{10}$	200	$5x10^{7}$	$3x10^{-3}$

*Permissible Dose for Internal Radiation; International Commission on Radiological Protection

Beta particles can penetrate the epidermis of the skin and irradiate the radiation sensitive dermis layer of skin. The following shielding data, based on U.S. Public Health criteria and on information by manufacturer of GPS Rubidium Frequency Standard should be considered:

Radioisotope	Maximum Energy	Equivalent Shielding for Complete Absorption of Beta Particle	Half-Value Layer Thickness
Rubidium-87	0.274 MeV	0.01 cm, Pb	0.002 cm*
		0.05 cm, Glass**	--------
		0.1 cm, Tissue**	--------
		0.1 cm, Al*	--------
		80 cm, Air**	--------

*U.S. Public Health criteria (Radiological Health Handbook, 1970 revised edition); minimum thickness of commercially available sheet lead is 1/64 inches (0.40 mm.).
**Information furnished by Efratom Systems Corporation.

Beta radiation sources that are outside the body lead to exposure of the skin. If taken into the body through inhalation or ingestion, they can lead to exposure of the whole body. Dose equivalent limits for occupational radiation exposure are:

Organ	Annual Dose Equivalent Limit* (REMS)
Whole Body	5
Hands and forearms; feet and ankles	75
Skin of whole body	30

* CFR 20;

C17.2.6 Special Range Safety Considerations

The Rubidium Frequency Standards contain an extremely small quantity of radioactivit in the physics package and do not require dosimeters to be worn by pers anel. The level of radioactivity at the face of the Rubidium Frequency Standard is essentially undetectable by portable dosimeters. Wipe test . a is not required, because no removable contamination would be present on the sealed physics package. The probability of leakage/release of radioactive material is remote.

The minor radioactive source contained in the physics package does not present an external radiation hazard because of the limited quantity of Category C Minor sources available. In the event of an uncontrolled release of the isotope, a very minor local contamination and airborne radioactivity might exist if the release occurred in a relatively small volume. There is no significant radiobiological hazard associated with a release of the material on-pad or above the Earth. No unique or special handling procedures of the Rubidium-87 minor radioactive source/frequency standard is required.

A launch pad fire involving the liquid and solid fuel would create temperatures near 5000°F, which would vaporize the sources. The Rubidium-87 will mix with the very large vapor cloud and move downwind. The wide distribution of 120 picocuries within this vapor cloud will not create a significant radiological health hazard. In fact, if all 120 picocuries were to be inhaled by one individual, it would be much less than the maximum permissible body burden of 200 micocuries set by the International Commission on Radiological Protection.

C17.2.7 Description of Minor Radioactive Sources

Rubidium-87 is incorporated into vehicle design as a source of ionizing radiation required by the NAVSTAR'S navigation time - frequency standard subsystem.

a. Radionuclide, mode of decay, and associated intensities and energies:

Radioisotope	Decay Mode	Half life (yrs)	Maximum Energy
Rubidium-87	B- (decays to stable Strontium-87)	4.7×10^{10}	0.274 MeV

b. Activity:

Radioisotope	Activity per Source	Loc W/N Freq Std	Total No. Sources	Total Radioactivity
Rubidium-87	0.030 Nanocuries	Lamp Cell	2	0.06 Nanocuries
Rubidium-87	0.030 Nanocuries	Resonance Cavity Cell	2	0.06 Nanocuries
			Total Activity	0.12 Nanocuries

c. Manufacturer: Efratom Systems Corporation

d. Source identification number: Efratom Model FRK-HM80 Rubidium-87

e. Cross-Sectional Sketch: See Figure C17-6

f. Source holder and material or construction:

The Lamp Cell source holder is constructed of glass 0.05 ± 0.02 cm thick.

The resonance cavity cell source holder is constructed of glass 0.12 ± 0.03 cm thick.

g. Physical Form: Silvery-White-Colored Metal

h. Chemical Form: Group 1 Alkaline Earth Metal

i. Date, Type, and Result of Last Wipe Test: Not Applicable

j. Method of Sealing To Prevent Leaks: Glass Encapsulated

k. Radioisotope Location in Satellite: See Figure C17-7.

l. Type of Protective Cover Material: Not Applicable

m. Radioisotope Soluble in Water: Yes

C17.2.8 Propulsion Data

Two propulsion subsystems are identified for use on the space vehicle. One is an orbital insertion subsystem (OIS), and the other is a reaction control subsystem (RCS). The OIS will incorporate a solid rocket motor, and the RCS will utilize liquid propellant hydrazine thrusters.

GPS Space Vehicle Ordnance System Data

Manufacturer: Thiokol Corp, Elkton, Maryland
Item: Solid-propellant rocket motor - STAR 37XF
Part No: TE-M-714-8
NEW: 1795 lbs
Chemical Composition: TP-H-3340 propellant (89% HTPB)
DOD Class: 1.3C
DOT Class: Rocket Motor, Class B Explosive
RF Susceptibility: TBD

The STAR 37XF motor is located on the centerline of the Z axis within the thrust cylinder and cone of the space vehicle structural body assembly. The rocket motor provides the impulse at the transfer orbit apogee to circularize the orbit, simultaneously providing a plane change of 25.05 degrees. The rocket motor is being developed as an apogee motor for the Intelsat V spacecraft and incorporates an 89 percent solids HTPB propellant in a high-strength titanium case and a submerged nozzle with integral torodial igniter and carbon/carbon exit cone. The only electrical power required by the rocket motor is for thermal control heaters. (See Fig. C17-8).

Autoignition sensitivity of the propellant is time and temperature dependent. Instantaneous ignition will occur at temperatures above 500°F. Autoignition will not occur when exposed to temperatures of up to 250°F for up to eight hours.

Manufacturer: Thiokol Corp, Elkton, Maryland
Item: Igniters - S&A/ETA/TBI/Initiator/Toroidal Pyrogen
Part No: Model 2134A (S&A)
NEW: 1.5 grams (S&A Detonators Only)
Bridgewire resistance: 0.90 to 1.10 ohms
Max. Safe-No-Fire: I amp/1 watt for 5 minutes
Minimum-Fire: 5.0 amp
All-Fire: 3.5 amps 20 milliseconds
RF Susceptibility: TBD
DCT Class: Nozzle & Toroidal Igniter - <u>Igniter, Rocket Motor, Class B Explosive</u>
Pyrogen Igniter TP-H-3174 Propellant - <u>Propellant Explosive (solid), Class B</u>
DOD Class: Nozzle & Toroidal Igniter - <u>1.3C</u>
Pyrogen Igniter TP-H-3174 Propellant - <u>1.3C</u>

The safe and arm (S&A) device and motor ignition train are designed to preclude any inadvertent ignition of the rocket motor propellant grain caused by static electricity, human error, or lightning while the GPS space vehicle is in the orbiter cargo bay. The S&A will be located on the forward face of the aft space-vehicle bulkhead. The S&A contains all fragmentation from the igniter ignition and will not cause damage to adjacent components. (See Fig. C17-9).

The S&A rotor assembly is powered by a 28-Vdc motor with an integral planetary gear speed reduction unit. The rotational power of the dc motor is transmitted to the output shaft through spur gears and a slip clutch. If the motor is activated with the safety pin in place, the slip clutch will prevent motor damage. The explosive rotor assembly, visual indicator, and rotary switches are located on the output shaft. The rotary switches control the electrical circuitry, including motor control, remote monitoring, detonator circuit shorting, and firing signals. In the safe position, the rotor assembly provides a physical barrier in the flow path between the detonators and the explosive transfer assemblies (ETA). When arming current is applied and the safety pin has been removed, the output shaft rotates 90 degrees to align the rotor lead assembly with the explosive train flow path. The S&A is remotely armed and safed electrically. (See Fig. C17-10)

C17.2.9 <u>Reaction Control Subsystem</u>

The reaction control subsystem (RCS) shall be provided to effect the required spacecraft maneuvers during the SV/PAM-D transfer orbit operations, SV drift orbit operations, and SV on-orbit operations. The RCS shall be a blowdown system using monopropellant hydrazine pressurized by gaseous nitrogen. The subsystem shall consist of a propellant/pressurant storage (PPS) assembly, a propellant distribution and control (PDC) assembly, and the rocket engine assembly module (REAM). The PPS components consist of two propellant tanks, each equipped with positive expulsion diaphragm, two propellant fill and drain valves, two pressurant fill valves, and two temperature transducers. The PDC components consist of two pressure transducers, two filters, and two latching isolation valves. The REAM components consist of twenty 0.44N (0.1 lbf) thrusters and two 22.2 N (5 lbf) thrusters, each equipped with a temperature sensor, thruster valve heaters, and catalyst bed heaters. (Fig. C17-11.)

C17.2.10 Interface Description

The interfaces of the cargo element are the GPS/PAM-DII interfaces and interfaces with the GBD and STS Orbiter considering the present mission planned for the GPS. The cargo element is comprised of:

PAM-DII (cradle, restraint mechanism, payload attach fitting, spintable and expendable vehicle)

Global Positioning Satellite

Global Burst Detector

C17.2.11 Interface Control Drawings

The interface controls specify the flight vehicle interface agreements among the contractors and agencies involved in the integrated program. The effects of the design and operation of the airborne elements and ground-based elements of the GPS/PAM-DII systems are contained in the following documents:

Shuttle Orbiter/Payload Assist Module-DII (PAM-DII) Interface Control Document (ICD) IDC-A-18413, dated 15 May 1984

Navstar GPS Space Vehicle/PAM-DII Interfaces, MH08-00004-600, dated 15 May 1984

GPS/Global Burst Detector (GBD) Interfaces (Block II), dated 22 August 1982

These documents serve as the instruments for contractual control of the Cargo Element interfaces.

C17.2.12 Interface Configuration

PAM-DII avionics is shown in Figure C17-12. Figure C17-13 depicts the cargo element. It is the GPS, with GBD installed, mounted on the PAM-DII. Figure C17-14 shows the integrated system with relation to the thermal protection by the sunshield.

The interfaces of primary concern consist of a physical/mechanical attachment at all interfaces and the associated loads therein; an electrical/avionics interface concerned with ordnance systems, separation systems, electrical power and command discretes; a thermal interface; and EMC interfaces. Of secondary concern are interfaces where ionising radiation, toxic materials, and flammable materials may be involved as well as considerations for possible operator error.

C17.2.13 Global Burst Detector

The global burst detector (GBD) consists of an optical sensor (BDY), a data processing unit (BDP) and either an x-ray sensor (BDX) or a dosimeter (BDD). The system operates onboard a global positioning system (GPS). In the configuration where the BDP, BDY, and BDX are connected, the GBD detects nuclear events. In the configuration where the BDP, BDY, and BDD are connected, the GBD detects background radiation and nuclear events.

The sensor outputs are processed, and the data is delivered to the IONDS global segment (IGS) portion of the satellite by the BDP. The BDP also accepts GPS timing signals from the satellite for time-tagging the event detection and formats the vent data for transmission by way of the GPS L3 data link. In addition, 27-volt power, timing signals, discrete commands, and other signals are received from the satellite. The GBD state-of-health (SOH) information is input to the satellite PCM unit for transmission on an S-Band telemetry link. Figure C17-15 defines the interface between the GPS and the GBD.

C17.2.14 Ignition of Flammable Materials

The GBD does not contain batteries or other energy sources which could ignite flammable materials. In addition, the GBD is turned off at all times while in the payload bay. Two faults would be required to turn the GBD on and provide an ignition source.

C17.2.15 Ignition of Flammable Atmospheres

There are no mechanical switching devices in the GBD that would provide sparks or arcs. The system is off while in the payload bay.

C17.2.16 Radiation, Ionizing/Non-Ionizing

There is no source of ionizing or non-ionizing radiation in the GBD.

1. There are no transmitters in the GBD equipment.

2. There is no radioactive material in the GBD equipment.

C17.2.17 Venting of Sealed or Unsealed Components

Adequate venting is provided in the GBD electronic equipment enclosures. All enclosures are contained within the GPS main structure.

C17.2.18 Hazardous Voltage Sources

All sources of hazardous voltages are of extremely low current. Positive barriers are provided for all circuits in the GBD, and, in addition:

1. The GBD is in the off condition during prelaunch, launch, ascent, descent, and abort. A 27-volt power source is provided by the GPS spacecraft through the load control unit. The following actions are required by the ground system to turn on the GBD:

   LCU power to GBD must be turned on;
   Magnitude command must be sent to the GBD (GND CMD);
   NDU Power must be commanded on (GND CMD);
   FSDU Power must be commanded on (GND CMD);
   Frequency standard must be commanded on (GND CMD);

2. The 27-volt GPS input to the GBD is fused on the spacecraft side with two medium-speed 2-amp fuses in parallel. This interface limits the input current to a maximum of 4 amps. If the GBD were commanded on while in the STS payload bay and a short existed in the GBD, the maximum current would be carried by the #26 and #28 wire. This current would be insufficient to ignite the insulation or to raise temperatures to 352°F.

C17.2.19 Toxic Materials

There are no toxic materials in the GBD.

C17.2.20 Ordnance

There is no ordnance in the GBD.

C17.3 MISSION SCENARIO

The GPS space vehicle has two missions. The first mission is to enhance user performance by providing a worldwide signal environment which will give three-dimensional position and velocity and time information to suitably equipped users. The GPS accomplishes this mission through the use of three segments, i.e., space segment, control segment, and user segment. The Block II space vehicle's navigation payload and L-Band subsystem provide the user segment with pseudorandom noise (PRN) navigation signals on the primary (L_1) and secondary (L_2) frequency bands.

The second mission is the nuclear burst mission. A nuclear detonation at or above the Earth's surface is detected by the GBD mounted on the Block II space vehicle. These data are processed in the burst detector processor, chipped, encoded, then bi-phase modulated at 200 bps with the navigation P(Y)-code and transmitted on the L_3 carrier frequency of 1381.05 MHz to IGS user terminals. The IGS user terminal integrates the detected burst data from four or more satellites to determine the Earth location of the nuclear detonation.

C17.3.1 GPS Production Space Vehicle Processing

The SV will arrive at the CCAFS skid strip aboard a 747 cargo aircraft without the orbit insertion motor installed. The SV will then be offloaded from the aircraft to a low-bed air-ride trailer and transported to the NPF airlock where it will be unpacked, the shipping bag cleaned, and the SV inspected for damage. If damage is detected, discrepancy reports will be initiated and dispositioned before processing the SV. The SV will then be moved into the SV checkout bay and connected to the GSE; air conditioner cooling ducts will be installed; and batteries will be charged in preparation for functional testing. The SV will then undergo a postfactory functional test, solar array illumination test, AFSCF compatibility test, control segment test, pin puller test and connection for flight, EED installation and verification, and OIS ordnance and safe and arm device functional test. The SV batteries are reconditioned and left discharged until delivery to the launch pad.

The Navstar Launch Base Processing Flow is shown in Figure C17-16.

C17.3.2 GPS Mission Phases

Fig. C17-17 defines the mission phases required to perform the Shuttle-launched GPS mission. The beginning and ending events for each mission phase and the associated time bases are defined in Figure C17-18. The time base is used from time zero and shall be the means for establishing the time performance of each event or sequence within a given mission phase.

C17.3.3 Preflight Operation Phase

This phase begins when the SV arrives at the Eastern Space and Missile Center (ESMC), initiating Time Base 1. This time base is a countdown keyed with the Shuttle launch countdown operations. The SV and PAM are either supplied from their respective production phases or are returned from the Orbiter landing site after an aborted flight. The SV will undergo checkout, servicing, and integrated checkout with the PAM cradle at ESMC. The SV/PAM/ASE/Orbiter interface will be verified, containerized, and transported to the Shuttle launch pad, where it will be installed into the Orbiter payload bay. SV status checks will be made prior to payload bay door closure.

C17.3.4 Shuttle Ascent Phase

This phase begins with Orbiter payload bay door closure and initiates Time Base 2. This time base is composed of two time segments: The first is coincident with the Shuttle launch countdown and is terminated at solid rocket booster (SRB) ignition; the second segment commences at SRB ignition and is terminated when parking orbit insertion is achieved, resulting in orbital maneuvering subsystem (OMS) engine cutoff. Propellant for the main engines is loaded into the external tank (ET) after the payload bay doors are closed, and the Shuttle vehicle is sequenced to launch status. First-stage ascent is accomplished under thrust from both SRBs and the three main engines. Second-stage ascent occurs under main engine power. The ET is separated after the delta-V conditions have been realized and after the main engines shut down. The OMS engines then provide thrust to insert the orbiter into the parking orbit. Normal phase termination occurs at OMS engine cutoff. An aborted flight can be initiated any time after SRB separation and before OMS engine cutoff at parking orbit insertion. Return to launch site (RTLS) is the first abort mode and will result in an Orbiter landing at KSC. Abort once around (AOA) is the second abort mode and will result in an Orbiter landing at KSC. Abort to orbit (ATO) is the third abort mode and may place the Orbiter in a lower orbit than the desired parking orbit. Orbiter landing could occur at KSC, Edwards AFB, or Vandenberg AFB.

C17.3.5 SV/PAM/Orbiter Parking Orbit Operations Phase

This phase begins with the Orbiter OMS engine cutoff and initiates Time Base 3. The Orbiter reconfigures for orbital operations by opening the payload bay doors, initiating radiator cooling, and increasing electrical power and cooling capability to onboard payloads as may be required. SV/PAM status check and preparation operations will be conducted to determine flight readiness using the STS communication network and tracking and data relay satellite system (TDRSS) in conjunction with onboard ASE control functions. The Orbiter will then cooperatively maneuver to the attitude required for SV/PAM/Orbiter separation and SV/PAM PKM ignition. The PAM spin table will then spin the SV/PAM to 75 $\pm$ 3.75 rpm. The SV/PAM separation is initiated from the aft flight deck of the Orbiter to fire a pyrotechnic separation device on the PAM cradle. The SV/PAM is ejected from the PAM cradle under a spining load to terminate the phase. An aborted flight can be initiated prior to SV/PAM Orbiter separation. Orbiter landing could occur at KSC, Edward AFB, or Vandenberg AFB.

C17.3.6 SV/PAM Transfer Orbit Operations Phase

This phase begins with SV/PAM Orbiter separation and initiates Time Base 4. The SV/PAM is spin-stabilized about its PKM thrust axis upon leaving the Orbiter and is oriented in the attitude required for the transfer orbit injection burn. The orbiter will perform a series of maneuvers in accordance with the NASA PAM separation sequence to obtain a safe separation distance. The PAM PKM will be fired when the programmed orbital point has been achieved and will thrust until burnout to inject the SV/PAM into an elliptical transfer orbit having an apogee altitude close to the operational orbit altitude. SV PAM separation will be commanded via the PAM sequencer following the PAM PKM burnout. The SV will continue its flight in the spin-stabilized mode. The SV will undergo an inversion procession maneuver to generally align the OIS motor axis with the expected drift orbit insertion thrust vector. The SV-attitude and transfer-orbit state vectors will be determined by the AFSCF and trim maneuvers performed as necessary to accurately align the OIS motor axis with the required orbital insertion thrust vector. The AFSCF will preform state-of-health (SOH) checks and maintenance as required.

C17.3.7 SV Drift Orbit Insertion Phase

This phase begins near apogee when the SV OIS motor is commanded to fire and initiates Time Base 5. The OIS motor thrusts to burnout to insert the SV into the initial drift orbit. The AFSCF will monitor the orbit insertion delta-V for satisfactory performance.

C17.3.8 SV Drift Orbit Operations Phase

This phase begins when the SV OIS burn is complete or when the decision is made to make an on-orbit storage vehicle operational and initiates Time Base 6. The SV, under control of the AFSCF, shall be despun about its Z axis, in increments, to a rate of approximately 25 rpm, (not applicable for on-orbit storage vehicles). If the angle of sun incidence on the stowed solar array panels does not provide adequate power and thermal control, the SV will then be precessed toward a more favorable attitude. A drift orbit correction maneuver (or maneuvers) may be necessary to correct for any PAM burn errors so that the final drift orbit will coincide with the desired drift orbit. A "horizontal" precession maneuver will be performed, as required, prior to any drift orbit correction maneuvers to modify SV attitude for correct delta-V fire alignment. The delta-V is performed with continuous firing of one five-pound thruster. The SV will then be processed, if required, such that the +Z axis will be nadir pointing at the planned time for Earth acquisition. Following this nadir maneuver, the SV will be despun, in multiple steps, to a final rate of approximately 1.2 rpm. The pitch and roll thrusters will be enabled and controlled by the combined Earth sensor (CES) control from spin-stabilized to two-axis-stabilized and thus effect "Earth acquisition." The -Y solar array wing will be deployed, and the SV rotation about the yaw axis will be stopped under control of the solar array boom-mounted yaw sun sensor. This will result in the vehicle being 3-axis stabilized, with the -Y solar array wing continuously tracking the sun under the control of its pitch sun sensors. Electrical power will be applied to the navigation payload and L-band subsystem heaters and a portion of the NAV payload equipment to thermally stabilize certain circuits. The +Y solar array wing will then be deployed and continuous tracking of the sun commenced. The L_1 and L_2 portion of the L-band subsystem and the remainder of the NAC payload will be powered up. An initialized NAV message shall be commenced on the L_1 and L_2 RF downlinks. Jet controls will be deactivated, leaving autonomous attitude control of the SV to reaction wheels and commandable magnets. The AFSC Camp Parks Facility may be used to evaluate the L_1 and L_2 RF parameters for comparison with the S-band NAV telemetry received at the STC. Following this series of events, the AFSCF will relinquish control of the SV to the Control Segment.

C17.3.9 SV Operational Phase

This phase begins with the transfer of SV control from the AFSCF to the control segment and initiates Time Base 7. Once the CS has received the SV orbital parameters from the AFSCF, the monitor stations will acquire the SV L_1 and L_2 downlinks for transmission to the CS. The CS will use the data received from the MSs to continue NAV performance evaluations and make any necessary NAV adjustments. The S-band SGLS shall be used to uplink commands and data. The CS will preform the necessary station acquisition delta-V maneuvers to place the SV into the final, on-station, operational orbit and will continue all orbit maintenance operations. The NAV payload will then be uploaded, placed into operation, and the GBD will be made operational. The GBD will undergo a less-than-one-week period of sensor trigger level adjustment and a characterization of sensor response. The expected L_3 transmission time for these operations will be less than 10 minutes per day. Normal GBD operating time will be approximately five minutes per day following the calibration period. The SV will be able to support GBD operation for a maximum of 2 hours per day. The L_3 link will be monitored by the IGS control subsystem co-located with the GPS control segment. The CS will check the SV health and status and perform all SV on-orbit maintenance operations. The AFSCF may be required to periodically perform and/or monitor S-band SOH supports on each SV to maintain the proper level of SV knowledge and operational proficiency necessary to perform contingency operations should the CS require assistance in the support of an ailing SV. The AFSCF will retain the backup capability to reset and upload NAV messages to the SV.

C17.3.10 Orbiter Descent Phase

This phase begins when the abort or deorbit decision is made and initiates Time Base 8. The Orbiter reconfigures for deorbit, when parking orbit operations are terminated, by decreasing electrical power and cooling capability to onboard payloads, initiating flash evaporator cooling, and closing the payload bay doors. The orbiter performs the deorbit delta-V and then orients to the atmospheric entry attitude. The atmospheric entry, terminal area energy management, and approach and landing subphases are used to control the orbiter vehicle on an entry profile to reduce the entry velocity and land at the designated landing site. This phase is terminated when the orbiter vehicle landing rollout is complete.

C17.3.11 SV/PAM/Orbiter Post-Landing Operations Phase

This phase begins when the orbiter vehicle landing rollout is complete and initiates Time Base 9. The landing site for nominal and abort flights will be KSC, or Vandenberg AFB, or Edwards AFB for flight safety and logistical reasons. Post-landing operations will be performed in accordance with an operational timeline. Orbiter and SV/PAM post-landing secure operations will be performed as necessary. The Orbiter vehicle will be placed on external electrical power and cooling shortly after landing and towed to a closed area. The payload bay doors will be opened using ground support equipment. The SV and PAM will be safed, then slings will be attached to the PAM cradle. The PAM ASE or the SV/PAM ASE will then be hoisted out of the payload bay and inerted into a protective shipping canister. The loaded canister will then be transported to either Seal Beach or the ESMC facility for processing for the next mission.

C17.3.12 On-Orbit Storage Operations Phase

Certain space vehicles may be placed into on-orbit storage in each of the six operational orbit planes to foreshorten the response time for replacing a vehicle that has failed. These vehicles may remain in orbital storage for as long as 2.5 years before their operational activation. On-orbit storage vehicles will be spin stabilized, their solar arrays will remain in the stowed position, and they will be monitored and maintained on a periodic basis.

Subphase operations for this phase commence after the SV OIS burn is complete. Despin to 25 rpm, attitude and orbit determination, and horizontal maneuver operations are identical with those defined for the SV drift orbit operations phase.

Vehicles selected for placement as on-orbit spares will be identical to operational vehicles. The mission operations for a spare SV will be identical with the normal deployment up to the time of drift orbit insertion. The initial drift orbit of the SV will not be corrected by DOC maneuvers until the SV is brought out of storage and is assigned to a specific location within its orbital plane.

Following the AFSCF checkout, the SV control will be transferred to the Cs, which will maintain the vehicle until activation to operational status is desired. During this maintenance period the SV AVCS will be deactivated, and attitude control will be maintained by spin stabilization and passive mutation damping. SV configuration during the storage period is identical to the launch configuration except that the TT&C S-band telemetry system is enabled for the automatic turn on/off (ATO) mode of operation.

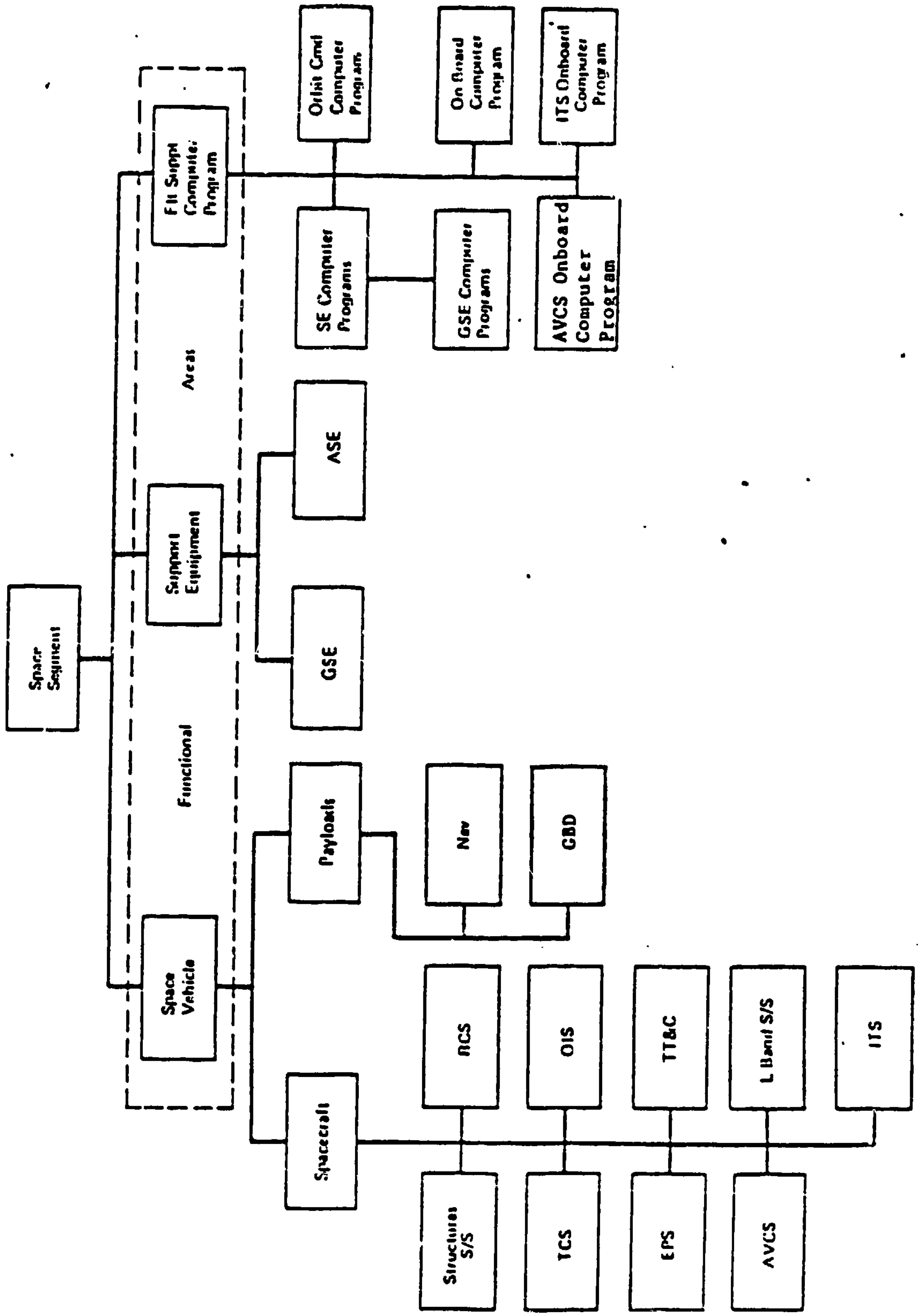


Figure C17-1 GPS Space Segment Functional Areas

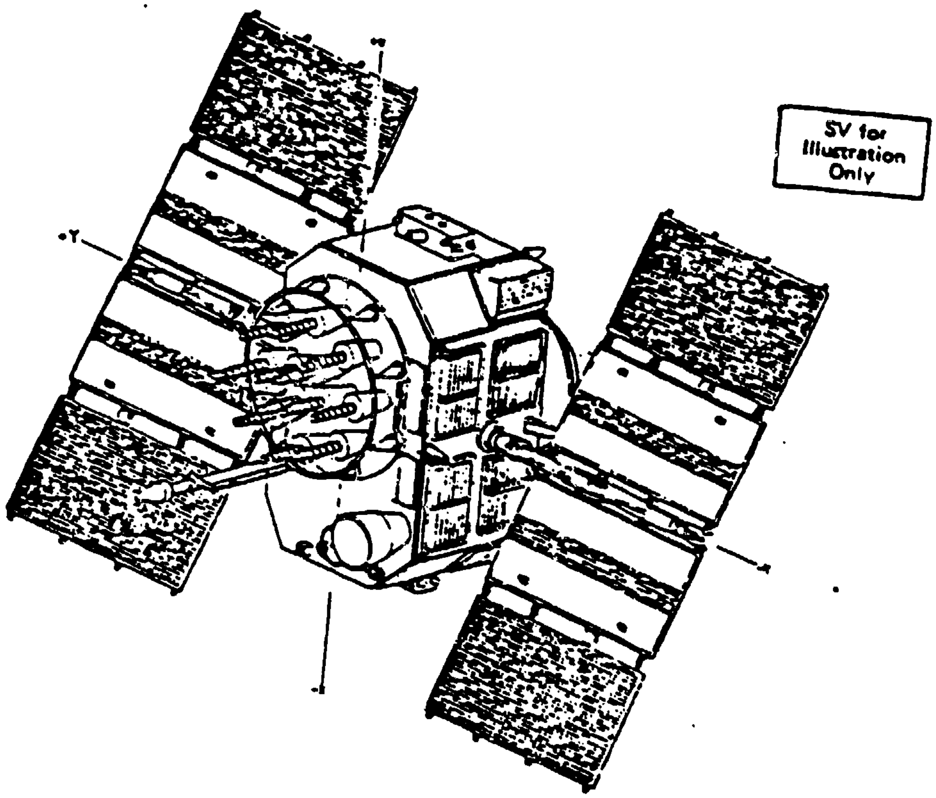


Figure C17-2 Typical Block II Space Vehicle Design

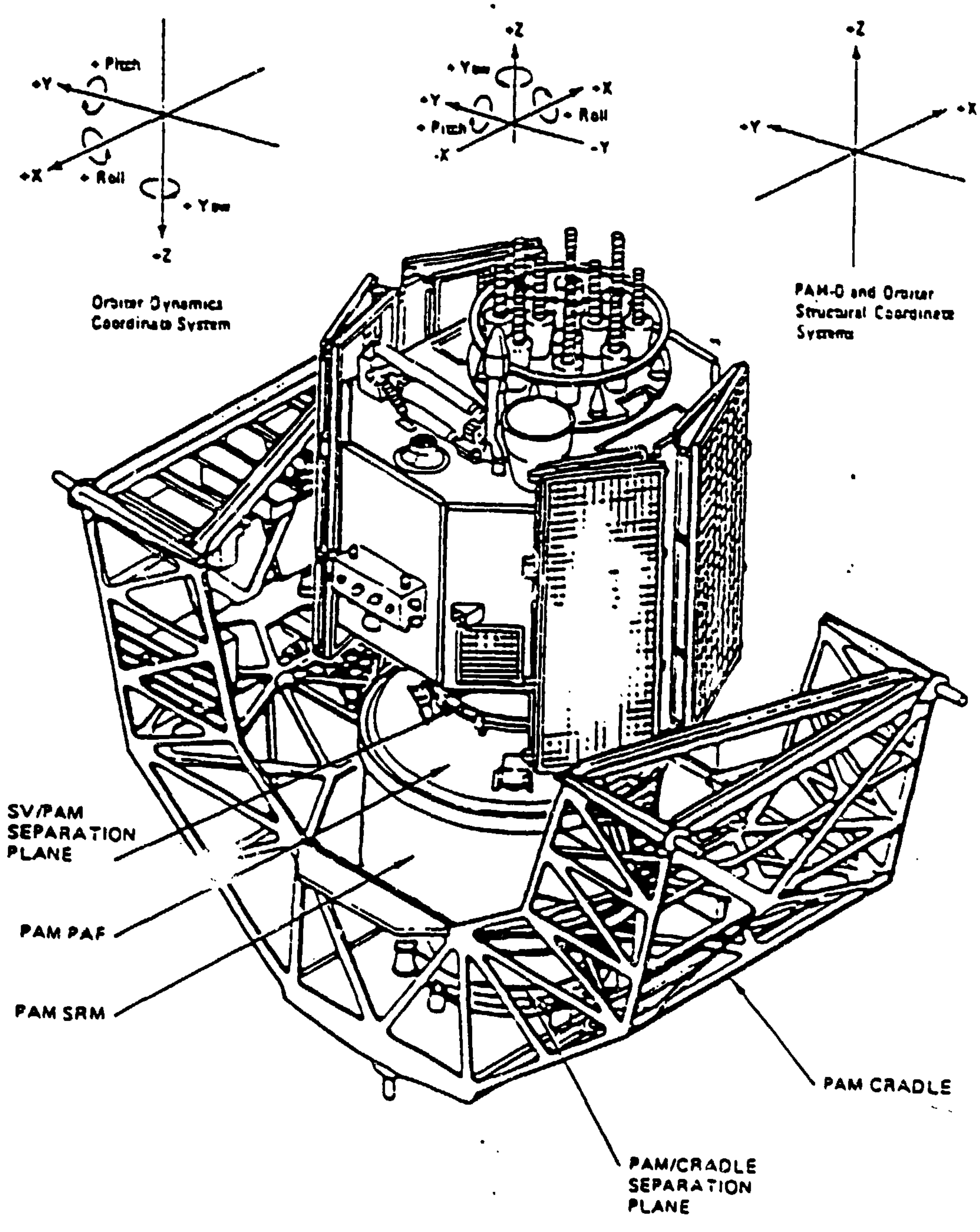


Figure C17-3 , PAM, and PAM Cradle

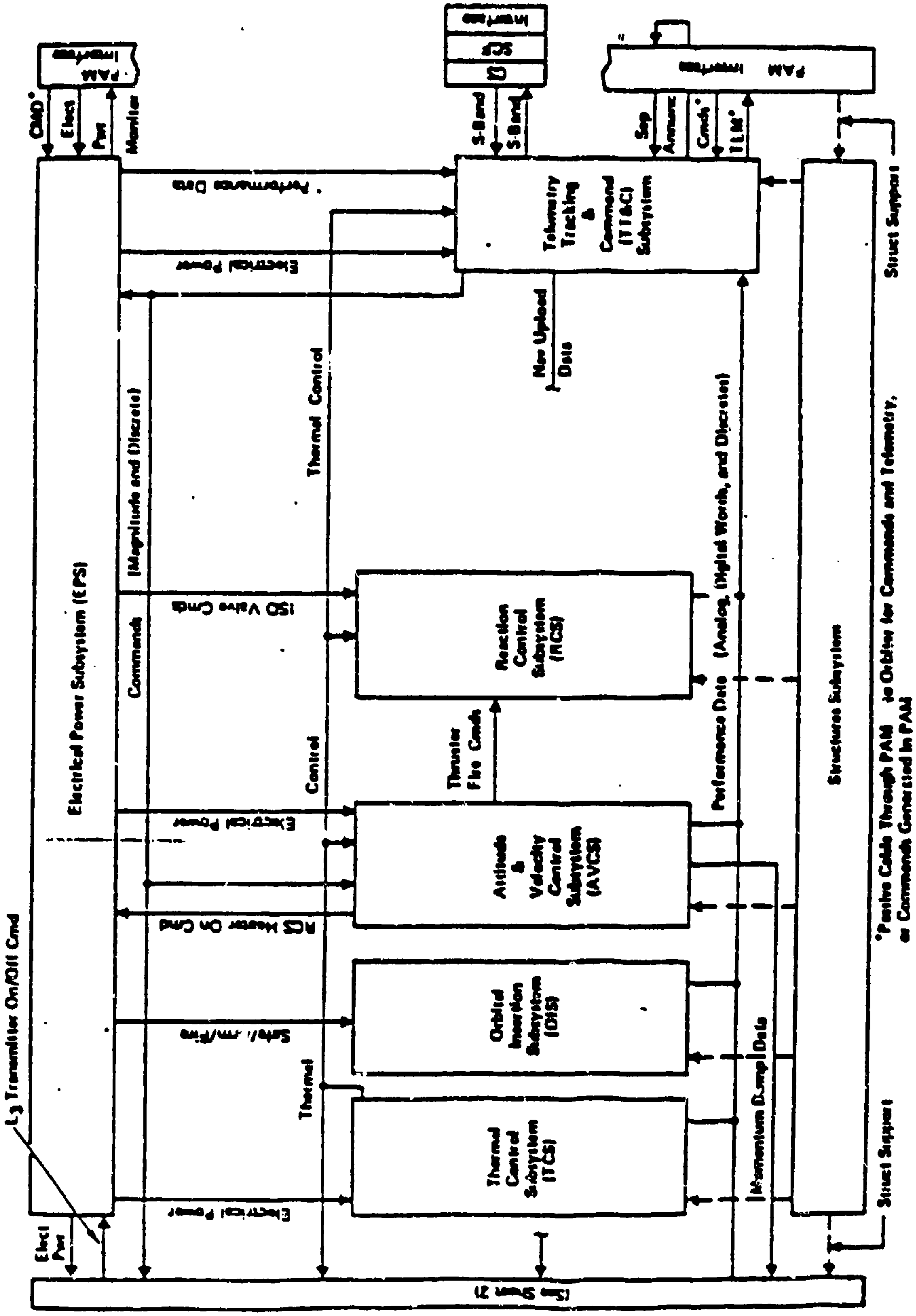


Figure C17-4 Shuttle-Launched Navstar GPS Space Segment Vehicle Block Diagram (Sheet 1 of 2)

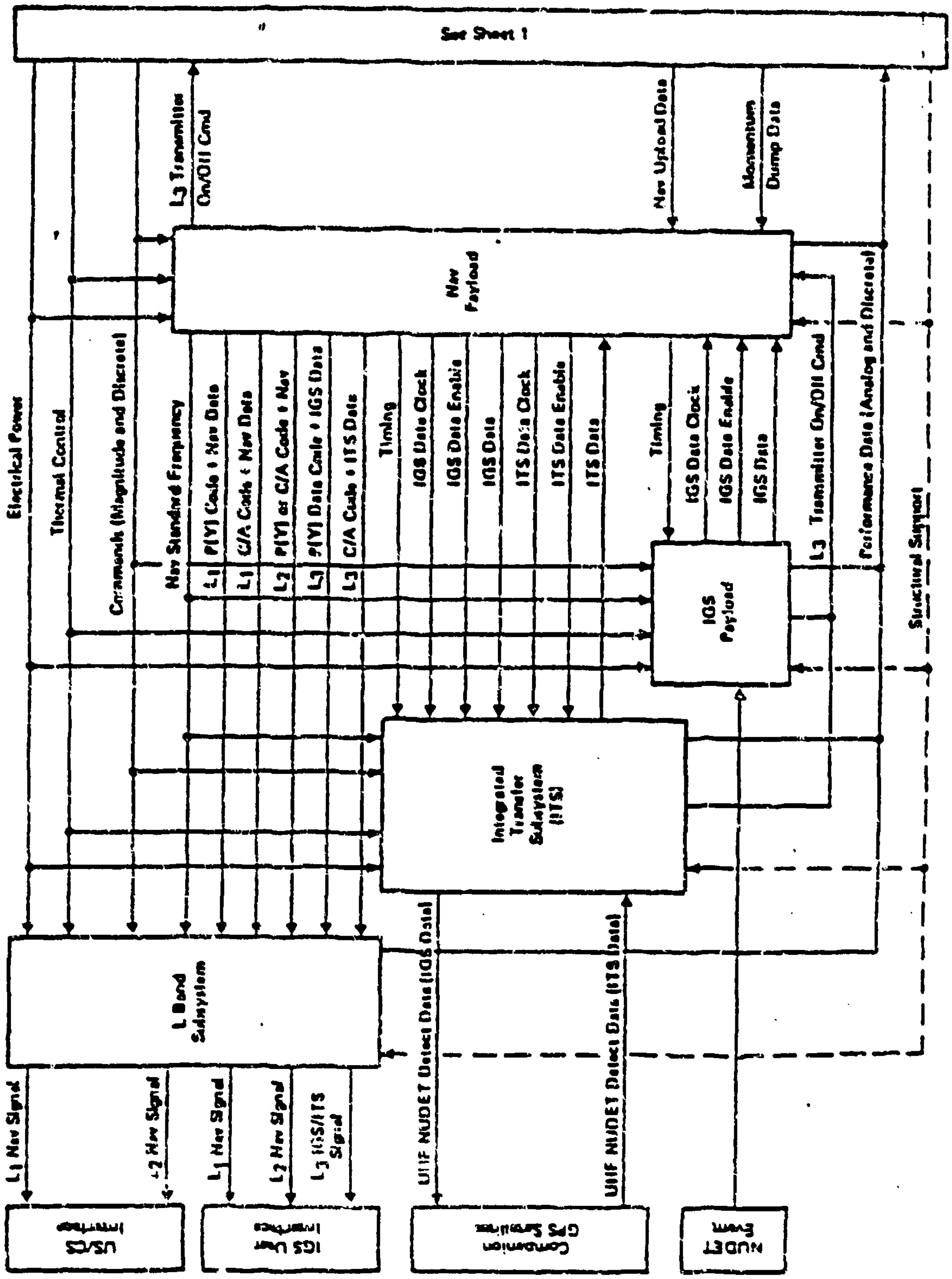


Figure C17-4 Shuttle-Launched Navstar GPS Space Segment Vehicle Block Diagram (Sheet 2 of 2)

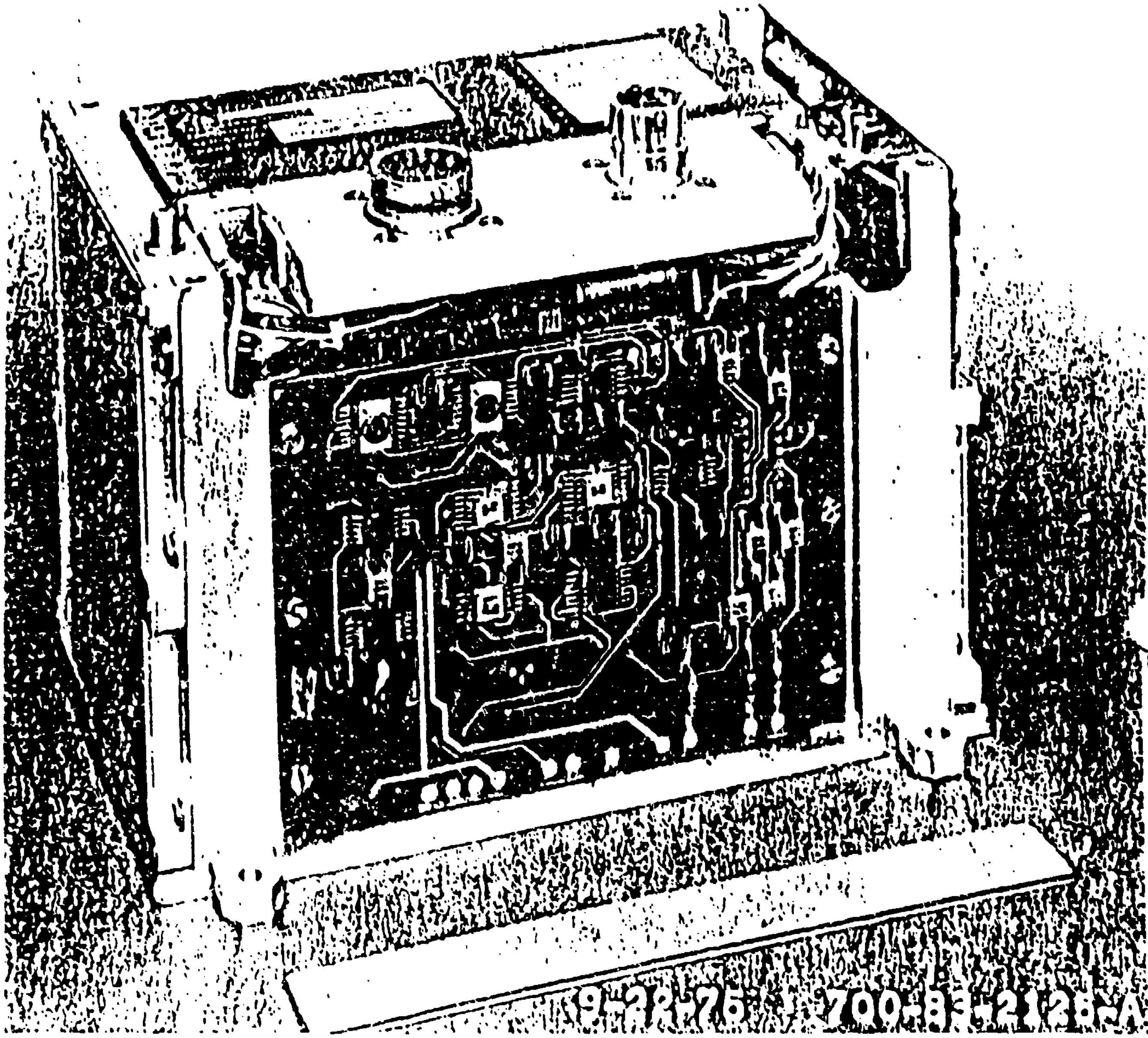


Figure C17-5 NAVSTAR BLOCK II Satellite Rubidium Frequency Standards

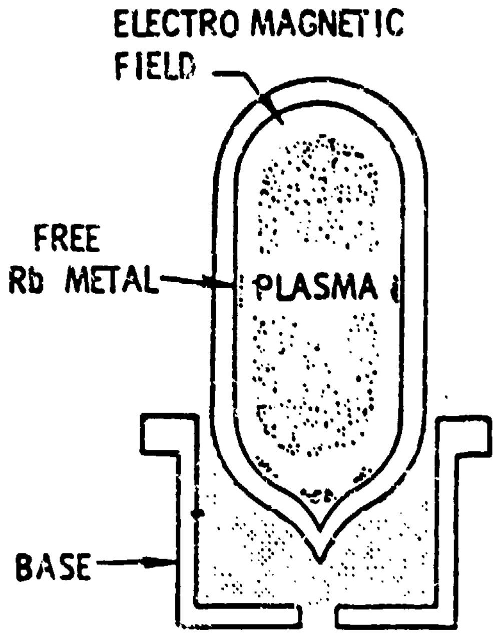
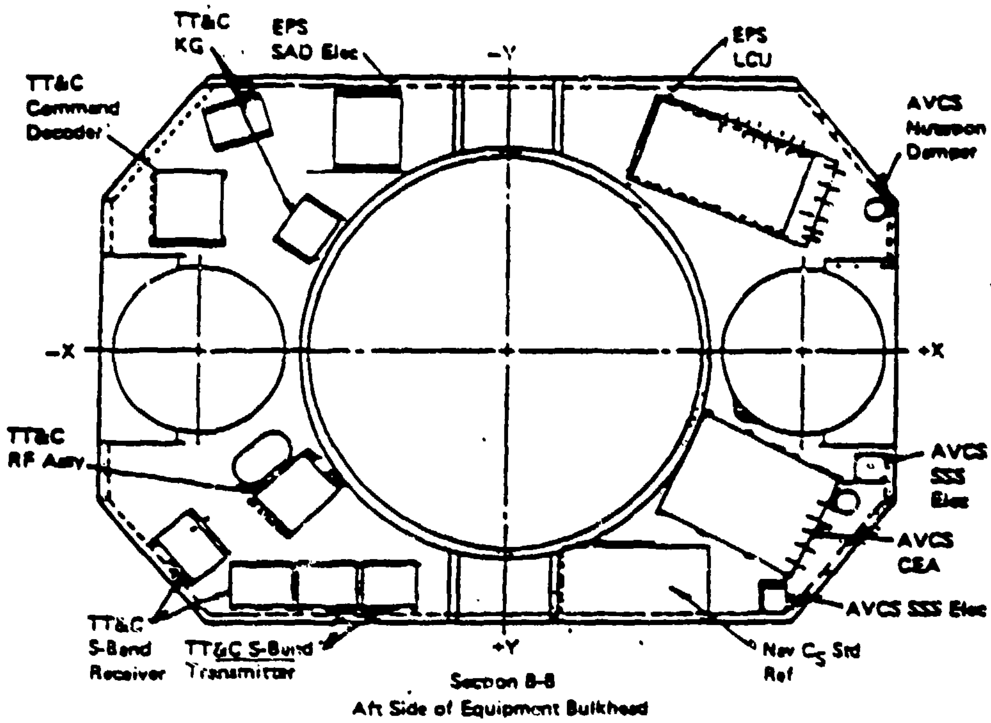


Figure C17-6 Cross-Sectional Sketch



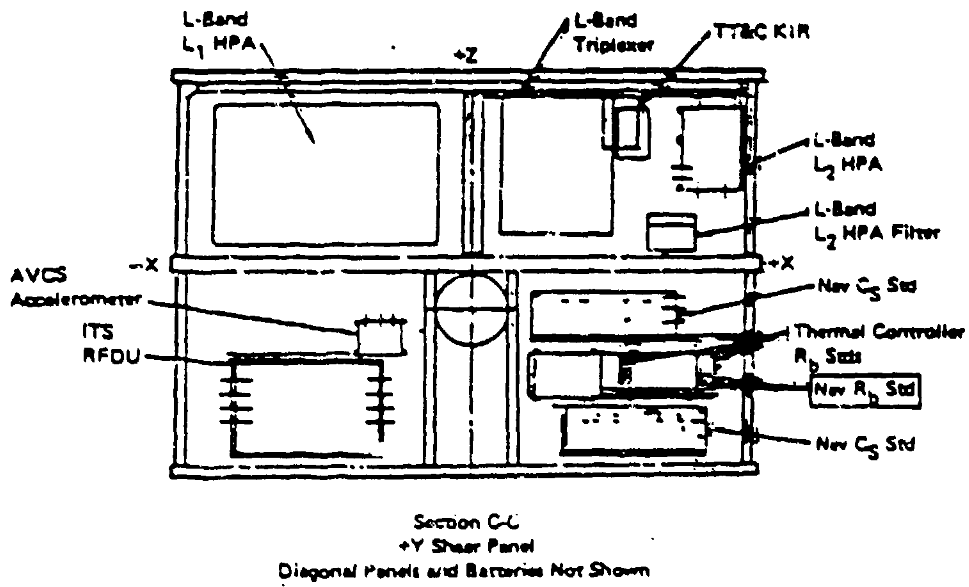
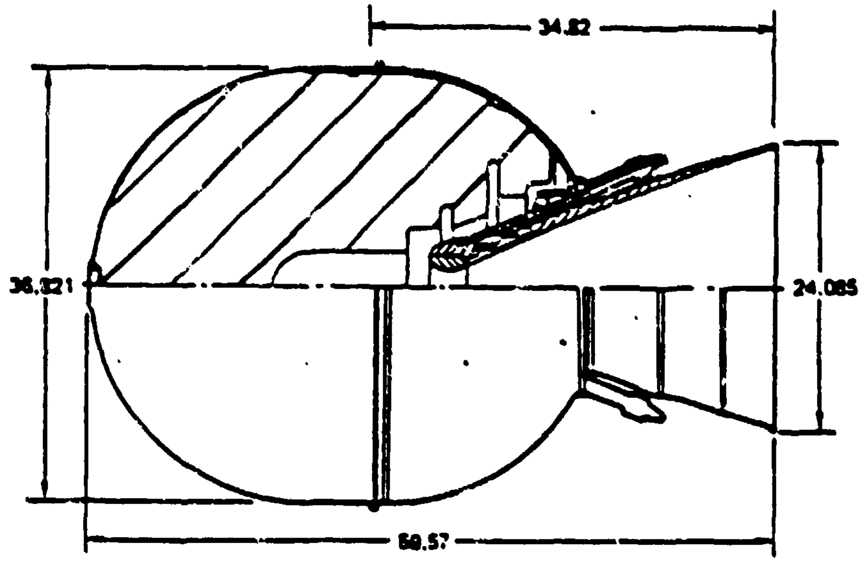


Figure C17-7 Space Vehicle Equipment Arrangement



CASE

Material	6Al-4V Titanium
Minimum Ultimate Strength, psi	170,000
Minimum Yield Strength, psi	160,000
Hydrostatic Test Pressure, psi	763
Minimum Burst Pressure, psi	[illegible]
Hydrostatic Test Pressure/Maximum Pressure	1.05
Burst Pressure/Maximum Pressure	1.25
Nominal Thickness, in.	0.065

NOZZLE

Exit Cone Material	2-D Carbon/Carbon
Throat Insert Material	3-D Carbon/Carbon
Initial Throat Diameter, in.	3.08
Exit Diameter, in	23.75
Expansion Ratio, Initial/Average	59.3/54.7
Expansion Cone Half Angles, Exit/Eff, deg	15.6/17.3
Type	Fixed, Contoured

LINER

Type	TL-H 318
Density, lbm/in.^{3}	0.038

IGNITER

Thiokol Model Designation	Model 2134
Type	SBA-ETA/TBI/Initiator/Toroidal Pyrogen
Minimum Firing Current, amperes	5.0
Circuit Resistance, ohms	1.1
No. of Squibs	2

PROPELLANT

Propellant Designation and Formulation	TP-H-3340
HTPB - 11%	
AP - 71%	
Al - 18%	

PROPELLANT CONFIGURATION

Type	Internal burning, Radial slotted

PROPELLANT CHARACTERISTICS

Burn Rate at 1000 psia (r_b), in./sec (60°F)		0.282
Burn Rate Exponent (n)		1.30
Density, lbm/in.^{3}		0.0661
Temperature Coefficient of Pressure (π_k), %/°F		
Characteristic Exhaust Velocity (C*), ft/sec		5110
Adiabatic Flame Temperature (T_f), °F		6113
Effective Ratio of Specific Heats	(Chamber)	1.13
	(Nozzle Exit)	1.18

CURRENT STATUS Qualification

Figure C17-8 Star 37XF TE-M-714 Motor Configuration and Data

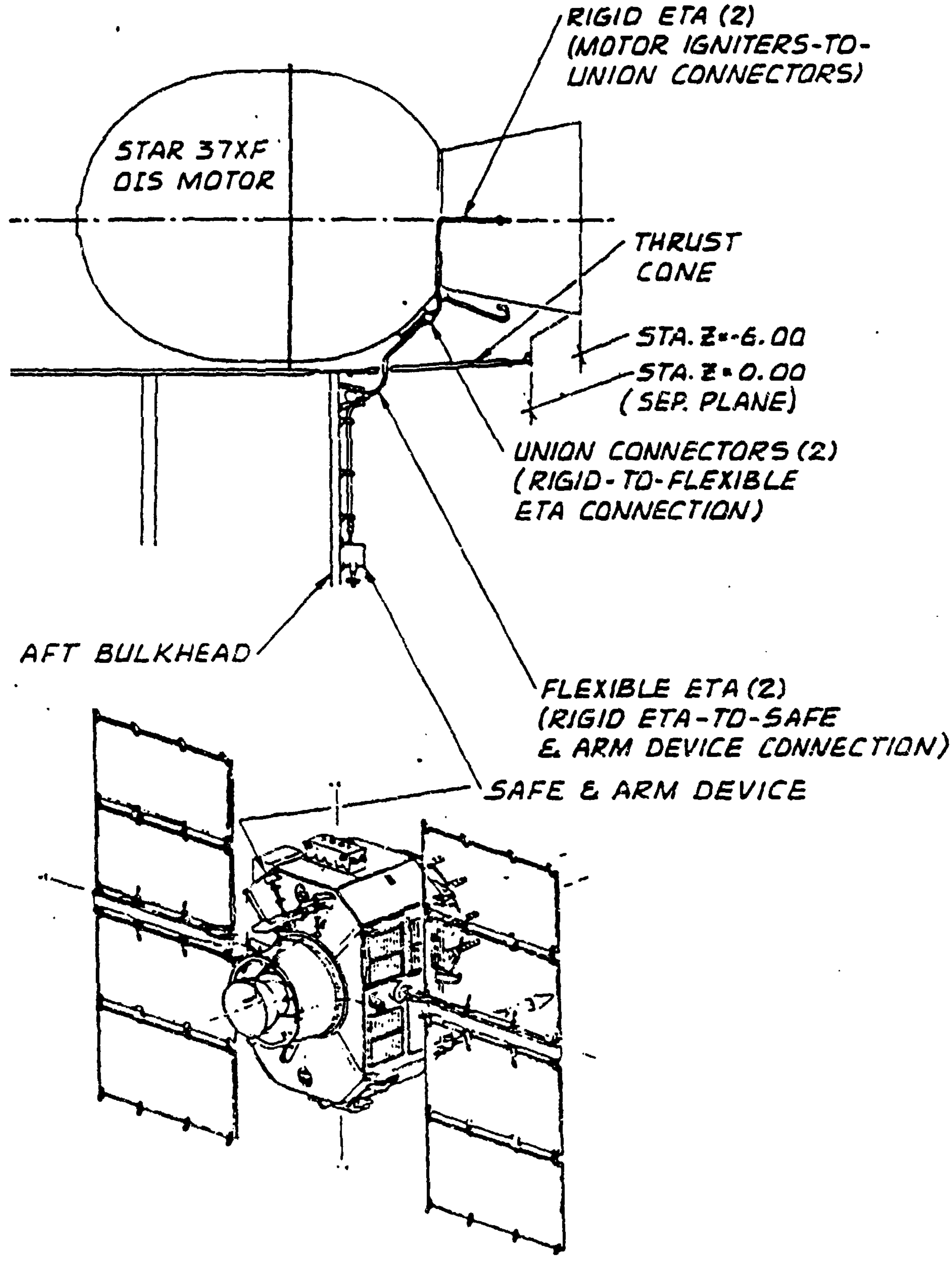


Figure C17-9 Location of SV Ordnance

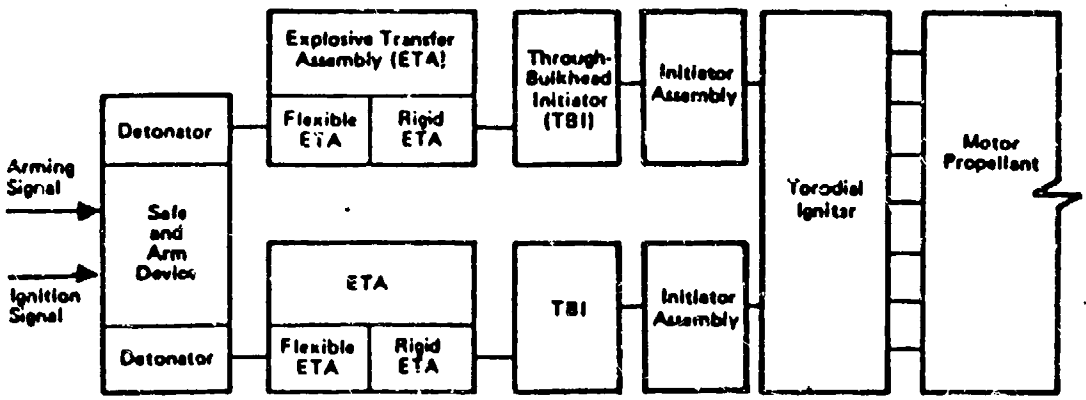


Figure C17-10 OIS Motor Firing Block Diagram

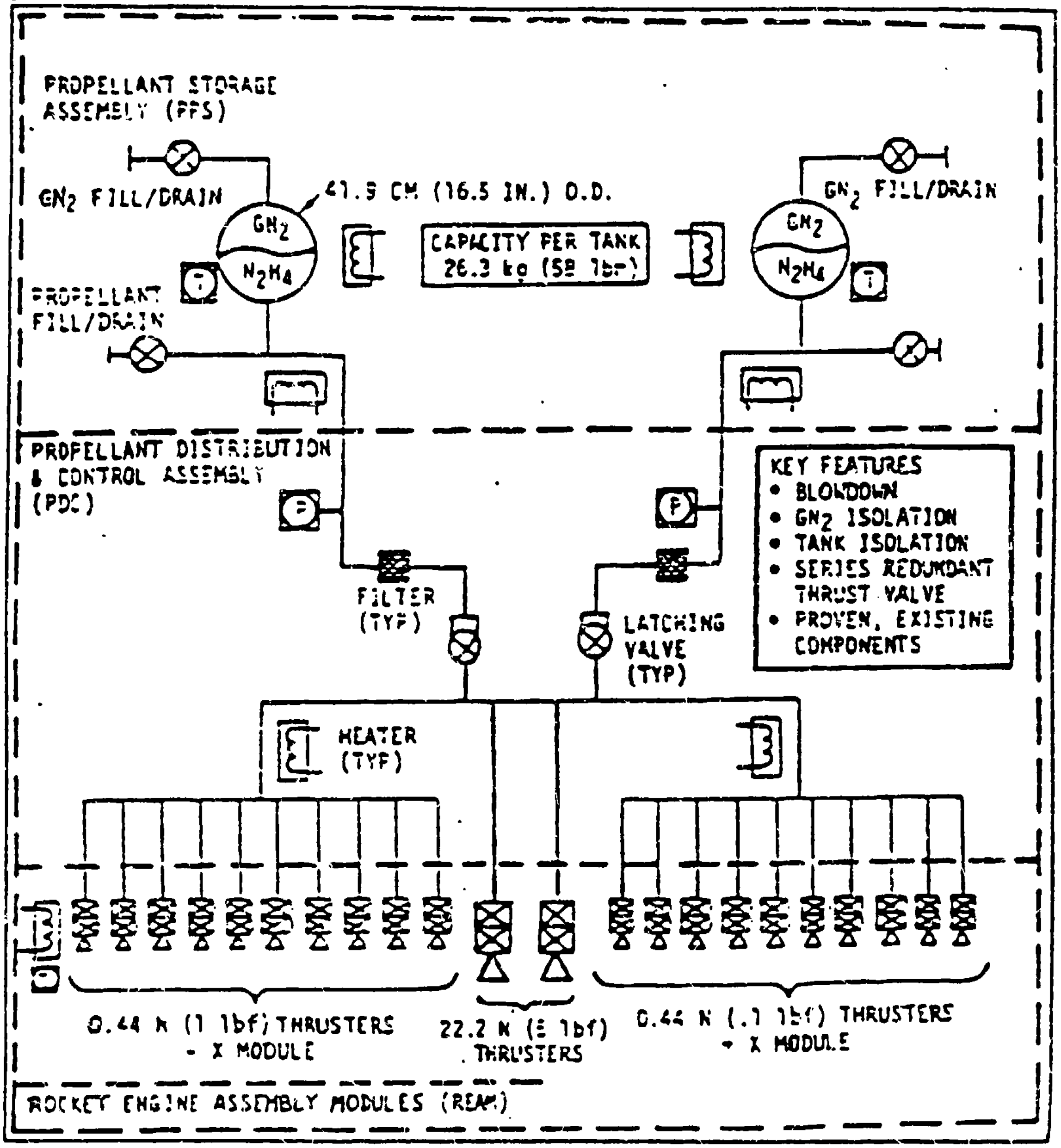


Figure C17-11 RCS Block Diagram

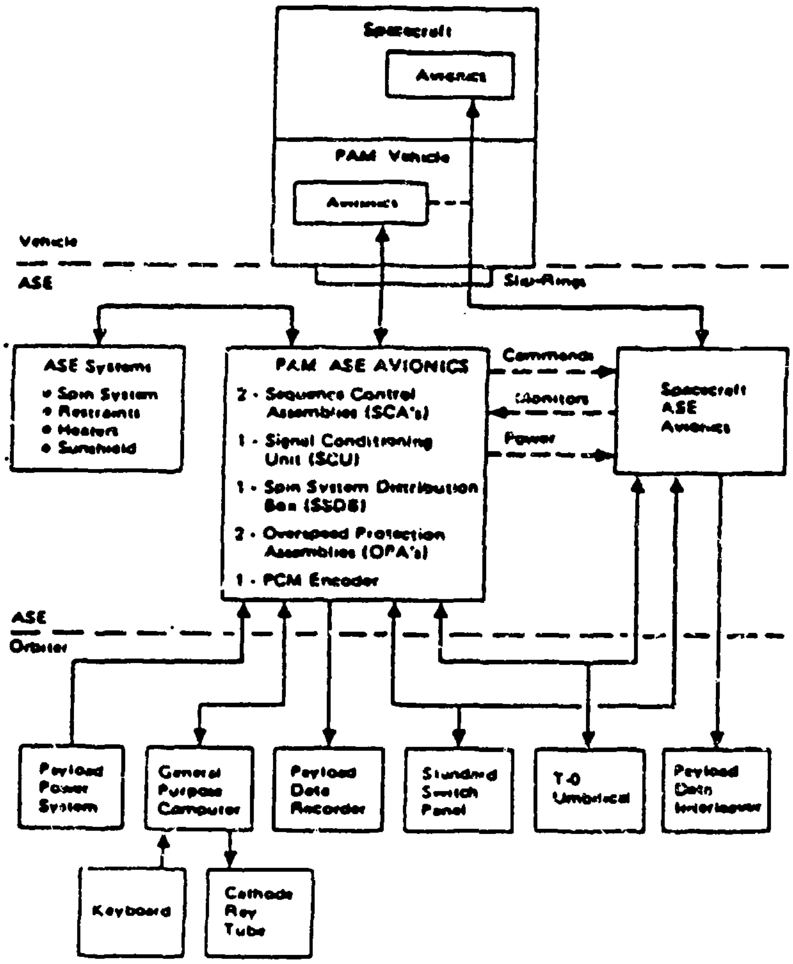


Figure C17-12 PAM Avionics

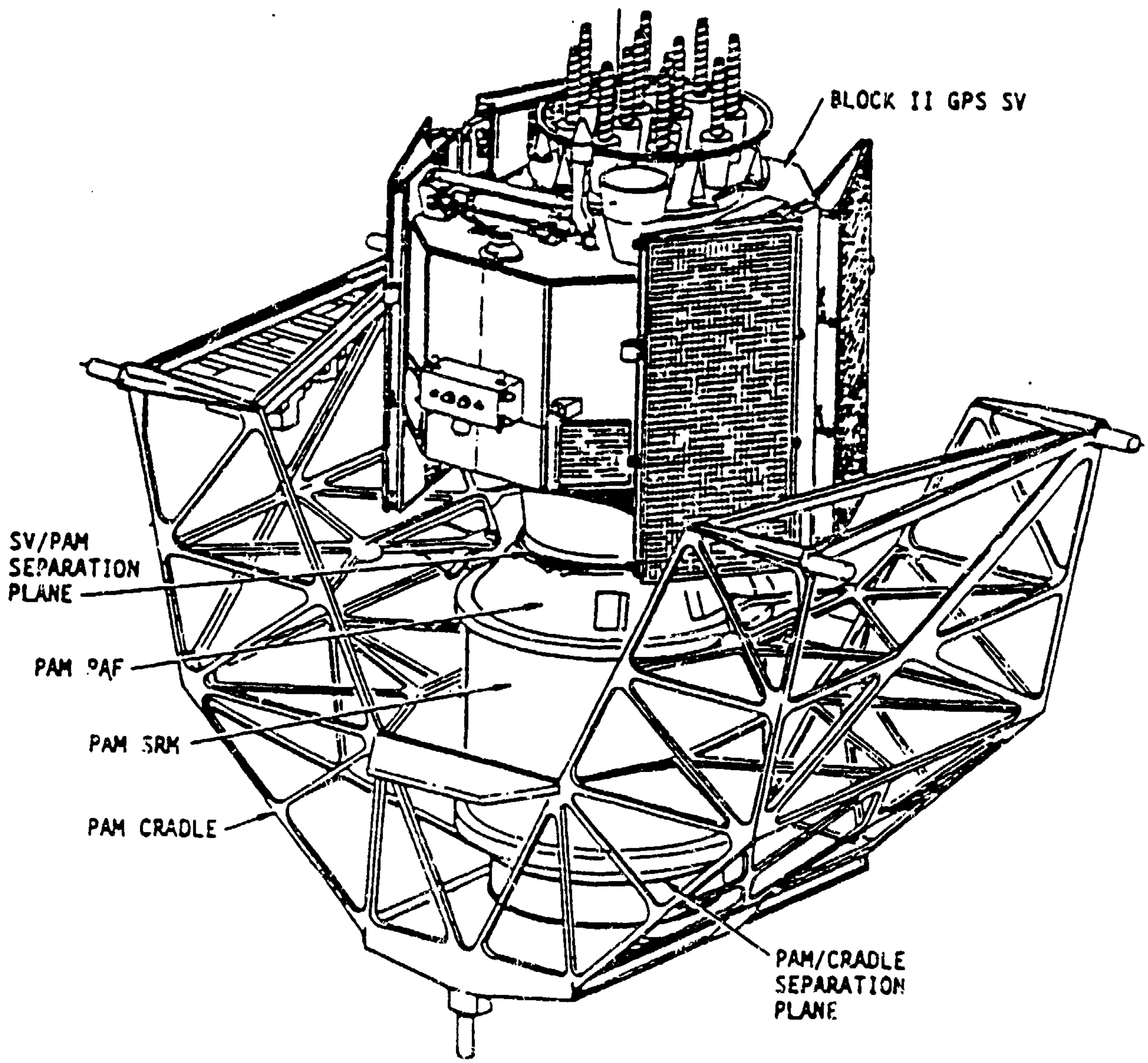


Figure C17-13 GPS Block II Space Vehicle on PAM
(Thermal Control System Not Shown)

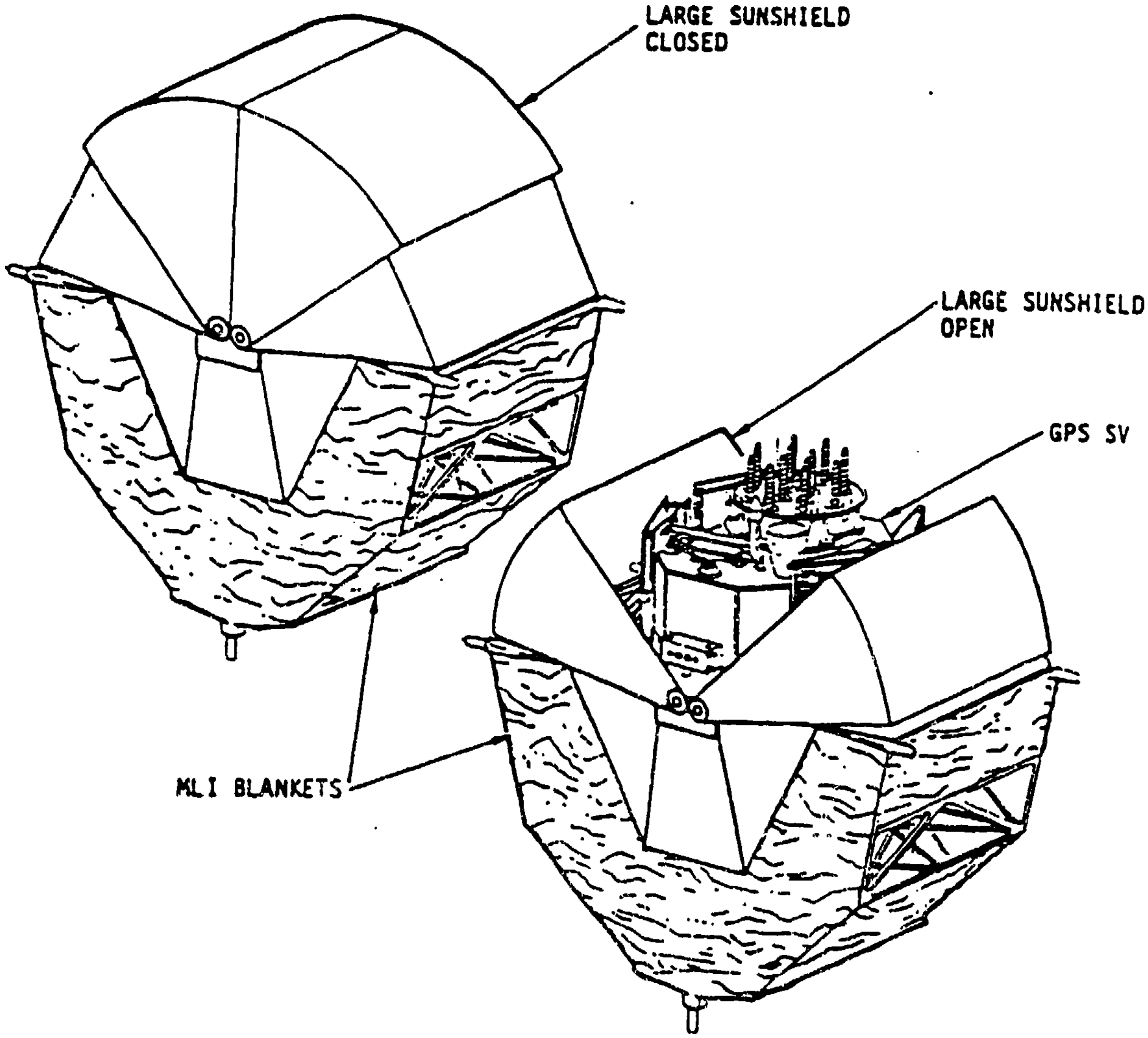


Figure C17-14 GPS Block II Space Vehicle on PAM

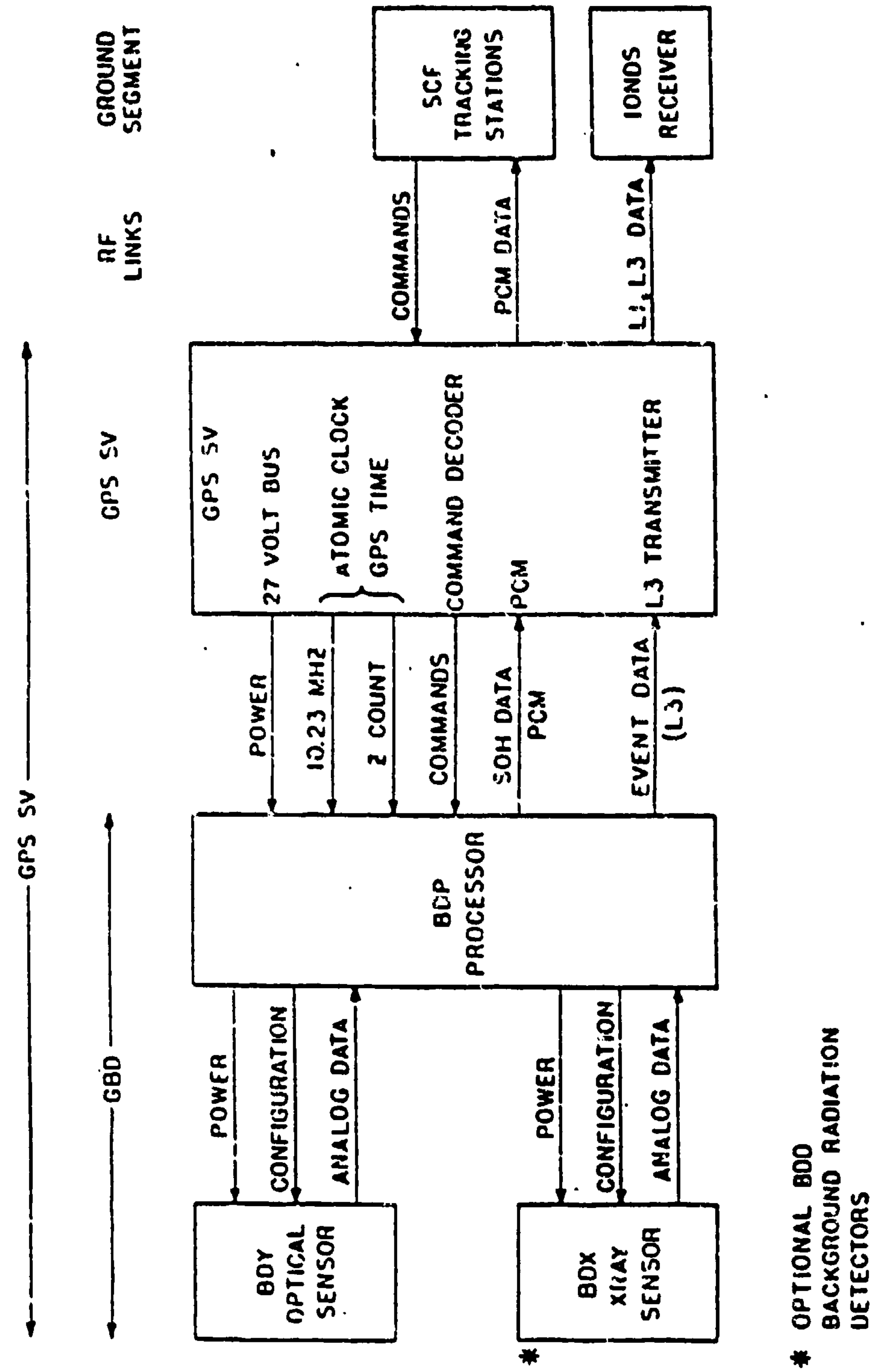


Figure C17-15 GPS Integrated Operational Nudet Detection System (IONDS) Segment

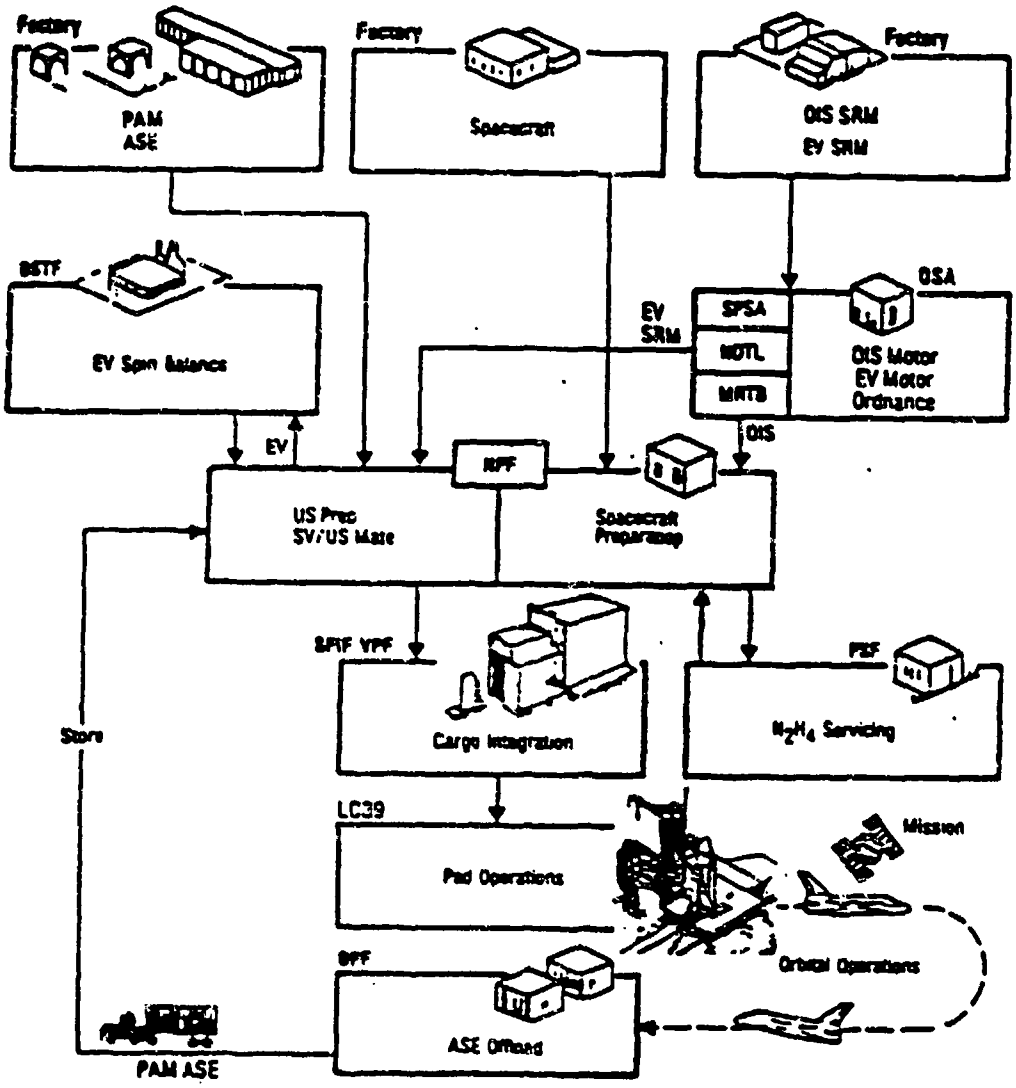


Figure C17-16 Navstar Launch Base Process Flow

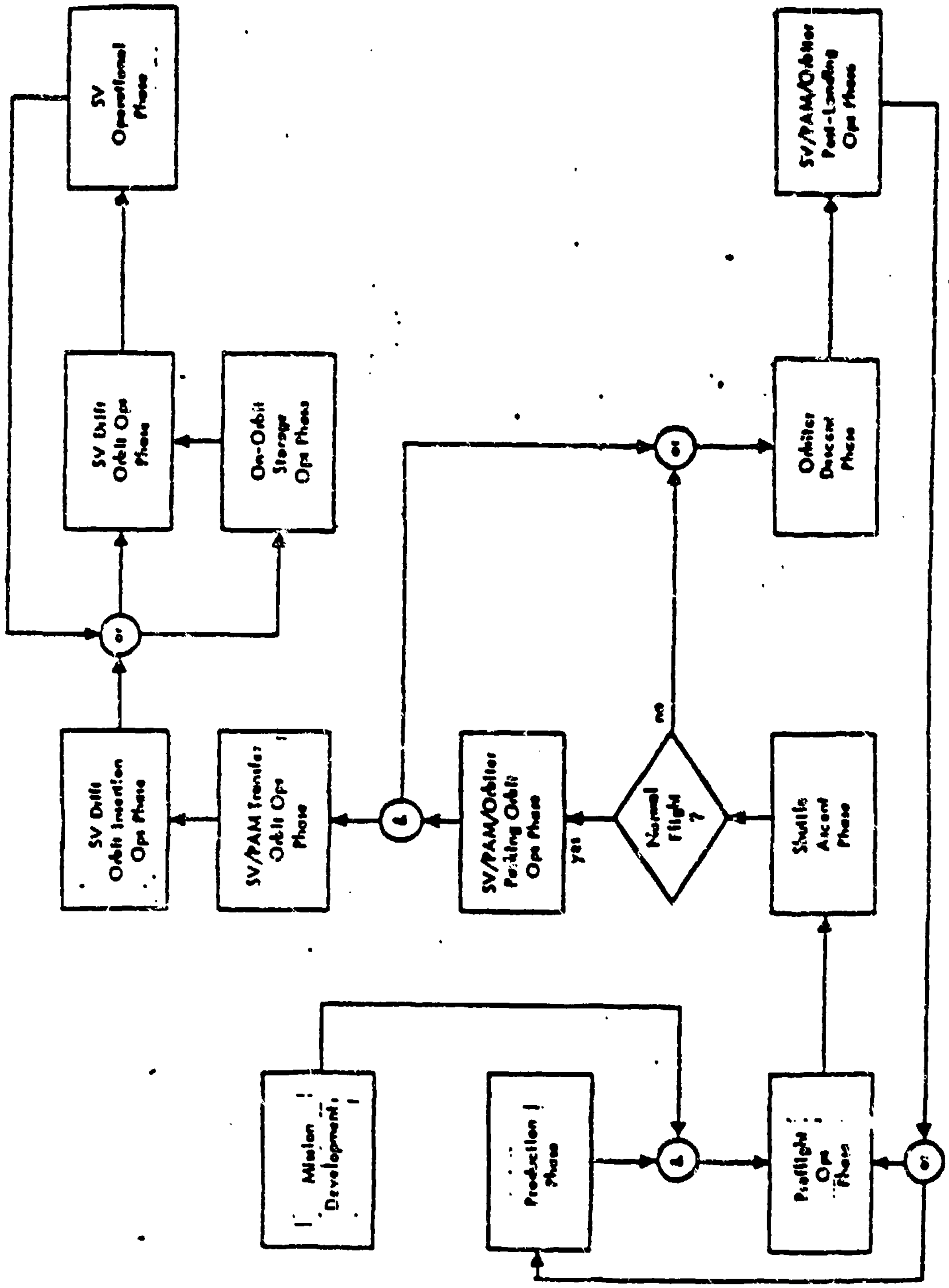


Figure C17-17 Shuttle-Launched GPS Missions

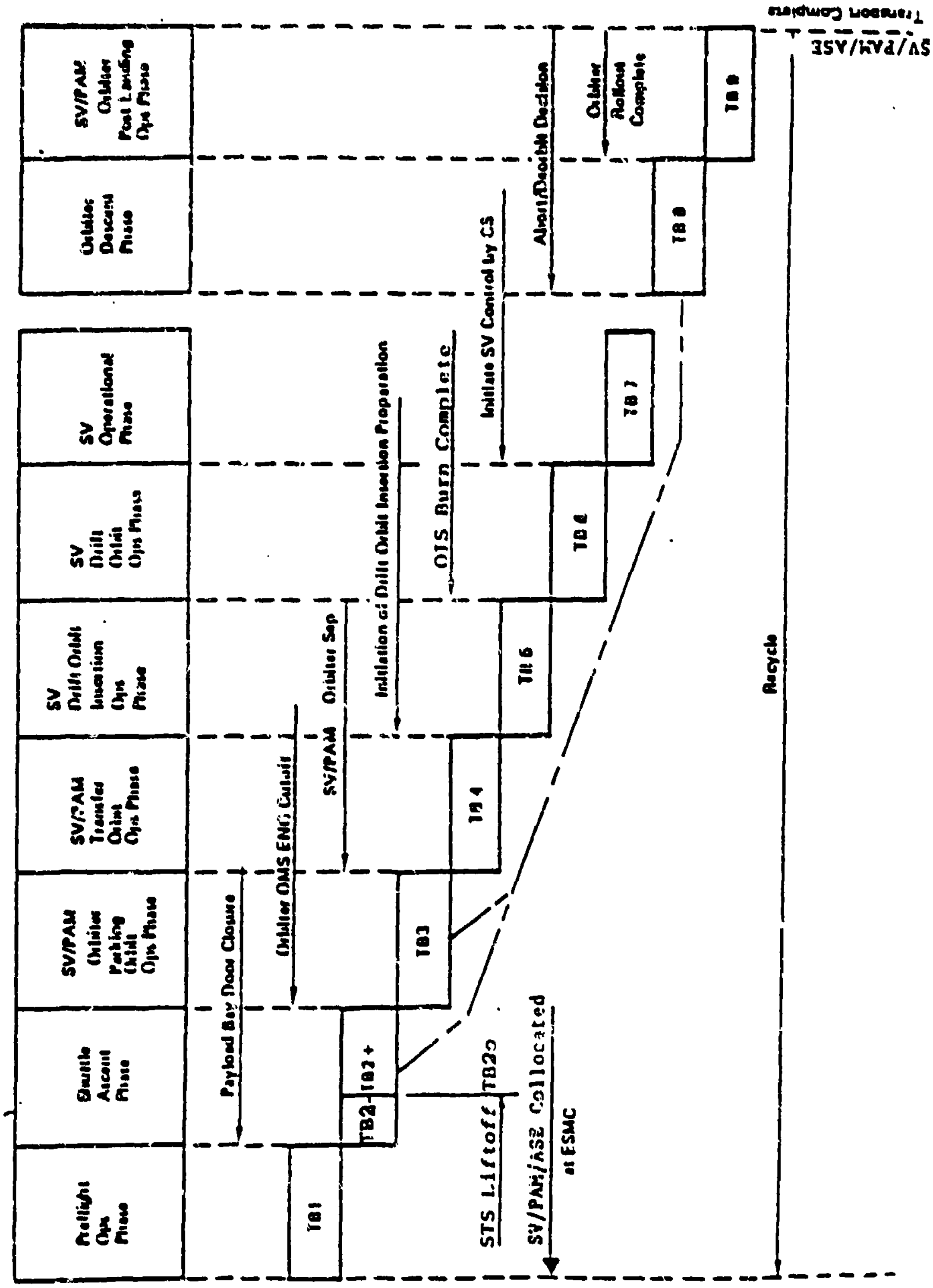


Figure C17-18 GPS Mission Phase/Time Base (TB) Relationships

Appendix C18
P80-1 Space Vehicle System

APPENDIX C18
P80-1 SPACE VEHICLE SYSTEM
(INCLUDING ION AUXILIARY PROPULSION SYSTEM (IAPS))

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APPENDIX C18
P80-1 SPACE VEHICLE SYSTEM
(INCLUDING ION AUXILIARY PROPULSION SYSTEM (IAPS))

C18.0 INTRODUCTION

The following data were extracted from the Annotated Bibliography Documents, Ref 024 (August 1984, update), and Ref 043 (June 1983, Rev. B, update). Also used were data from the P80-1 program, Formal Training Space Vehicle System Overview Document, Rockwell International, Satellite Systems Division, Feb. 1985.

C18.1 GENERAL DESCRIPTION

The P80-1 Space Vehicle System, consisting of a spacecraft and required Airborne Support Equipment (ASE), has been developed and prepared by the United States Air Force to meet Space Transportation System (STS) safety requirements. It will be launched aboard the Space Shuttle from the NASA Eastern Launch Site. See Figures C18-1, C18-2, C18-3, C18-4, and C18-5.

C18.2 SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

C18.2.1 Structures

Major P80-1 structures include the ASE cradle, propulsion module, the spacecraft, and associated experiments.

C18.2.2 Propulsion Subsystems

Orbi njection Subsystem (Propulsion Module) - This system injects P80-1 into final orbit. See Figures C18-6 through C18-10.

C18.2.3 Hydrazine Reaction Control (HRC) System

The function of the HRC system shown in Figure C18-11 is to provide control during orbit injection system burn.

C18.2.4 Cold Gas Reaction Control System

The cold gas reaction control system shown in Figures C18-12 and C18-13 provides 3-axis attitude control after Orbiter separation through on orbit reaction wheel turn on. It consists of two tanks which contain 10 pounds of GN_2 each, at 3700 psi.

C18.2.5 Electrical Power Subsystem (EPS)

The electrical power subsystem (EPS) shown in Figures C18-14 and C18-15 is comprised of components that store energy (batteries) and charge (solar arrays) and that control and distribute power. The EPS functions are the following:

1. Spacecraft Power Requirements
   - Solar Array Power Generation
   - Battery Energy Storage for Eclipse and Peak Power Requirements
   - Bus Voltage Control
   - Fault Protection, Switching, and Distribution to Loads

2. Special Power Requirements
   - 51-90 Vdc to IAPS Experiment
     - o 60% Mode A, 30% Mode B, 10% Mode C
     - o 7-10 Days Continuous Operating Capability for One Thruster

3. Special Control Functions
   - EUV Power Control (On in Eclipse, Off in Sunlight-with Override)
   - Sun Safe Indication of Sunlight on Array

4. Safing, Arming, and Firing Circuitry for Electroexplosive Devices and Non-Explosive Initiators

5. Airborne Support Equipment Electrical Functions
   - Battery Energy Storage
   - Fault Protection, Switching, and Distribution
   - Display and Control Panel Capability
   - Deployment EED Safing, Arming, and Firing

C18.2.6 Attitude Control and Determination Subsystem (ACDS)

The ACDS provides for attitude sensing, logic, and control to maintain satellite orientation and provide rate-of-change information.

C18.2.7 Telemetry, Tracking, and Command (TT&C) Subsystem

The TT&C subsystem provides telemetry, tracking, and command functions to support the operations.

C18.2.8 Thermal Control Subsystem (TCS)

The TCS maintains all components within their prescribed temperature limits. It has the following elements: coatings and paint, insulation, and heaters.

C18.2.9 Airborne Support Equipment (ASE)

The ASE for the P80-1 cargo element consists of the satellite support structure and a control panel in the Orbiter aft flight deck, all of which remain with the Orbiter after spacecraft separation.

C18.2.10 ION Auxiliary Propulsion System (IAPS)

The IAPS experiment is a mercury ion thruster system. Two units, mounted on the spacecraft, each contain about 19 pounds of mercury at 35 psia. The IAPS is built by Hughes Aircraft Company. (See Figures C18-16 through C18-19).

C18.2.11 IAPS System Configuration

As delivered to the spacecraft contractor, the IAPS Experiment consists of a number of Hughes-furnished items, which have been installed in two Rockwell-furnished IAPS Module structures. One of these structures is to be mounted on the -Z surface of the P80-1 spacecraft, and the other is to be mounted on the -X surface. Each module contains a mercury feed system. Each mercury feed system, as shown schematically by Figure 1, includes a mercury reservoir, a pressure transducer, a solenoid-operated bi-stable latching valve, and a mercury feed line. The spherical portion of the reservoir (Item (1)) and the feed line (Item (11)), up to the closed latching valve (Item (8)), contain mercury. This volume of mercury is under a lower pressure (35 psia) resulting from the gaseous nitrogen pressurant (Item (4)) in the gas pressure shell of the mercury reservoir (Item (5)). The pressurant is introduced through the gas fill/vent valve (Item (6)). A pressure transducer (Item (7)) is provided to monitor mercury feed system pressure during ground operations and during in-orbit operation. By means of a mechanical joint (Item (9)) the mercury feed system connects to the inlet fitting of the thruster, gimbal, beam shield unit.

C18.2.11.1 IAPS Mercury Tanks - Two mercury reservoirs are used on the IAPS flight experiment. Each has an approximate volume of 80 in.^{3} (0.064 ft.^{3}), of which 48% (approx. 38.4 in.^{3}) is occupied with mercury. The remaining 41.6 in.^{3} is filled with pressurant gas (100% N_2).

The Propellant Tank operates at 35 psia (20.3 psig) maximum, and has been proof-tested to 50 psig, 2.46 times the maximum operating pressure. Burst pressure calculations for the reservoir indicate the minimum burst pressure to be 559.7 psi.

This assessment indicates that the IAPS mercury tanks do not present a hazard from to explosion/implosion and/or debris due to the ascent/descent depressurization/repressurization of the Orbiter.

All IAPS containers were determined to have a minimum safety factor of 2.5. Structural collapse of the IAPS containers considered the containers as evacuated. The containers were considered unpowered during ascent, on orbit, and during an aborted landing, therefore not performing a safety critical function and not presenting a hazard to the Orbiter.

None of the IAPS containers analyzed are considered sealed containers, none contain safety-critical inhibit circuits, and none contain fluids which can present a hazard to the Orbiter. Upon venting, the gases expelled are the normal atmospheric constituents.

This assessment indicates that no IAPS-supplied container will present a hazard from explosion/implosion or debris due to the ascent/descent depressurization/repressurization cycle of the Orbiter. This structural design analysis indicates that there are no failure modes in these mission phases.

C18.2.11.2 Background Information (Mercury) - Some potential hazard exists because of the elemental mercury contained within the IAPS experiment (8.6 Kg in each of two reservoirs). However, because the design of the equipment and the manner in which it is to be handled and tested, the probability of this potential hazard becoming a reality is extremely low if handling, assembly, and test operations are conducted with reasonable care. Mercury is a metal that is liquid at room temperature. It is silver in color and extremely heavy. For comparison, a gallon of mercury weighs approximately 112.8 lb, while a gallon of water weighs approximately 8.3 lb. Table C18-1 summarizes properties of mercury.

As temperature increases, the mercury vapor pressure increases. The vapor pressure of mercury is more than doubled by each 10°C increase in local temperature. The amount of vapor produced varies directly as the surface area, if all other conditions remain constant.

In still air, the heavy vapor formed at the surface of the liquid collects close to the exposed mercury surface and opposes further vaporization. In this case, the mercury vapor concentration near the exposed liquid mercury surface would be high (at or near saturation), while the concentration just a few inches above the surface would be at or near zero. In the presence of convective air currents, the vapor concentration--at any given location--would be substantially lower than saturated conditions, but the contaminating vapors would be disseminated over a much larger volume.

Clean mercury is readily divided into globules, but these globules coalesce when brought together again. On the other hand, mercury soiled by grease and dust will divide into minute globules which do not coalesce. The resulting separate particles are so minute that they cannot be seen with the naked eye, and the mass assumes a dull grayish color. A comparatively small amount of mercury (in an almost invisible state of subdivision) may present an exposed surface equivalent to that of a large mass of mercury coalesced in a dish or a pan. The vapor has neither taste nor odor to give warning of its presence.

It has been determined that prolonged exposure to mercury vapor concentrations of no more than 0.05 milligrams of mercury per cubic meter of air will not result in any health hazard to personnel. Brief exposures to higher concentrations are not injurious, but poisoning may occur in workers exposed to concentrations above 0.05 mg/m^3. In addition to inhalation of mercury vapor in air, exposure may result from inhalation of a mist of the metal or from liquid mercury being absorbed directly through the skin or being ingested.

Paragraph 5.4.3.7 of MIL-STD-25886A (USAF), mercury and many compounds containing mercury can cause accelerated stress cracking of aluminum and titanium alloys.

Table C18-1 Properties of Mercury

Chemical Formula	Hg
Flammability	Nonflammable, but mercury forms explosive combinations with ammonia and with nitric acid and ethanol.
Odor	None
Natural State at Normal Temperatures and Pressures	Liquid
Molecular Weight	200.61
Specific Gravity	13.534 at 25°C
Boiling Point	357.3°C
Melting Point	-38.9°C
Vapor Pressure	0.000775 mm at 10°C 0.00182 mm at 20°C 0.00407 mm at 30°C
At 25°C and 760 mm Hg	1 ppm of vapor: 0.0082 mg/liter 1 mg/1: 121.5 ppm
Solubility	Insoluble in water; soluble in inorganic acids.
Corrosiveness	Mercury is not considered corrosive in the ordinary sense. However, many metals dissolve in or amalgamate with mercury.

C18.3 MISSION SCENARIO

The flight operations for the P80-1 program are addressed in the DOD-STS P80-1 Mission Plan, SAMSO-LV-0032, dated October 1982. The following paragraphs provide a summary of operations. Mission Overview is shown in Figure C18-20.

During the prelaunch terminal countdown, the health of the P80-1 and the STS are verified. The P80-1 is powered off during ascent; therefore no P80-1 status telemetry is available. Following verification of a safe park orbit, the Orbiter GPCs are reconfigured to the OPS-2, the on-orbit configuration.

Once the SSV is on station, P80-1 deployment operations can begin as summarized below:

The payload bay doors are opened and thermal radiators activated approximately 01:15 mission elapsed time (MET). Accomplishment of these two sequences is critical to avoid a mission abort contingency due to Orbiter payload by heating constraints.

The crew then performs an IMU alignment, and the Orbiter Ku-Band antenna is deployed on a non-interference basis of P80-1 for potential contingency operations. The Ku-band system will nominally remain off until P80-1 is TBD ft away from the Orbiter. It is assumed that the orbiter is now in an RCS attitude hold. Therefore, tracking of the coherent S-band Doppler via TDRS generates meaningful orbit-determination data for resetting of the P80-1 onboard clock. Power is applied to P80-1 via the Orbiter AFD panel approximately 01.48 MET. A TT&C clock reset is performed at 01:50 MET.

At approximately 01:51 MET, the P80-1 predeployment checkout is initiated via a pass over TCS.

A final "Go for deployment" is received at 04:45 MET, and at 05:15 MET the deployment sequence is initiated. Physical separation is accomplished at 05:17 MET with separation springs providing a minimum separation velocity of 1.0 ft/sec. P80-1 enables a cold-gas attitude control system to arrest any satellite tipoff rates during deployment.

Approximately five minutes after deployment, the Orbiter performs a separation maneuver (approximately 29 ft/sec) to provide a safe separation distance within 45 minutes from deployment.

After verification of the separation maneuver, full operational control of P80-1 is assumed by the AFSCF. The P80-1 PIM enable is sent via ground command (NHS) at 06:23 MET for a perigee burn at 06:43 MET. At 06:35 MET the Orbiter maneuvers to a preferred orientation to minimize particulate damage to the Orbiter. Following the P80-1 perigee burn, the Orbiter resumes on-orbit activities.

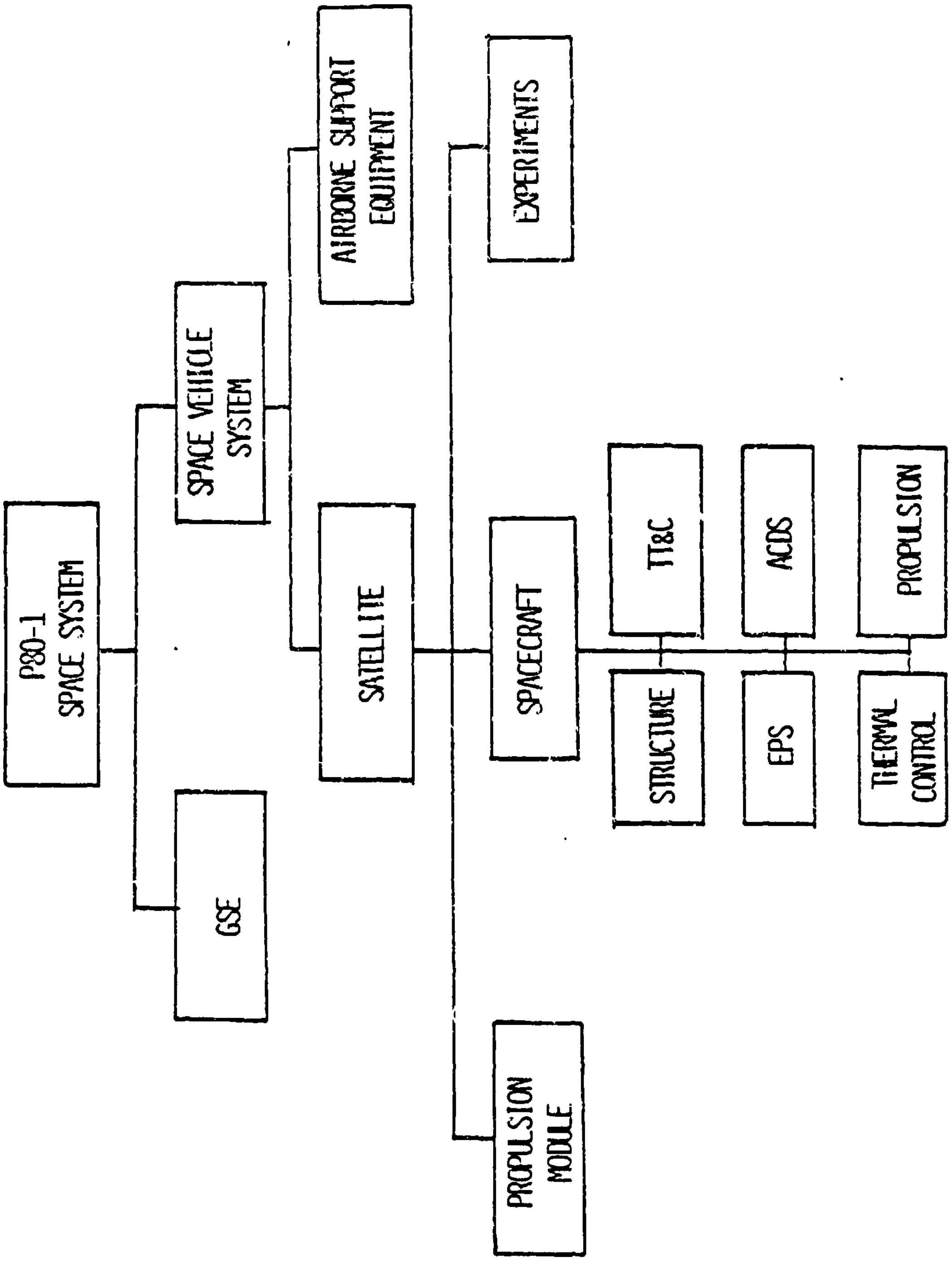


Figure C18-1 P80-1 Space System Elements

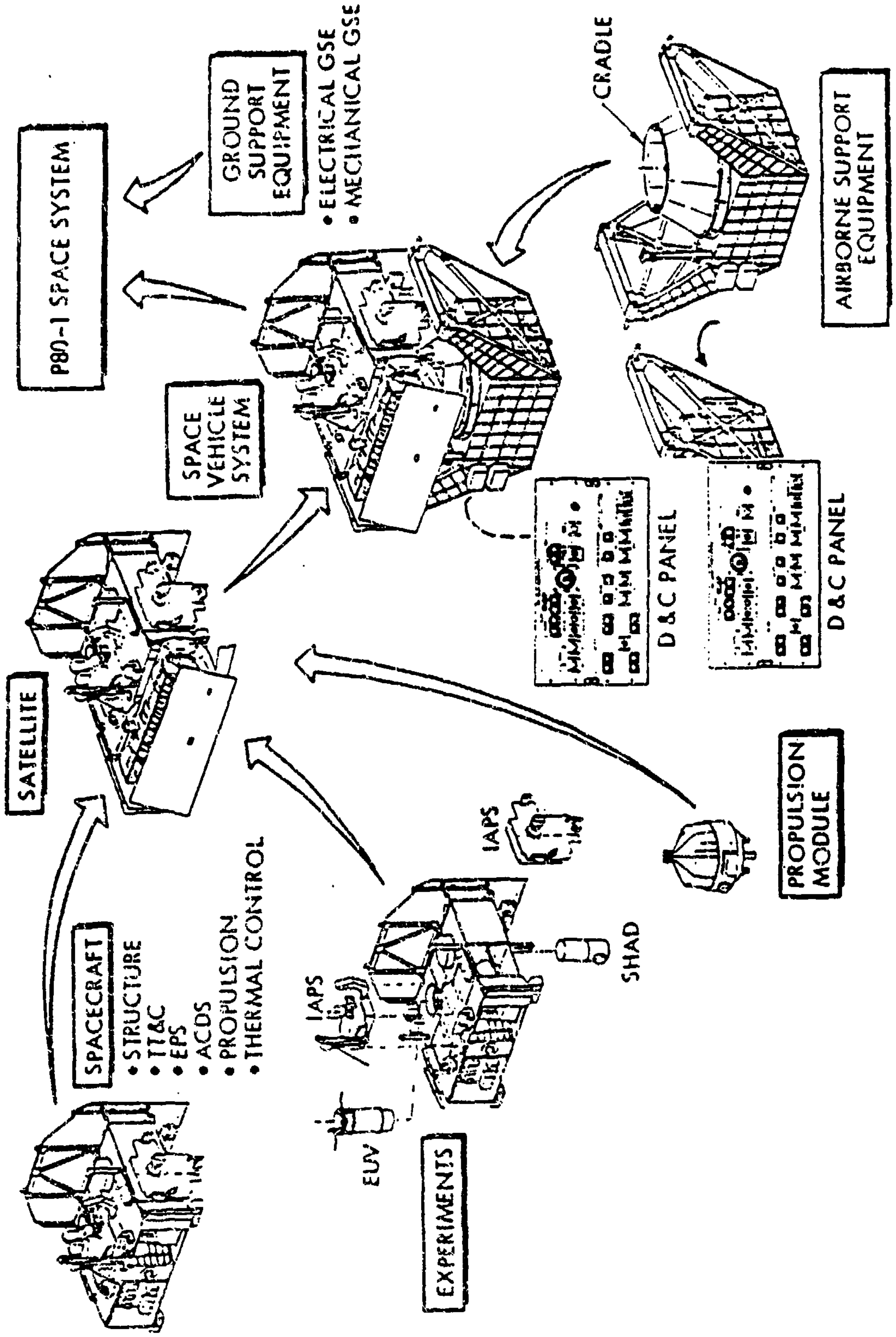


Figure C18-2 Space System Elements (WTR)

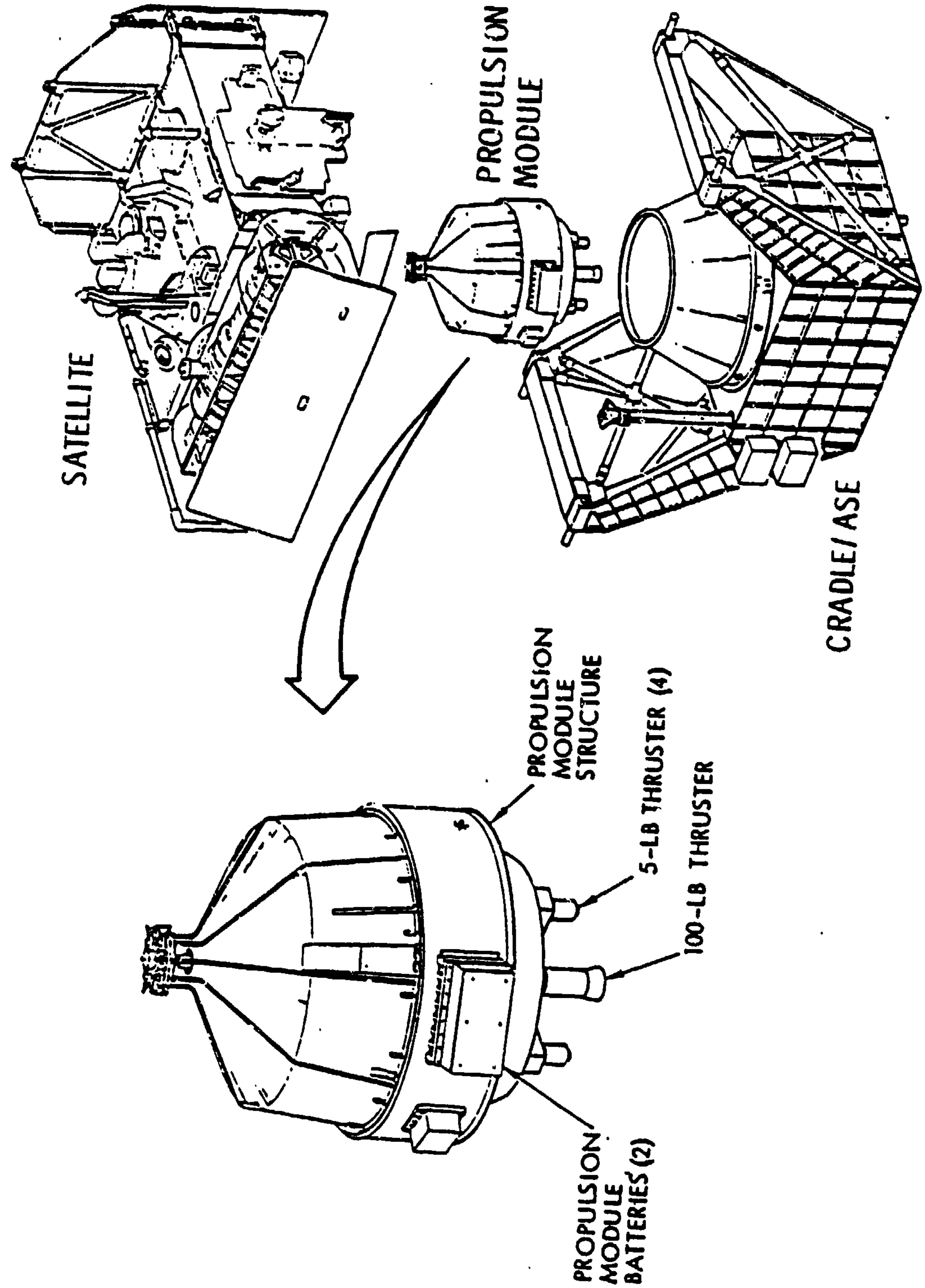


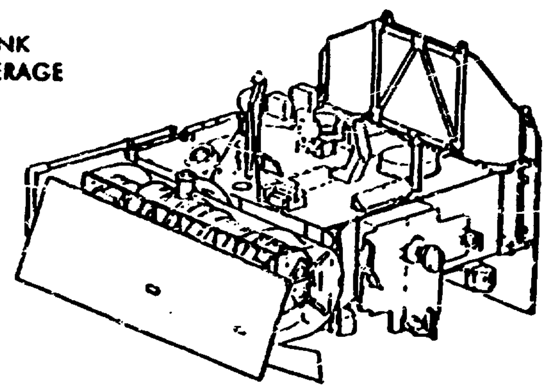
Figure C18-3 P80-1 Satellite

TT&C

- CARRIER 1
  - 32 KBPS DATA RATE
  - PRN RANGING
- CARRIER 2
  - 1.024 MBPS DATA RATE
  - TRE REAL TIME RECORDER PLAYBACK
- ENCRYPTION FOR UPLINK & DOWNLINK
- NEAR OMNIDIRECTIONAL ANT. COVERAGE
- THREE TAPE RECORDERS
  - 32 & 1024 KBPS RECORD RATES
  - 1024 KBPS PLAYBACK RATE

OIS/PROPULSION

- PROPULSION MODULE
  - TDRSS TYPE 42 IN. DIA DIAPHRAGM TANK
  - ONE 100-LB N_2H_4 THRUSTER
  - FOUR 5-LB N_2H_4 THRUSTERS
  - TRANSFER P80-1 FROM SHUTTLE ORBIT TO FINAL ORBIT
- NITROGEN RCS ATTITUDE CONTROL
  - TWELVE 0.2 LB THRUSTERS
  - PARKING/TRANSFER ORBIT
  - BACKUP TO WHEELS DURING OPERATIONAL ORBIT



ACDS

- FOUR SKEWED REACTION WHEELS
- MAGNETIC MOMENTUM DUMP
- PRIMARY ATTITUDE DETERMINATION SYS (PADS)
  - ATTITUDE CONTROL
  - ATTITUDE & RATE DATA
- THREE AXIS STABILIZED
- BACKUP SYS HORIZON SCANNER
- SUN SAFE HOLD MODE

STRUCTURE

- AL HONEYCOMB FLAT PANELS
- MONOCOQUE THRUST CYLINDER

THERMAL CONTROL

- COLD BIASED SYSTEM
- SUBSYSTEM RADIATORS
  - EXTERNAL PANEL SURFACE AT COMPONENT LOCATIONS
- MULTI-LAYER INSULATION BLANKETS
- HEATERS (REDUNDANT CIRCUITS)
  - HYDRAZINE SYSTEM
  - IAPS
  - EPS COMPONENTS
  - TT&C COMPONENTS
  - ACDS COMPONENTS

ELECTRICAL POWER

- 130 FT^2 ARRAY
- 35.2° TILT ANGLE
- SINGLE DEGREE OF FREEDOM ORIENTATION
- (2) 35 AH NI-CD BATTERIES
- 24 TO 33 VDC TO LOADS
- (2) 85 AH AgZn BATTERIES (PM)

Figure C18-4 P80-1 Spacecraft

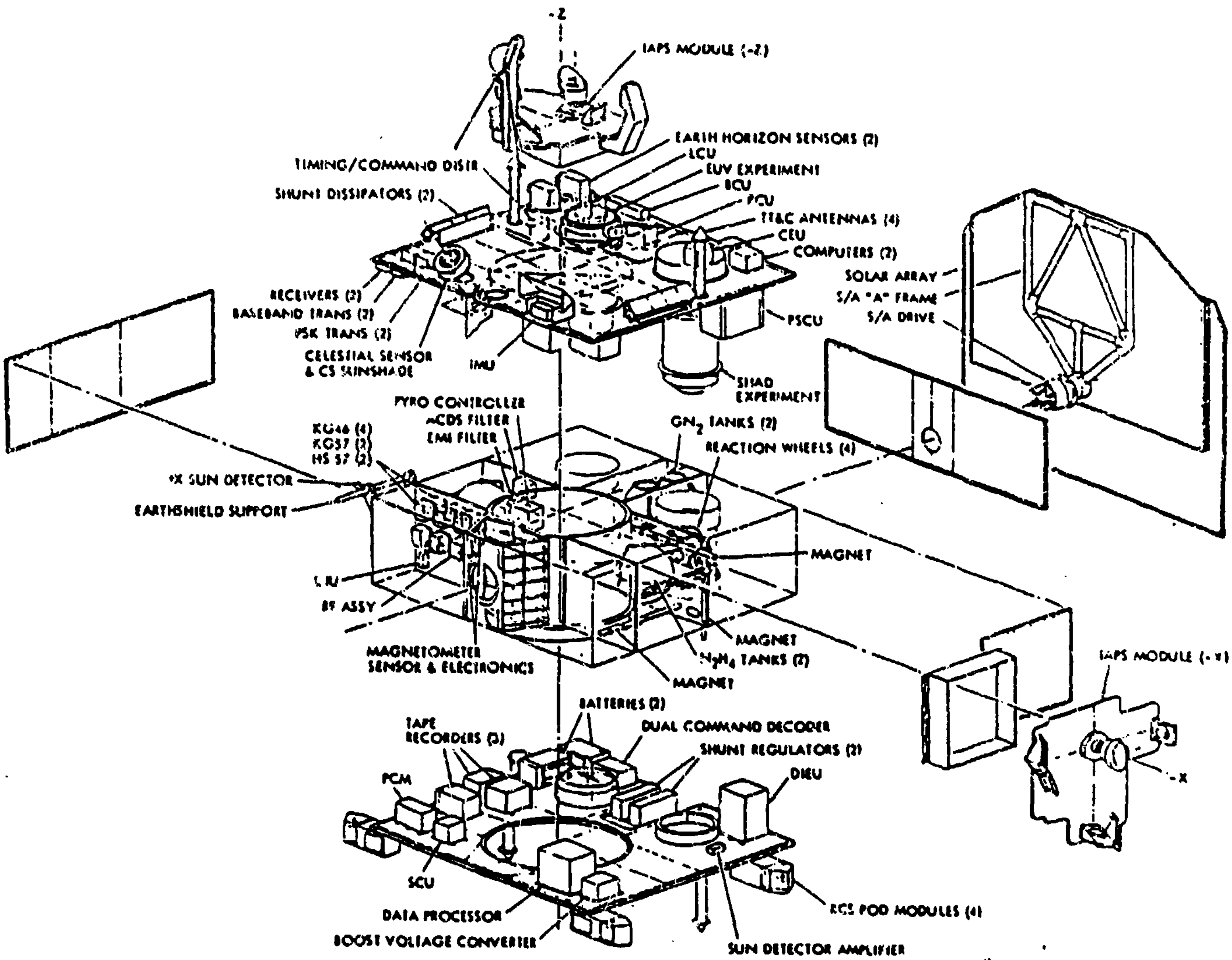


Figure C18-5 Spacecraft Component Locations

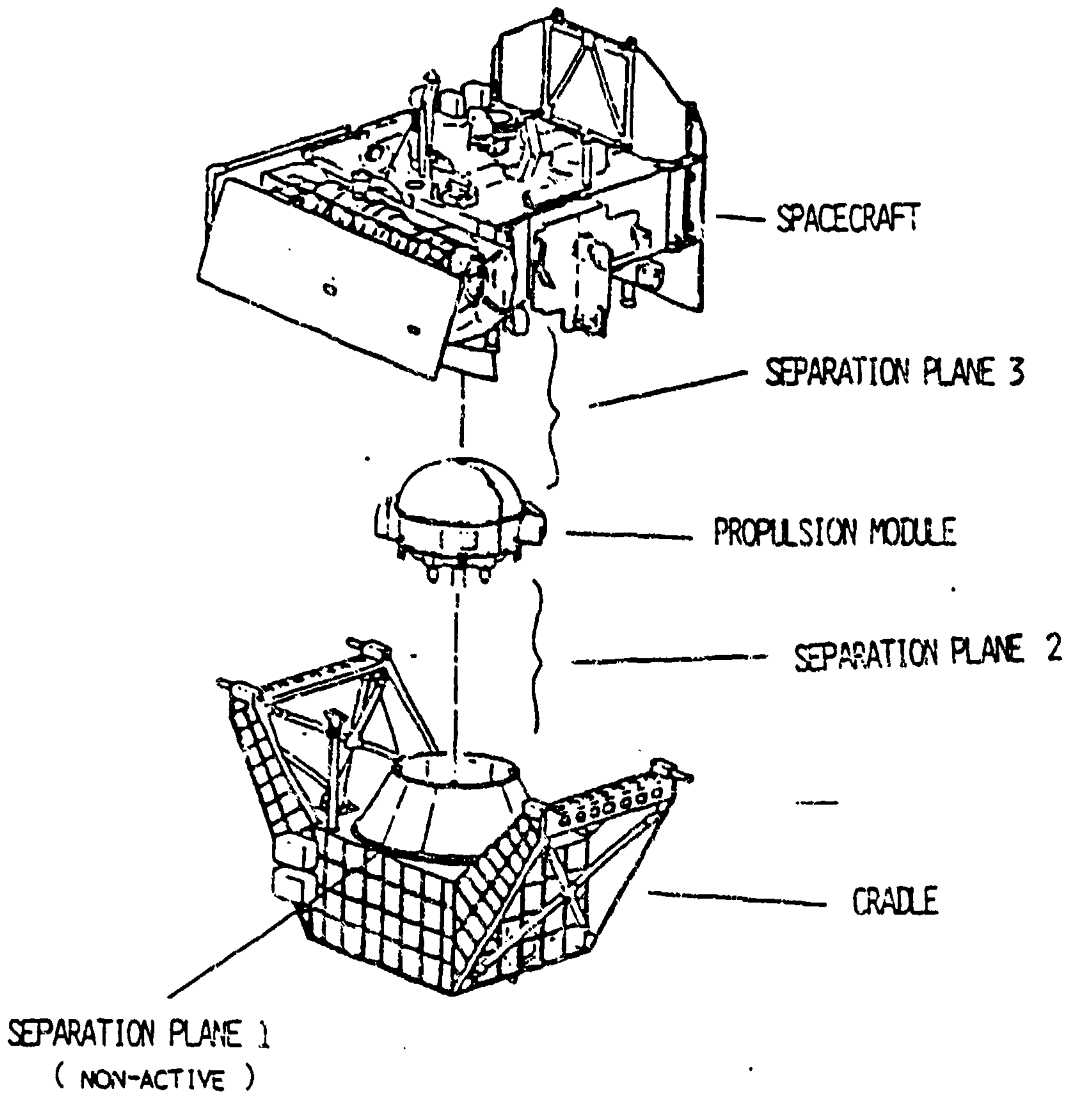


Figure C18-6 Separation Planes

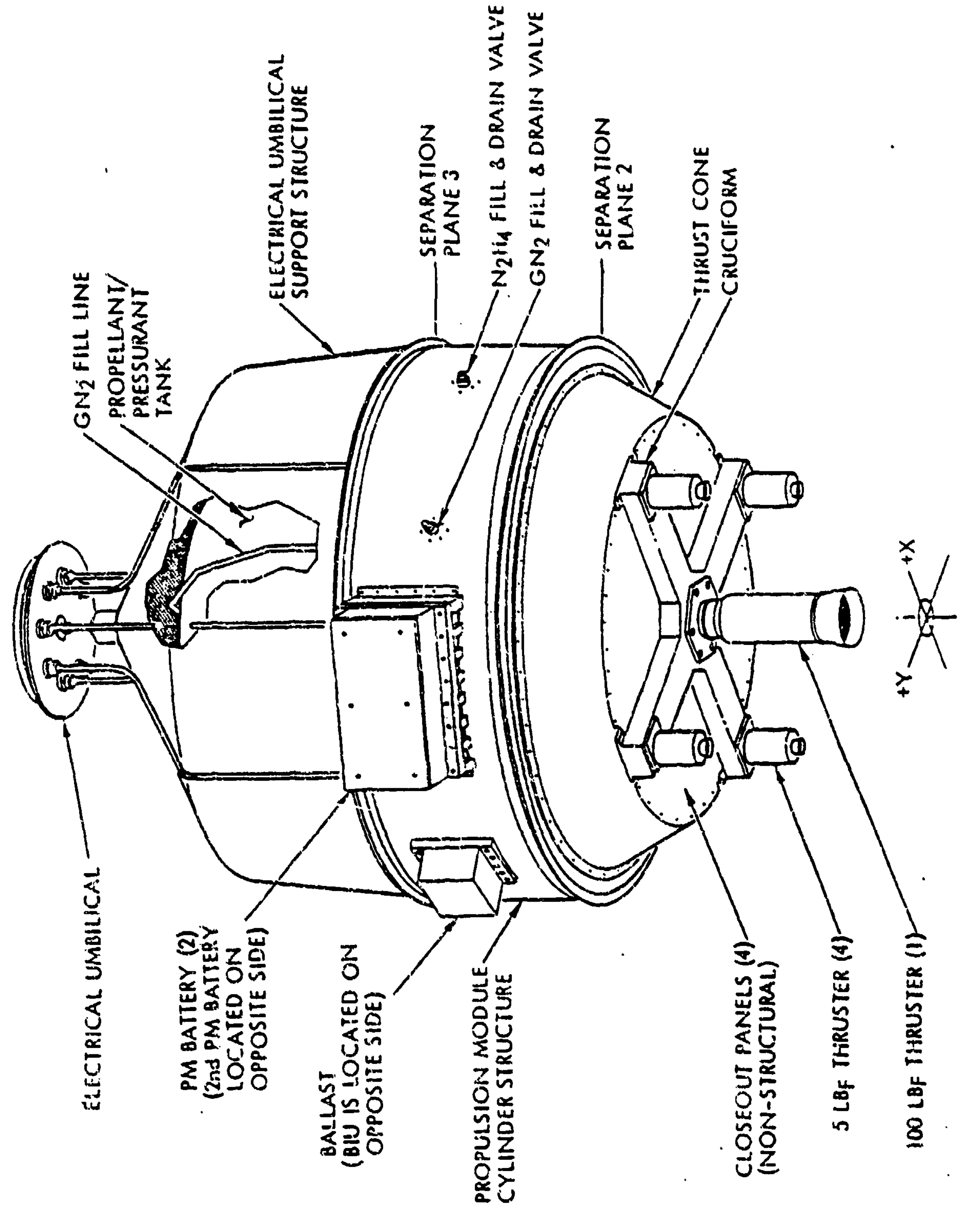


Figure C18-7 Propulsion Module Perspective

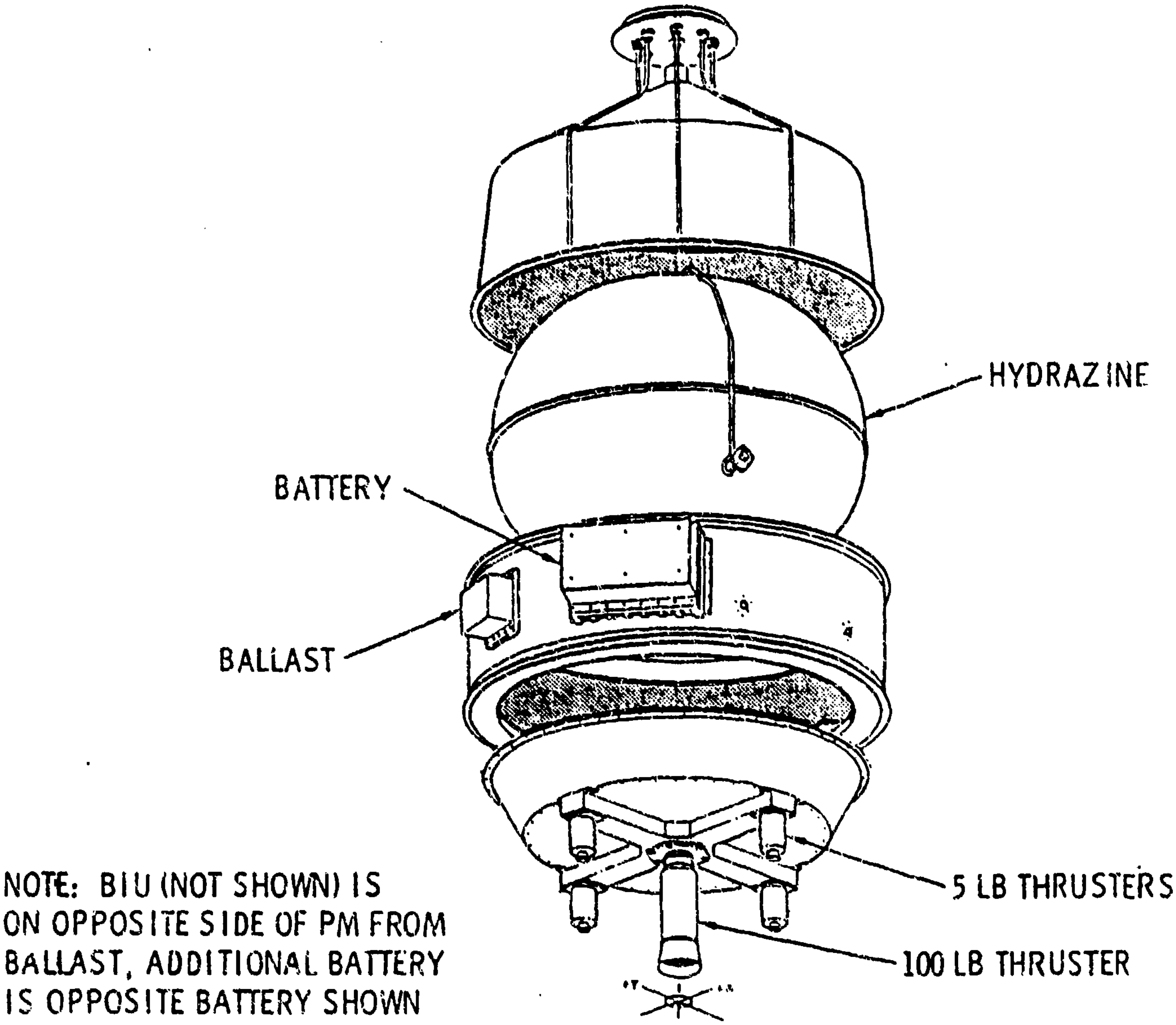


Figure C18-8 Propulsion Module Component Locations

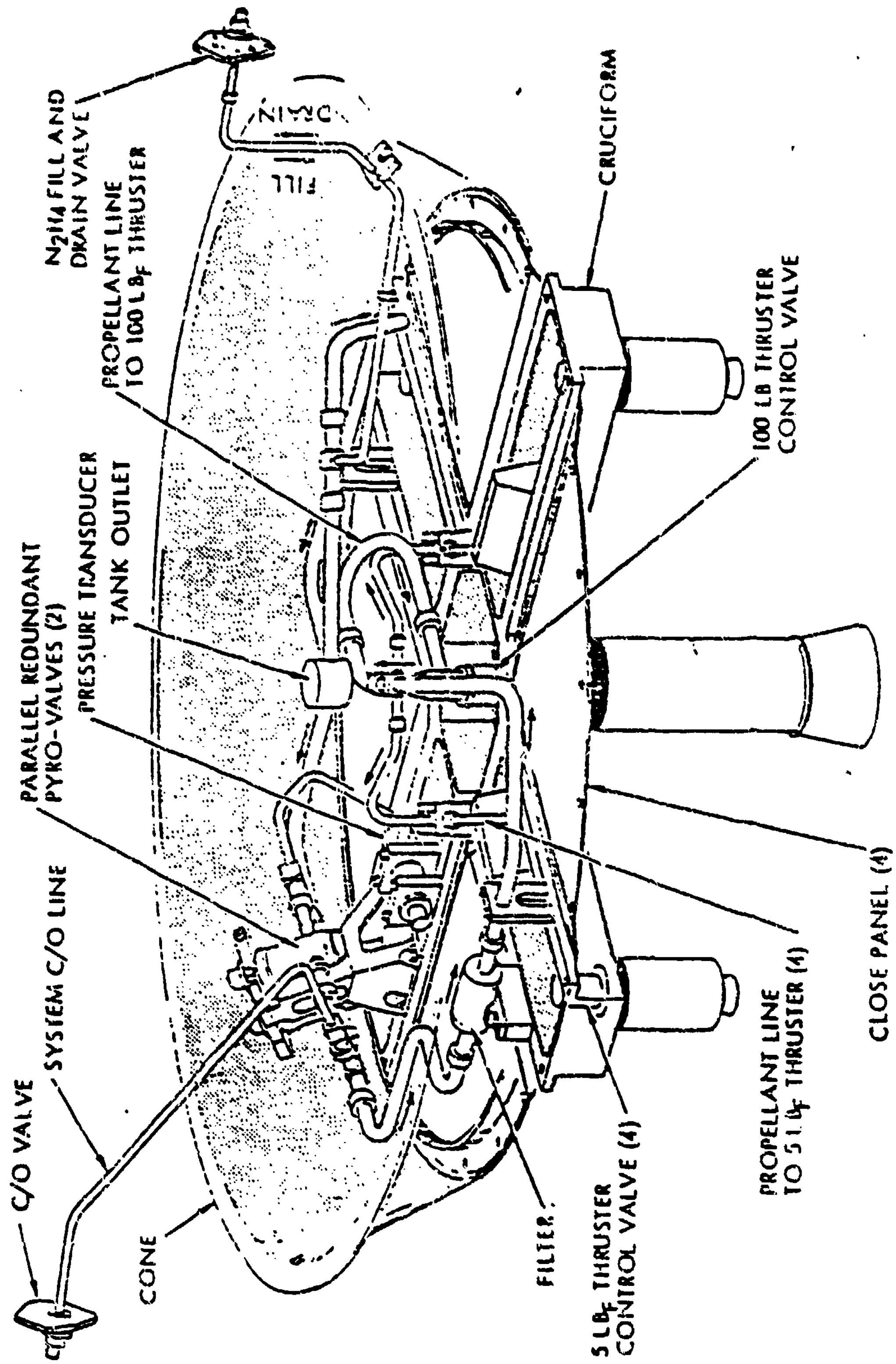


Figure C18-9 Propulsion Module Internal Perspective Showing Fluid Systems Components and Plumbing

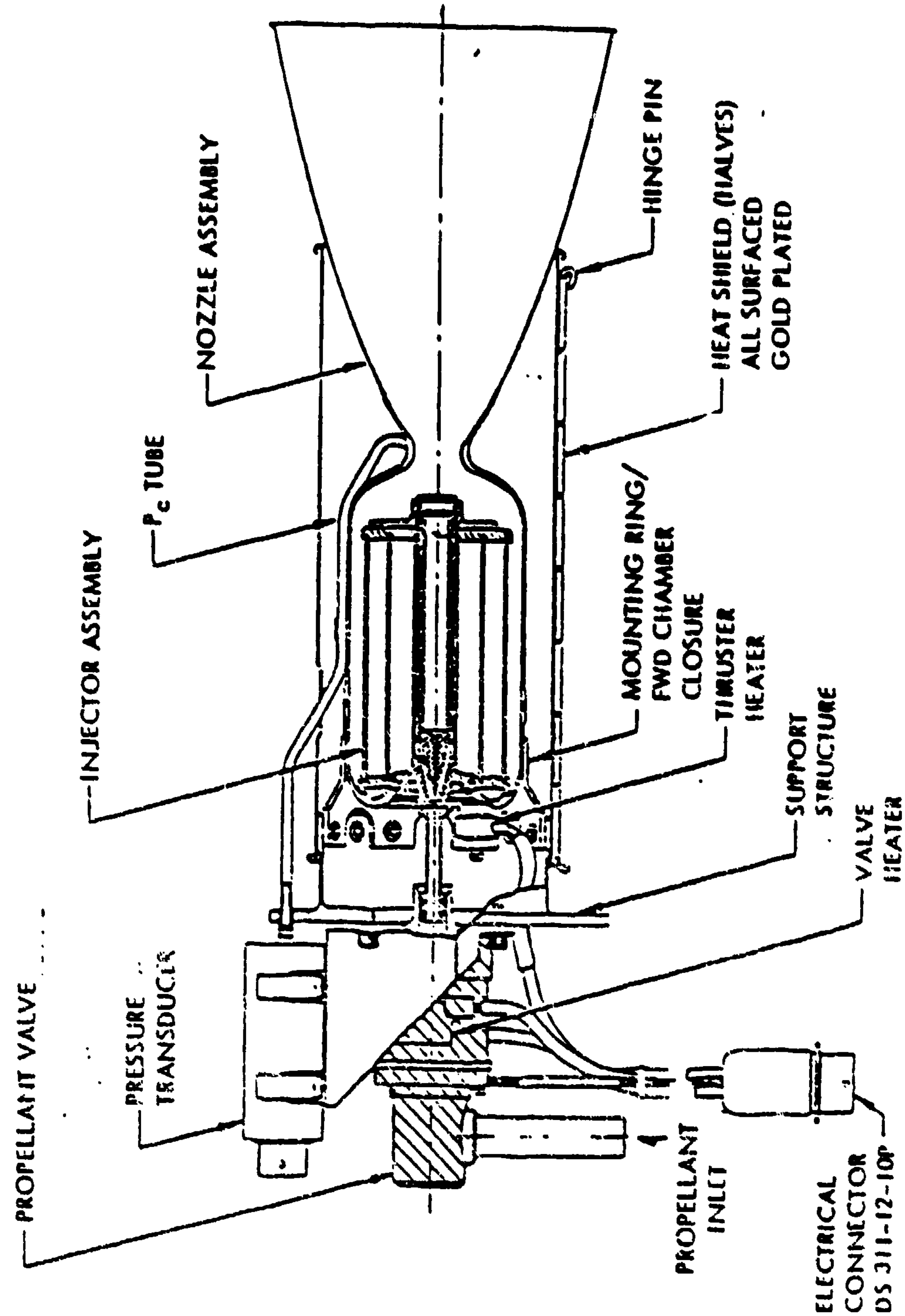


Figure C18-10 Rocket Engine Assembly - MJS 100 LBf

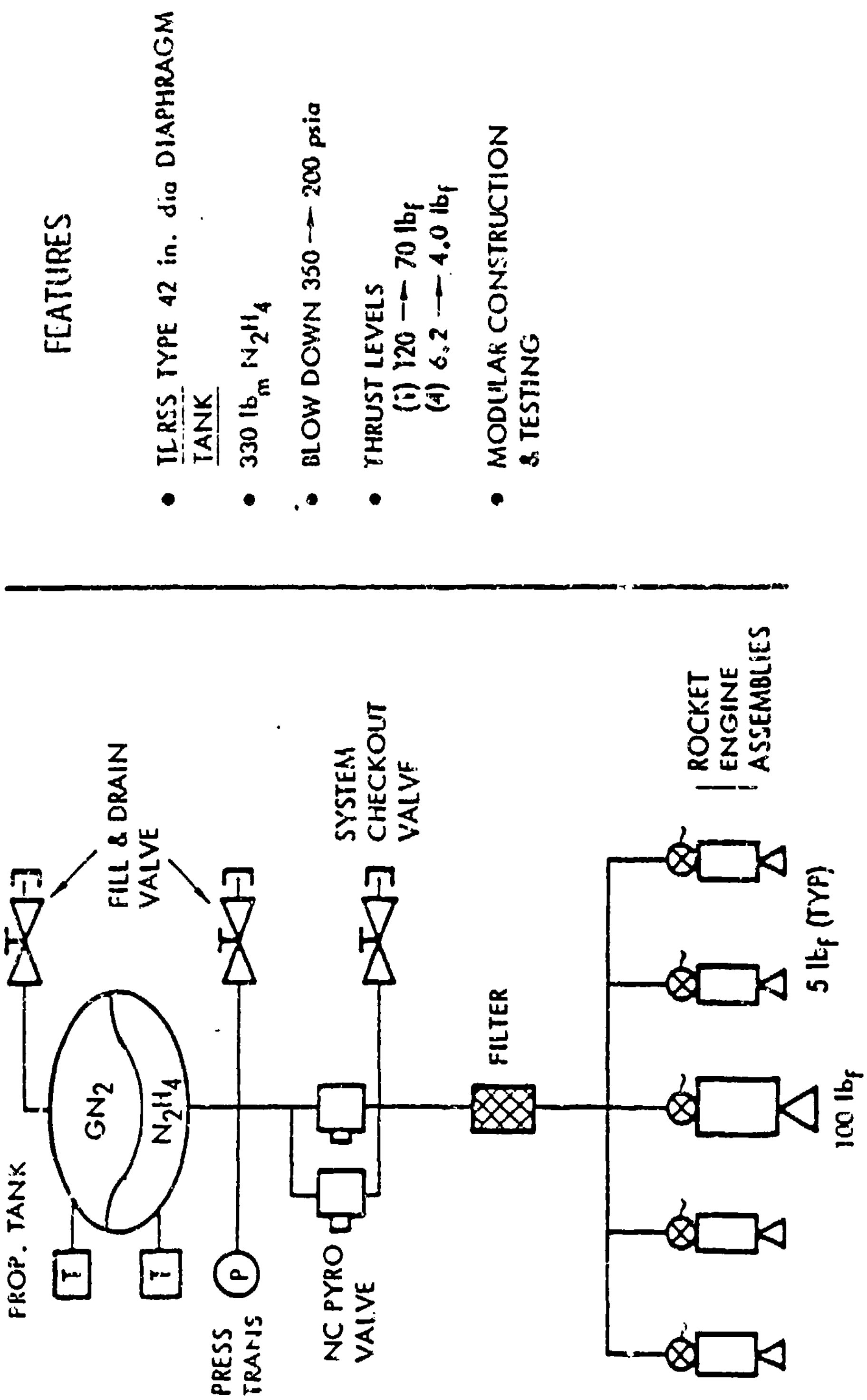


Figure C18-11 Hydrazine Propulsion Module
(Reaction Control System)

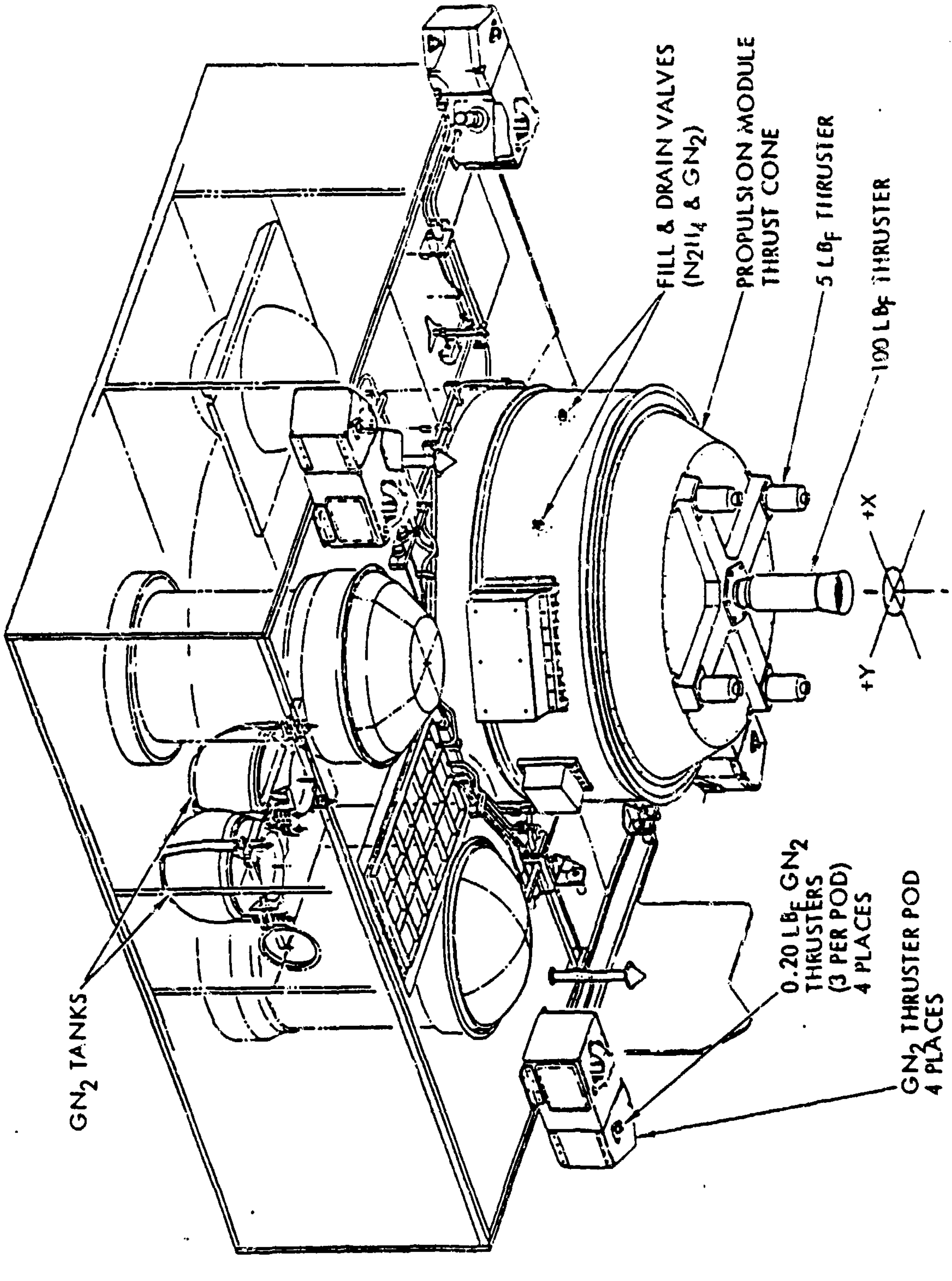


Figure C18-12 P80-1 Propulsion Module and GN_2 Reaction Control System Perspective

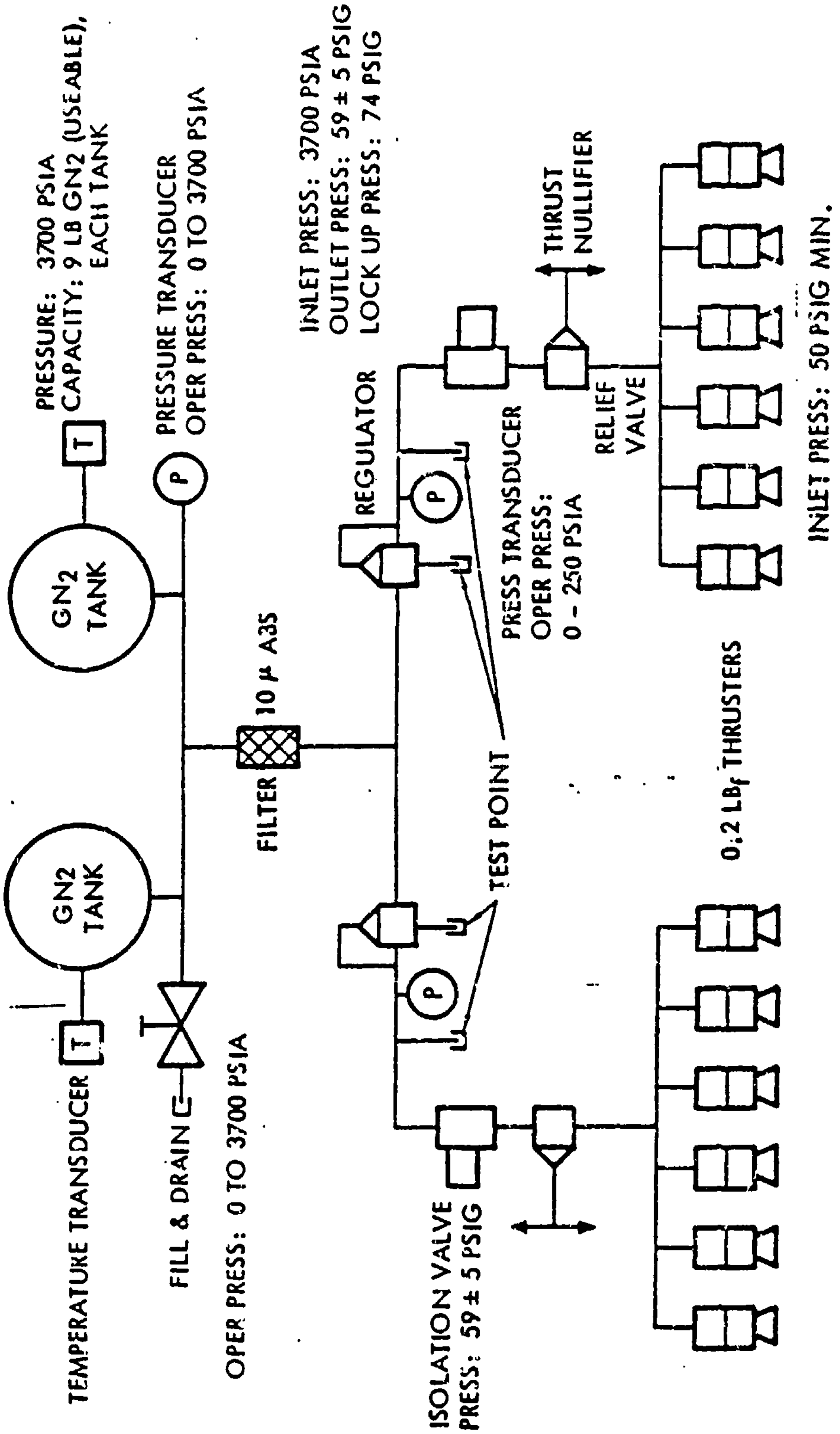


Figure C18-13 Cold Gas System Schematic

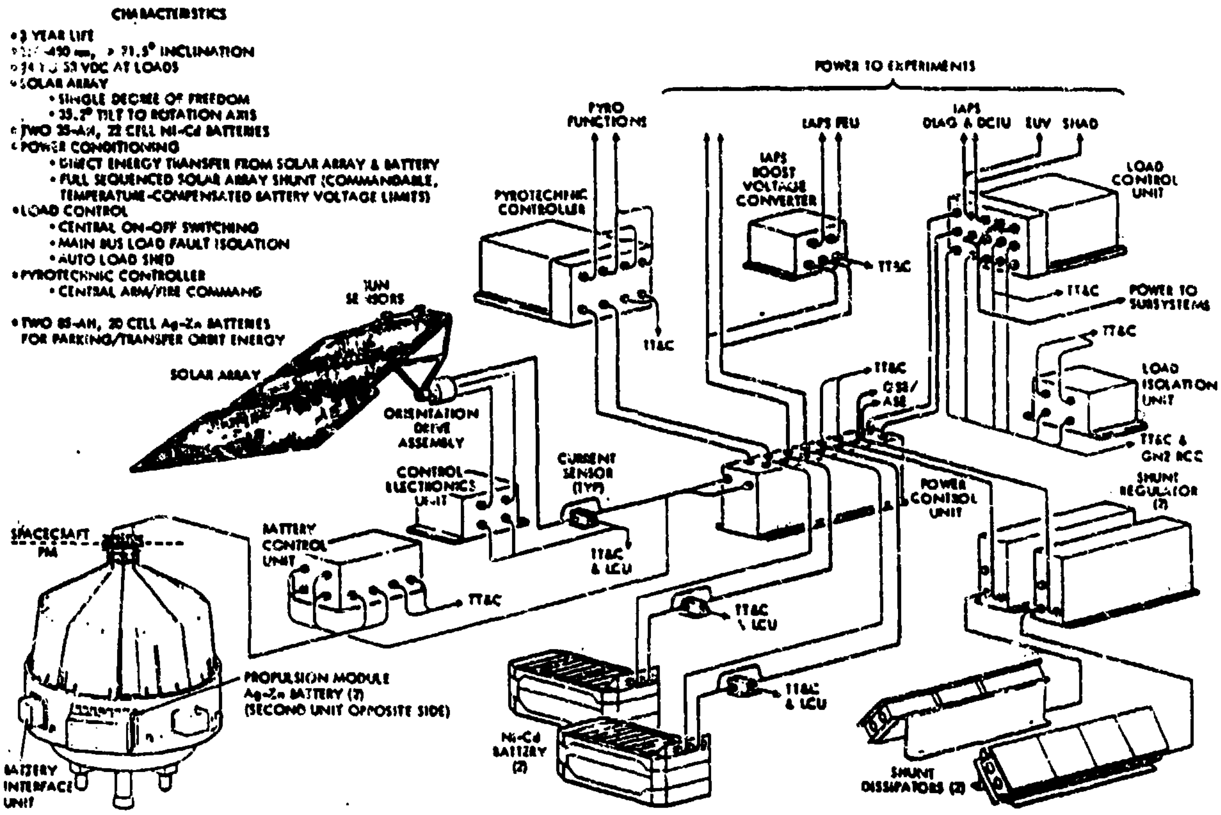


Figure C18-14 P80-1 Electrical Power Subsystem

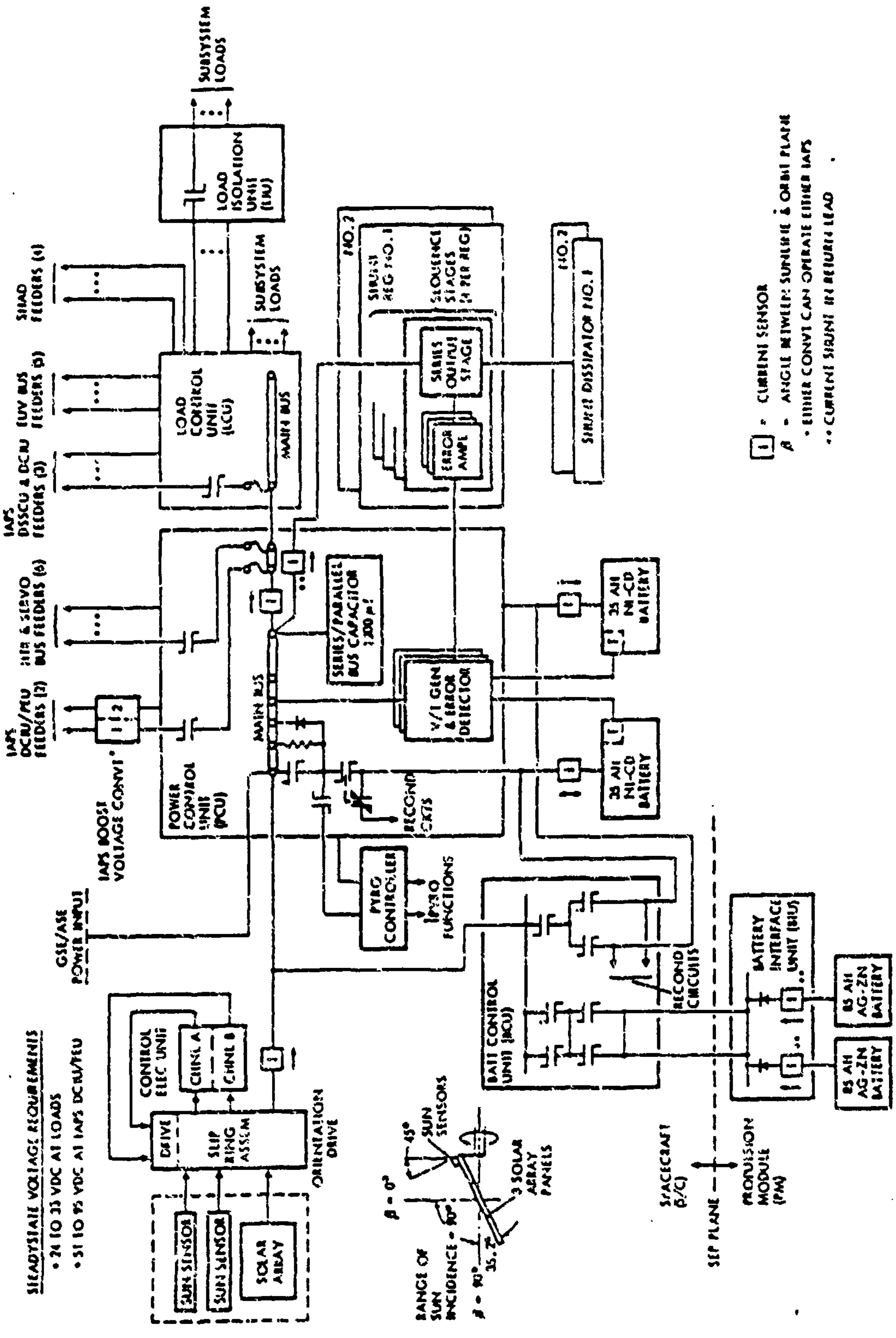


Figure C18-15 EPS Block Diagram

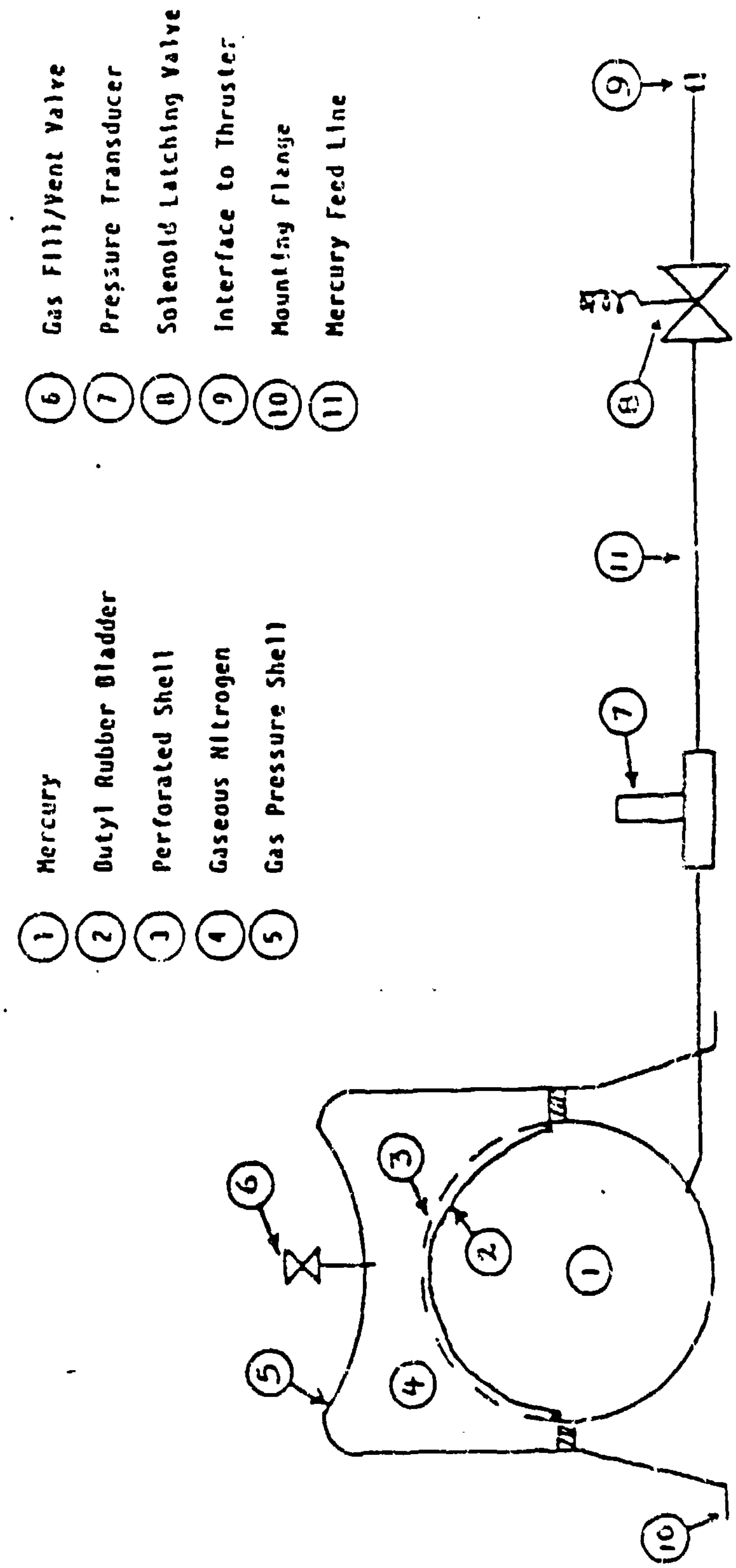


Figure C18-16 Schematic Diagram, IAPS Mercury Feed System

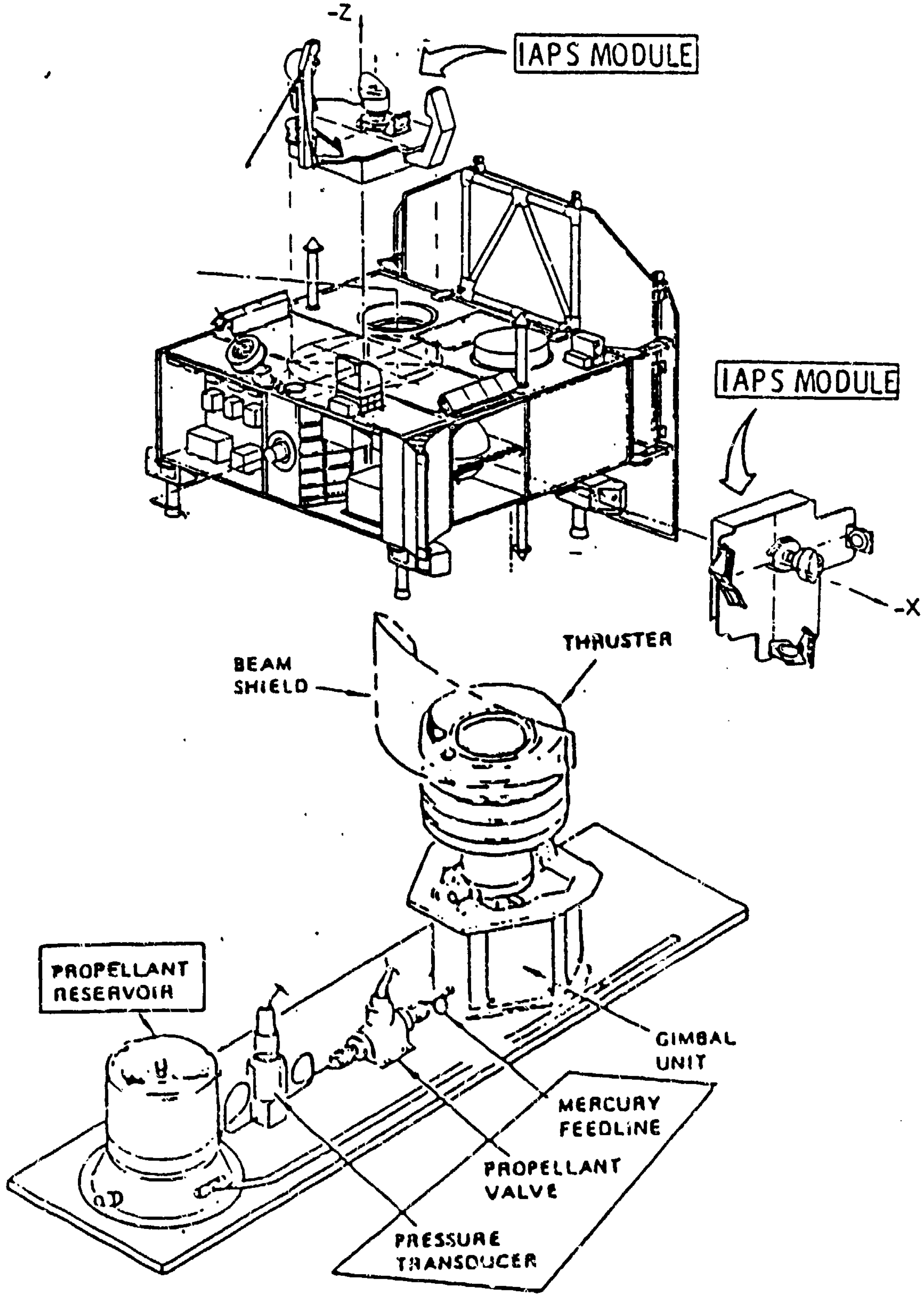


Figure C18-17 IAPS System

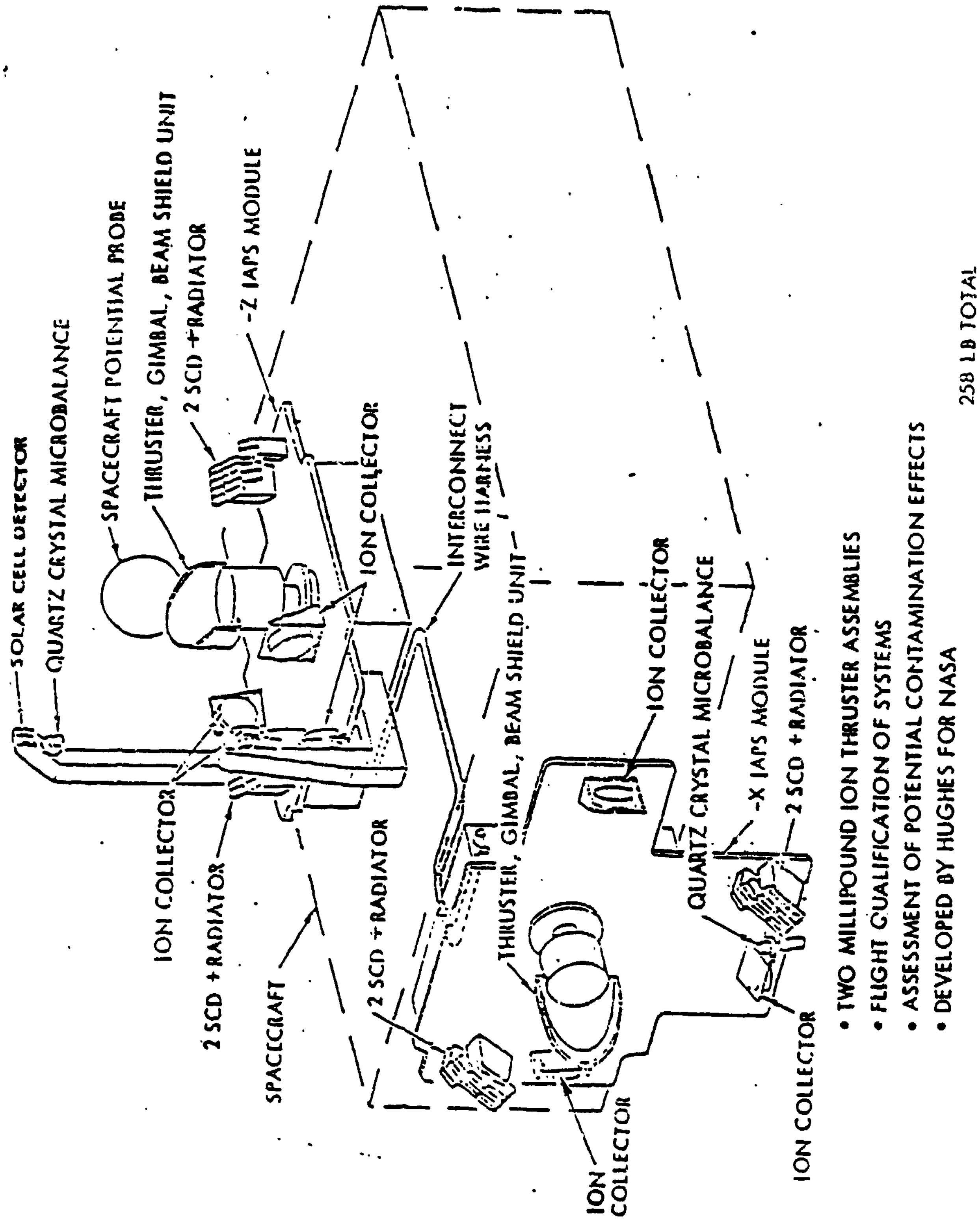


Figure C18-18 ION Auxiliary Propulsion System

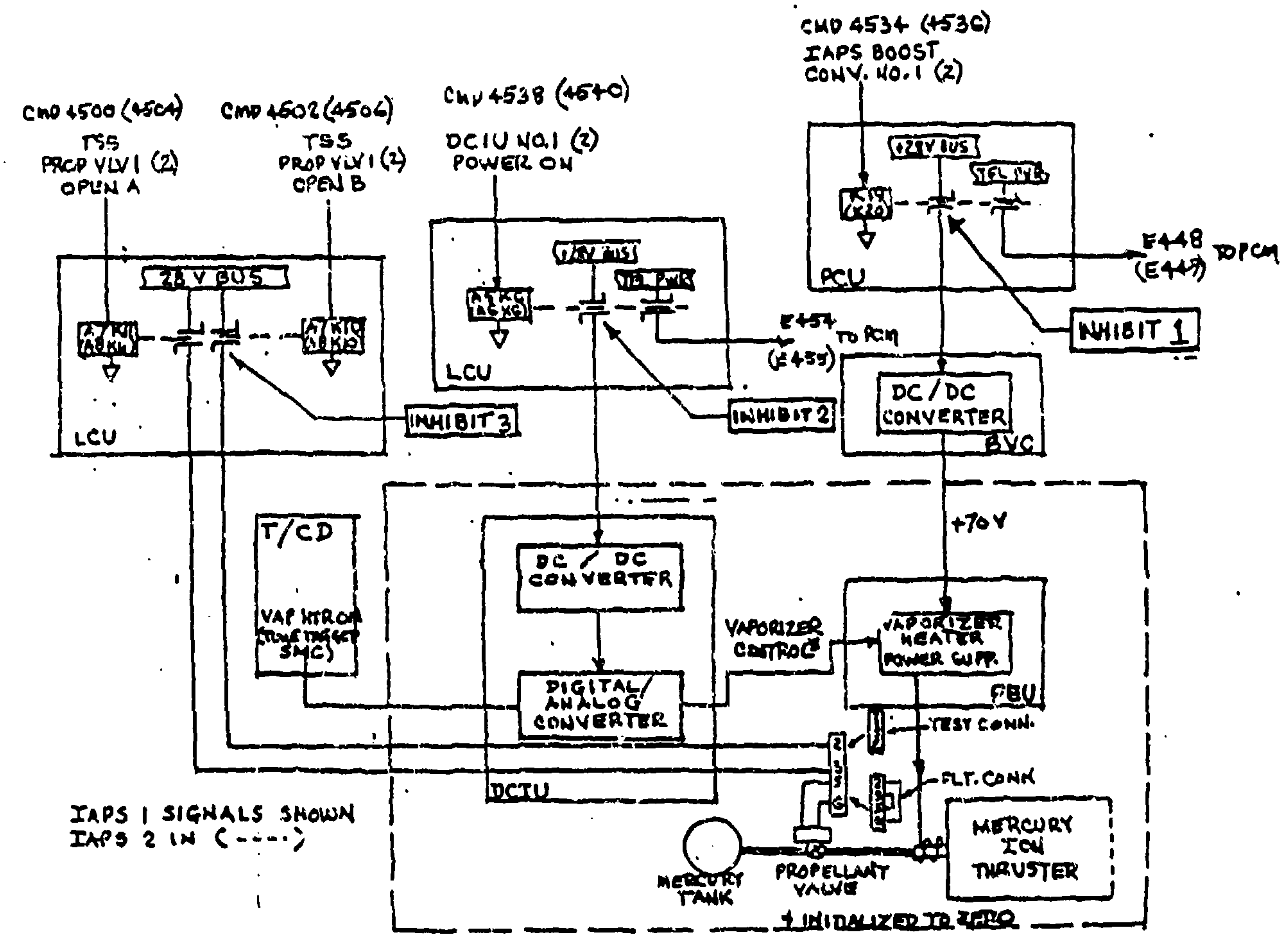


Figure C18-19 IAPS ION Thruster Propellant Flow Control and Inhibits

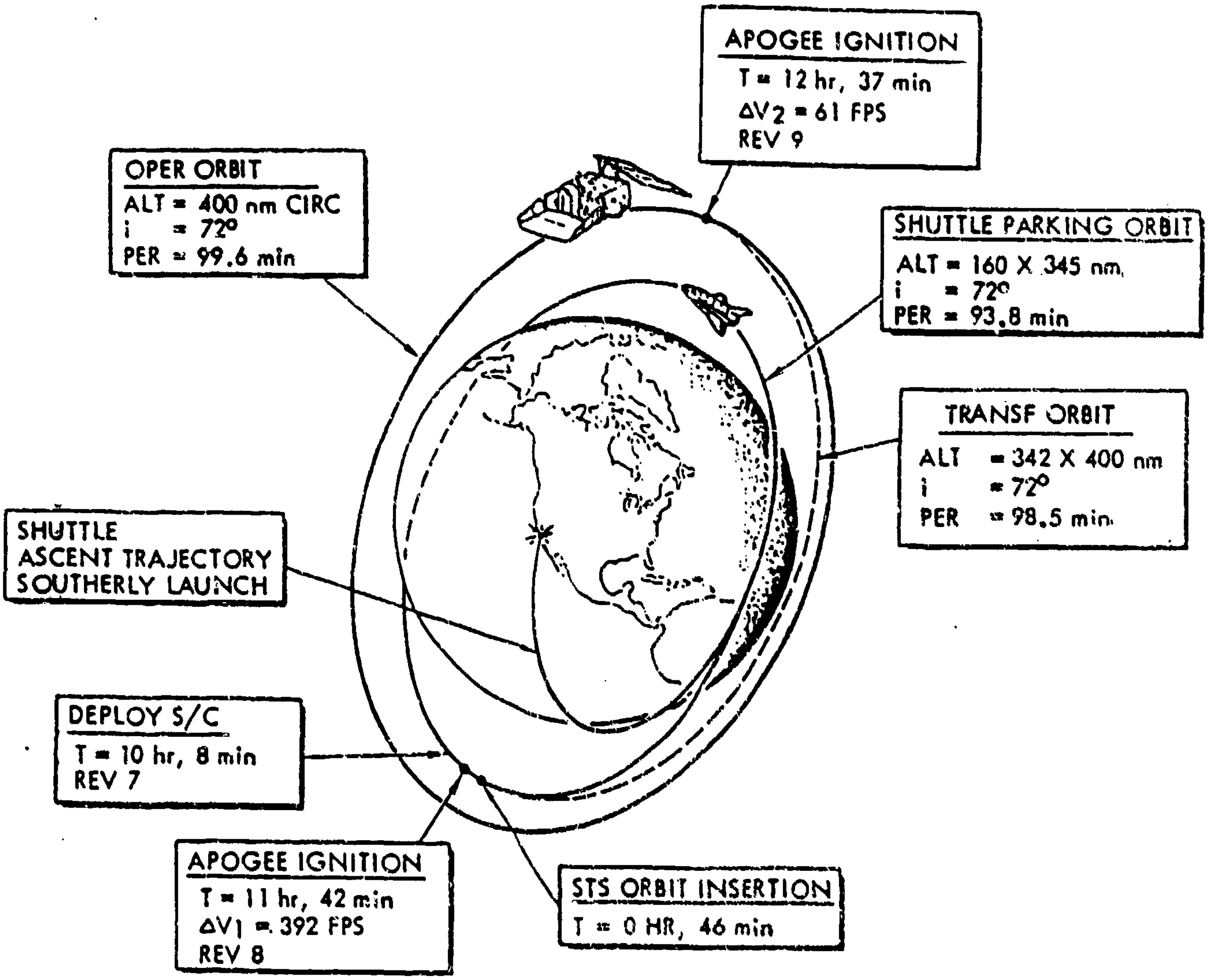


Figure C18-20 Mission - Overview

Appendix C19
Galileo/Ulysses

APPENDIX C19
GALILEO SPACECRAFT AND ULYSSES SPACECRAFT

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APPENDIX C19
GALILEO SPACECRAFT AND ULYSSES SPACECRAFT

C19.0 Introduction

The following data were extracted from the National STS Program Document "Space Shuttle Data for Planetary Mission Radioisotope Thermoelectric Generator (RTG) Safety Analysis," JSC 08116, Feb 15, 1985, NASA, L.B. Johnson Space Center, Houston, Texas 77058.

The Galileo and Ulysses spacecraft, which are scheduled to be flown on the Space Transportation System, carry Radioisotope Thermoelectric Generators (RTGs) to provide electrical power for their respective planetary missions. RTGs pose a potentially hazardous condition in the event of a catastrophic explosion of the vehicle and its payload (Centaur) during applicable phases of the mission.

The mission goal is to obtain more detailed scientific data on Jupiter and its satellites. To obtain such data, a Jovian Orbiter and a Jovian atmospheric probe will be used, both being parts of the Galileo spacecraft. The overall scientific objectives of the Galileo mission are to conduct comprehensive investigations of the Jupiter planetary system by making in situ and remote measurements of the planet, its environment, and its satellites. Close-up study of the planet and its principal satellites will greatly extend our knowledge of the role of the Jovian system in the complex and analogous relationships existing between the Sun and its planetary system.

C19.1 GALILEO SPACECRAFT GENERAL DESCRIPTION

The Galileo spacecraft (S/C) consists of a Jovian Orbiter and an atmospheric probe and weighs over 2500 kg; it is mounted atop a Centaur upper stage. Figure C19-1 shows the Galileo Shuttle launch configuration. The gross weight of the vehicle, with a maximum 65,000-lb payload, is nearly 4.5 million lbs. Figure C19-2 depicts the S/C in its stowed configuration; Figure C19-3 depicts the S/C in its cruise configuration. Separation of the spacecraft (S/C) from the Centaur occurs shortly after the Centaur burn, which impels the S/C from Earth orbit toward Jupiter. The S/C is controlled by the Orbiter during the cruise to Jupiter. During this cruise, both status telemetry and occasional probe checkouts are powered by the Orbiter. The probe and Orbiter remain an integral unit until about five months before arrival at Jupiter, when the probe begins to operate on its own internal power just prior to separating from the Orbiter.

C19.2 ULYSSES GENERAL DESCRIPTION

The Ulysses (ULS) mission is a joint effort by European Space Agency (ESA) and NASA. The European contribution to the ULS consists of the provision and operation of the spacecraft and about half of the experiments. NASA will be responsible for providing the launch by the STS/Centaur, the remaining experiments along with the RTG, and mission support using the

DSN. The primary scientific objectives of the ULS are to investigate, as a function of solar latitude, the properties of the solar corona, the solar wind, the structure of the sun-wind interaction, the heliospheric magnetic field, solar and non-solar cosmic rays, solar radio bursts and plasma waves, and the interstellar/interplanetary neutral gas and dust. Secondary science objectives of the mission include interplanetary physics during the initial Earth-Jupiter or ecliptic phase and measurements of the Jovian magnetosphere during the Jupiter flyby phase.

Although the original ULS plans included both a European and a NASA spacecraft, subsequent cancellation of the latter has left the European entry as the sole surviving spacecraft. This spacecraft is illustrated in Figure C19-4.

Prominent characteristics of the spin-stabilized spacecraft are the large diameter (1.65 m) parabolic High Gain Antenna (HGA) on top of the spacecraft, the RTG, the 5.5-meter radial boom that provides an appropriate environment in terms of electromagnetic cleanliness for certain experiments, and the 7.5-meter-long axial boom. The body of the spacecraft contains all of the subsystems and science instruments with the exception of those experiment sensors mounted on the 5.5-meter boom. All internally mounted units are carried on a honeycomb center panel that also supports the hydrazine fuel tank. In general, the experiments that are more sensitive to nuclear radiation are mounted on the portion of the spacecraft farthest from the RTG, while the less sensitive subsystems are nearer to it.

Mass properties and balance of the spacecraft have been a driver in the design. Requirements on the launch configuration and for the HGA pointing are met. The spin axis of the launch configuration is the geometric centerline Z-axis of the spacecraft. The electrical axis of the HGA will be aligned with the theoretical orbital spin axis. The tilt angle of the principal axis with maximum inertia has been minimized (0.10 degrees half cone). In order to ensure quasi-omnidirectional coverage for at least near-Earth operations, two low-gain antennae (LGA) have been chosen to complement the HGA.

C19.3 GALILEO SYSTEMS DESCRIPTIONS, HAZARDOUS MATERIALS, SCHEMATICS

The design of the Galileo Orbiter is based on experience gained with several previous satellite and planetary spacecraft programs. Table C19-1 presents the Orbiter subsystem design base.

Table C19-1 Orbiter Subsystem Design Base

Subsystem	Design Base
Engineering	
Structures	Mariner/Voyager/New
Antennas	Voyager/TDRSS
Radio Frequency	Voyager
Modulation/Demodulation	Voyager
Power/Pyrotechnics	Voyager/Modified Viking
Control and Data	New/Voyager
Attitude Control	New/Modified Voyager
Propulsion	Symphonie
Temperature Control	Voyager
Data Storage	IRAS
Mechanical Devices	New/Voyager
Science	
Payload	Voyager/Mariner Venus Mercury/Helio/New

C19.3.1 Galileo Description

The Galileo Orbiter (Fig. C19-3) is a dual-spin spacecraft. Part of the spacecraft will be three-axis stabilized to provide a steady base for the remote-sensing instruments. These instruments must be precisely pointed. The despun section carries its related electronics. The main portion of the Orbiter will spin at three revolutions per minute to provide stability and to allow its instruments to continuously sweep the sky to make their measurements. The spun section contains both the high- and low-gain antennas, the RPM for all propulsive and attitude maneuvers, the nuclear power sources, most of the electronics, most of the command and data equipment, and the fields and particles science instruments.

The RPM is being built by Messerschmitt-Bolkow-Blohm GmbH near Munich, Federal Republic of Germany, and is managed by Deutsche Forschungsund-Versuchsanstalt fur Luft-und Raumfahrt e.V (DFVLR), a German research agency under the aegis of the German Ministry for Research and Technology. The RPM has one 400-newton engine and two clusters of 10-newton thrusters that are used for attitude control and the smaller propulsive maneuvers. Because of minimum burn size and total wetted lifetime constraints, the 400-newton engine will be used only for the deflection maneuver after probe release, orbit insertion, and the perijove raise maneuver. Interplanetary maneuvers will be performed using the 10-newton thrusters.

Galileo will use a 4.8-meter-diameter (16-foot) furlable antenna to communicate with Earth. This antenna is similar to the one developed for NASA's Tracking and Data Relay Satellites. A small (1-meter) relay antenna will ride on the despun portion of the Orbiter to receive data from the probe for relay to Earth.

Since Jupiter is too far from the Sun for solar cells to provide enough electrical power, Galileo will use radioisotope thermoelectric generators (RTGs) similar to those flown on the two Voyagers. These nuclear power sources are developed by General Electric Company for the U.S. Department of Energy.

The despun section of the Orbiter carries four remote-sensing instruments: a solid-state imaging system, a near-infrared mapping spectrometer, a photo-polarimeter radiometer, and an ultraviolet spectrometer. They are mounted on a scan platform attached to the despun section. Five instruments--a magnetometer, instruments to study plasma, energetic particles, plasma waves, and a dust detector--ride the spinning section of the Orbiter, most of the instruments mounted on a single boom. In addition, the spacecraft radio is used to perform celestial mechanics and radio wave propagation experiments. The purpose of each of the instruments mentioned is given in Table C19-2. Seven of the nine instruments are extremely sensitive to contamination and, therefore, have protective covers that are latched during launch. Six of these instruments use bellows actuators for unlatching (the star scanner

Table C19-2 Orbiter Science Investigations

Instrument Name	Objectives
Remote Sensing Instruments	
Solid Stage Imaging (SSI)	Image Jupiter and its satellites for studies of atmospheric dynamics and physical geology.
Near Infrared Mapping Spectrometer (NIMS)	Study mineralogy of satellite surfaces and morphology and structure of Jovian clouds.
Photopolarimeter Radiometer (PPR)	Study photometric and thermal properties of satellite surfaces and cloud and haze properties in Jovian atmosphere.
Ultraviolet Spectrometer (UVS)	Study composition and structure of high neutral atmosphere of Jupiter and Galilean satellites.
Fields and Particles Instruments	
Magnetometer (MAG)	Study magnetic field of Jupiter and search for magnetic fields associated with the satellites.
Plasma (PLS)	Study Jovian plasma.
Plasma Wave (PWS)	Study time-varying electric and magnetic waves in the Jovian plasma.
Energetic Particles Detector (EPD)	Measure detailed energy and angular distribution of protons, electrons, and ions.
Dust Detector (DDS)	Study physical and dynamical properties of small dust particles in the Jovian environment.
Radio Science (RSS)	
Celestial Mechanics	Study the space environment and gravity fields of Jupiter and its satellites.
Radio Propagation	Study structure of atmospheres of Jupiter and satellites by use of radio signals from Orbiter and probe.

uses the same concept), while the seventh uses a wax pellet. Although most covers give little concern in case of premature release, they are all designed to be two-fault tolerant. Single-watt radioisotope heater units (RHU), used throughout the spacecraft (105 total), also are used in several of the instruments (Fig. C19-5). The energetic particles detector also contains several small radioisotope calibration sources. The only other potential hazard that the instruments might impose results from the use of high voltages in four of the instruments. These instruments are not operated during the launch phase.

C19.3.2 Orbiter Communications Subsystem

Commands are received via an S-Band transponder, and telemetry is returned to Earth using either a 10/30-watt S-Band Traveling Wave Tube Amplifier (TWTA), or a 10/22-watt X-Band TWTA. The S-Band system can use either the low-gain antenna or the high-gain antenna, while the X-Band operates only over the high gain antenna. The S-Band transmitter is turned on after separation from the Shuttle orbiter. During the Centaur burn, the high-gain antenna's 4.8-meter reflector is in a folded condition, enveloping the high-gain feed system. The nonexplosive initiators (NEI) and the motor required for release of the reflector are controlled by the Orbiter CDS and are not operated until after the Centaur burn. Both the motor and the NEI are designed to be two-fault tolerant against premature activation.

C19.3.3 Command and Data Subsystem (CDS)

The CDS is an active, redundant microprocessor-based system with a total of 176K words of memory that uses data buses for interaction with other engineering subsystems, the science instruments, and the probe relay receivers. Its functions are uplink command processing, sequence control, fault protection, downlink data collection and formatting, and onboard intercommunications, i.e., movement of data between subsystems.

C19.3.4 Attitude and Articulation Control Subsystem (AACS)

The AACS is the other major Galileo Orbiter subsystem that is computer-controlled and contains redundant 31K, 16-bit words in Random Access Memory (RAM), and 1K in Read-Only Memory (ROM). Its functions are pointing the S/C for Earth communications, controlling trajectory correction maneuvers and pointing for probe release; pointing the instruments mounted on the scan platform; spin rate control of spun section and clocking of despun section containing the scan platform and the probe relay antenna; and thruster firings. The AACS receives commands and program updates from the ground via the CDS and receives commands directly from CDS programs. It uses inputs from gyros, accelerometers, star scanner, Sun acquisition sensor, spin bearing, and scan actuators position encoders for control.

Propulsion control consists of latching isovalve control, drive signals to the 400-N main engine and to the 12 10-N thrusters, and heaters on the two thruster clusters. No pyro valves are controlled by the AACS.

The subsystem operates in five different modes, including the launch mode, with mode selection by CDS command. During the launch mode, all propulsion and control drive signals are inhibited by software, and only monitoring and fault protection functions are active. Transition to the next, or deployment, mode does not occur until about 55 minutes after separation from the Shuttle. The primary function here is the RPM vent sequence, during which CDS commands AACS to open all latching isolation valves and thruster valves in order to bleed the RPM fuel and oxidizer lines.

C19.3.5 Power Subsystem

The Galileo Orbiter power source is a two-component Si-Ge 580-watt general purpose heat source (GPHS) RTG that provides 30 Vdc to all subsystems. The two RTGs (290W each), in which heat from the radioisotope fuel (plutonium-238 oxide) is converted to electricity by silicon-germanium thermoelectric converters, are located on the spun portion of the Orbiter. The RTGs are connected in parallel and isolated via diodes. The 30 Vdc is maintained via a shunt regulator, which dissipates excess RTG power in shunt heaters located near the RPM propellant tanks. The Orbiter also uses 2.4 kHz ac generated from the dc bus. During cruise, RTG power is routed to the probe for telemetry and checkout.

The RTGs are installed after the S/C is in the cargo bay, and power is on from then on. An RTG shorting function is implemented automatically in parallel with the Centaur propellant dump in the event of an abort. This function will shunt RTG power out of the S/C and thereby provide extra thermal margin on the propellant tanks during or after an abort. No post-abort spacecraft hazard exists after the Shuttle Orbiter is connected to ground purge of the cargo bay with cooled air. While the RTS is in the cargo bay, water cooling is provided by the STS auxiliary cooling system. These cooling lines are purged just prior to payload release from the cargo bay. They are routed to the S/C via the Centaur, so separation occurs at cargo separation from the Shuttle Orbiter.

The activation of the RTG pressure release devices is controlled by a sequence initiated at the STS-Centaur separation event. The Centaur, using the appropriate timer, also removes the pyro short-and-arm inhibits.

Power to all subsystems is provided via latching relays that are controlled by the CDS. A second set of relays provides additional safeguards for pyro, RF transmitters, HGA deployment motors and probe power on, and instrument cover releases.

A power summary with RTG capability requirements is provided in Table C19-3.

Table C19-3 Orbiter Power Summary, Watts

Mission Event	Power Required*
Launch	409
Jupiter orbit insertion	495
Orbital cruise	496
Jupiter occultation	460
Satellite encounter	473

RTG capability requirements

At Beginning of Mission................580 W, Minimum

At End of Standard Mission
(37,000 hours or 4.2 years)........506 W, Minimum

At End of Backup Mission
(50,000 hours or 6.3 years)........488 W, Minimum

*Per Galileo Quarterly Power Report No. 27, August 15, 1984

C19.3.6 Pyrotechnic Subsystem

Two pyro units (one spun and one despun) provide the energy necessary to fire all S/C electro-explosive devices and nonexplosive initiators. Each unit employs capacitor banks for energy storage and silicon controlled recitifiers (SCRs) for pyro initiation. The capacitor banks are shorted and unarmed in the Shuttle Orbiter. Centaur signals from two of three 45-minute timers initiated by independent STS-Centaur breakwire signals are required to remove the capacitor short and to arm the pyro switching units. Power to the pyro switching units and fire commands are controlled by the CDS, with critical functions requiring two separate commands, one to enable the firing circuit, and one to trigger the SCR. Critical enables for probe release and 400-N motor firing require ground commands to actuate.

NSIs are used for all functions except Radioisotope Thermoelectric Generator Pressure Release Device (RTG PRD) and S-Band/X-Band Antenna (SXA) unlatch, which are NEIs and subsystem cover deploy functions, which use bellows actuators.

C19.3.7 Retropropulsion Module (RPM)

The RPM provides all of the propulsive maneuvers of the spacecraft occurring after Centaur separation. The RPM is a mechanically separable self-contained module integrated as a load carrying part of the spacecraft spin section.

The RPM is a bipropellant pressure-fed subsystem using the hypergolic propellant combination of MMH and a mixture of N_2O_4 with a small fraction of nitric oxide added. Its four propellant tanks have a maximum usable capacity of 935 kilograms. The unified feed system supplies a central 400-N engine for the large deterministic maneuvers and twelve 10-N thrusters for trajectory and attitude control.

The RPM consists of the following main elements:

a. Structure
   1. Central structure (S/C load-carrying part)
   2. Two booms/cluster housings (for mounting of six 10-N thrusters each)

b. Four propellant tanks

c. Two pressurant tanks

d. One 400-N engine assembly (mounted on S/C-provided spin bearing assembly (mounted on S/C-provided spin bearing assembly with an associated support structure)

e. Twelve 10-N thrusters

f. Pressurization and feed system
   1. Two pressurant control/propellant isolation assemblies (PCAs/PIAs) on two opposite equipment panels (part of central structure)
   2. Tubing

g. Thermal control
   1. Boom/cluster thermal blankets and electrical heaters
   2. 400-N engine thermal shield and electrical heaters

h. Electrical cabling
   1. Pyro cabling
   2. Main harness (for valve control, electrical heaters, and housekeeping)

C19.3.8 Galileo Jovian Atmospheric Entry Probe

The baseline Jovian probe, is a stage vented system which consists of a deceleration module and descent module. The deceleration module consists primarily of structure and heat shields, while the descent module contains the science instruments, probe electronics and power source and is vented to the Jovian atmosphere. Hughes Aircraft Company was selected as the probe source.

The general appearance and dimensions of the probe are shown in Figure C19-6. Figure C19-7 displays a cross-section of the probe descent module to highlight the features of the design as it has evolved to date. The two-shelf design of the vented descent module has the heavy lithium battery pack on the front of the forward shelf to provide both a stable location of the center-of-gravity of the probe and shielding of the instruments located between the two shelves. The communication equipment is located on the rear of the aft shelf. The current estimate of the probe spacecraft mass by major subsystems is shown in Table C19-4.

A parachute is used to separate (stage) the descent module from the deceleration module and to control the probe descent rate in order to provide ample time for science measurements (see Fig. C19-8). In-situ science measurements will be made before and during high speed entry and while descending through the atmosphere to a depth corresponding to at least 10 bars of pressure.

Environmental protection will be provided as necessary to assure the functioning of all required systems until a pressure level of at least 10 bars is reached. A battery will provide probe power during autonomous operation; the Orbiter will supply probe power during interplanetary cruise. The probe science data will be transmitted to the Orbiter for subsequent transmission to Earth.

C19.3.9 Probe Structure Subsystem

The descent module structure consists of a mainshelf of aluminum honeycomb 27.5 inches in diameter and 2.0 inches thick. The aft compartment is circular with titanium honeycomb panels 25 inches in diameter, supported by three titanium bipods. The areofairing system is comprised of a titanium forward dome and six midfairing panels. The forward dome is attached to the downturned ring flange of the shelf interface ring. There are eight penetrations for scientific instrumentation, five sensors, and three atmospheric inlets. For thermal protection, multilayered aluminized Kapton insulation blankets are used.

The deceleration module consists of the aeroshell/heatshield, aft cover, and parachute. The aeroshell is a 45° blunt cone with a 49.8-inch diameter and an 8.75-inch nose radius. A carbon phenolic heatshield is bonded to the aluminum ring-stiffened monocoque structure. An 8.1-foot-diameter conical ribbon main parachute made of daceon is deployed by a 2.4 foot diameter pilot chute, which is mortar ejected. The aft cover provides thermal protection with a phenolic nylon shield.

Table C19-4 Probe Spacecraft Mass Estimate*

System/Subsystem/Item	Mass, kg			
Probe System				
Descent Module				
Science Instruments	29			
Engineering Subsystems	90			
SubTotal		119		
Deceleration Module		218		
Sub-Total			337	
Contingency			2	
Total Probe Launch and Entry Mass				339
Radio Relay Hardware (Orbiter-Mounted)		21		
Probe-to-Orbiter Adapter (Orbiter-Mounted)		7		
SubTotal			28	
Contingency			2	
Total RRH + Probe Adapter Allocation				30

*Per Galileo Quarterly Mass Report 26, August 15, 1984

The adapter assembly is made up of the hardware that provides the mechanical interface between the probe and the Orbiter. The primary elements are three titanium bipods, each having a female shear cone on the probe interface side and two clevises on the Orbiter side. Each female shear cone interfaces with a carbon phenolic cone on the probe, and the cones are held together by an explosive nut and bolt. The clevises on the Orbiter side interface with tangs on the Orbiter structure, and a bolt is installed in each fitting to complete the tie.

Each bipod is designed with provisions for accommodating the explosive nut and separation springs. Furthermore, two of the bipods also hold the redundant separation switches that indicate when the probe is separated. The bipod nearest the main umbilical harness also has provisions for supporting the cable cutter.

C19.3.10 Probe Communications Subsystem

The communications system is a redundant system of two solid-state power amplifiers, each providing 23-watt modulated outputs of 1387.0 and 1387.1 MHz. The RF carriers are generated by two exciter drivers, a stable oscillator, and a temperature-compensated crystal oscillator. The two channels, which transmit nearly redundant telemetry information from the data command processor unit, are radiated from a crossed dipole cup antenna. The signals are each frequency multiplied to RF and biphase modulated by the exciters amplified by the transmitter, which in turn drives the antenna. When the probe is attached to the Orbiter, it is checked out via a hardwire from the exciters to the receiver input. While the probe is attached to the Orbiter, the transmitter is inhibited from turning on.

C19.3.11 Probe Pyrotechnic Subsystem

Eight pyrotechnic functions are controlled by the probe through the pyro control unit (PCU). This unit contains three arm relays that provide series-parallel redundancy for connecting the main battery (Li/SO_2) to the pyro bus, which fires the thermal battery and the neutral mass spectrometer squibs. Each thermal battery squib is fired from a series-redundant pair of current-limiting squib drivers. The firing circuit for the neutral mass spectrometer functions is within the instrument, and the PCU merely provides a voltage source. Once the thermal batteries are initiated, the PCU has ten seconds to fire the squibs for the following functions in the order listed:

1. Deploy pilot parachute (mortar)

2. Separate cable to aft cover (cable cutter)

3. Release aft cover (separation nuts)

4. Separate cable to aeroshell (cable cutter)

5. Release aeroshell (separation nuts)

6. Deploy nephelometer optical unit sensor (pin puller)

The squib firing circuits in this portion of the PCU are similar. Each squib is fired through two series-redundant current-limiting squib drivers. The thermal battery outputs are not cross-strapped, preventing any short from pulling down both thermal battery modules. Hence, this is a block-redundant squib firing system for all staging events and nephelometer sensor deployment.

C19.3.12 Probe Electrical Power Subsystem

The Galileo probe electrical power subsystem uses a lithium-sulfur dioxide (Li/SO_2) battery as its energy source. The battery is comprised of three identical modules, each module containing 13 series-connected Li/SO_2 D-size cells and bypass diodes for each cell. Should any cell develop an open, the bypass diode will permit the continued use of the module, albeit at a reduced voltage. The bypass diode prevents reverse current flow in the cells that discharge first in any given string. The batteries are also relay isolated in series with the diodes. The battery provides nominal 33.8 volts to 38.05 volts during the mission.

The Galileo probe thermal battery is a primary battery comprised of two 14-cell stacks wired in parallel. It is a molten-salt electrochemical device using calcium/calcium chromate thermal cells. The thermal battery is a completely inert device (no voltage potential) until the internally contained squibs are initiated. Firing one or both squibs will immediately ignite the internal heat paper (zirconium-barium chromate) and, within one second, the battery will be at full potential (approximately 39 volts). The initiators are 1-amp/1-watt no-fire, 3-amp all-fire squibs qualified to military specification MIL-I-23659C.

C19.3.13 Safety Design Features

The most significant of the Galileo design features implemented to meet the STS safety requirements are described in the following subsections. Specific safety features of the probe are not emphasized except with regard to the probe interface with the Orbiter.

C19.3.13.1 Electrical Inhibits - One of the key spacecraft design features driven by the STS safety requirements is the two-fault tolerant-electrical inhibit scheme. The purpose of the inhibit scheme is to prevent premature activation of spacecraft events potentially catastrophic to the STS. It is implemented by placing three independent electrically controlled inhibits in series between the power source and the spacecraft event in question.

The safety inhibits are implemented in the following four areas of the spacecraft design:

a. Spun pyrotechnic used to unlatch the magnetometer boom, HGA, and science instrument covers and to actuate pyro valves for RPM pressurization.

b. Despun pyrotechnic used to unlatch RTG and science booms, scan platform, despun electronics, and science instrument covers and to fire the spacecraft-Centaur separation and the probe release devices.

c. Probe power used to enable firing of the probe pyrotechnic devices (including the probe parachute).

d. Protected power loads consisting of the RF S-Band exciters, the HGA deployment motors, and the Energetic Particle Detector (EPD).

Figure C19-9 shows a functional block diagram of the two-fault tolerant safety inhibits and the related power circuits. Figure C19-10 shows an overview of the safety inhibit relays and their interfaces with Centaur.

The purpose of the electrical inhibits for the spun and despun pyrotechnics is to protect against premature actuation of spacecraft pyro events, e.g., boom deployment, RPM pressurization, cover release, etc. The firing energy for all spacecraft-commanded pyrotechnic functions is provided by two pyro switching units (PSU), one on the spun and one on the despun section of the spacecraft.

Two separate CDS commands (power distribution and pyro fire) are needed to activate a spacecraft pyro event, plus one more command (pyro enable) for certain mission-critical pyro events (e.g., probe release and 400-newton engine start).

All three spacecraft pyro inhibits will remain in place until 45 minutes after separation from Shuttle. At this time, three independent Centaur timers, initiated by separate STS-Centaur breakwire signals, will remove two of the inhibits (PSU safe/arm and PSU short/unshort), using a two-of-three voting scheme. The two-of-three voting is implemented together with redundant wiring to provide single-fault tolerance, as required by Galileo project policy for mission success. About 10 minutes later, at Centaur Main Engine cutoff (MECO), a Centaur discrete signal will initiate a stored sequence in the CDS, which will power on PSU and start the spacecraft boom deployment sequence.

The Centaur PSU arm and PSU unshort signals will also remove the two probe power inhibits, located in the spacecraft fuse and bleed assembly (FBA), and the two protected loads inhibits, located in the spacecraft memory power subassembly (MPS).

The purpose of probe power inhibits is to ensure two-fault tolerance against premature power turn-on for three potentially catastrophic spacecraft loads:

a. S-Band Exciter. While in the Shuttle bay, premature turn-on of the S-Band exciter with the traveling wave tube amplifier (TWTA) inadvertently powered would result in RF radiation potentially hazardous to the Shuttle crew.

b. HGA Deployment Motors. If HGA deployment is initiated while in the Shuttle, the antenna or pieces broken from it may become lodged in the opening/closing mechanisms of the Shuttle bay doors. As a result, the bay doors may not be able to close properly for reentry.

c. EPD Instrument. The EPD cover is released by actuation of a wax pellet commanded by CDS. Power to the wax pellet is inhibited when EPD instrument power is off. If EPD is powered while in the Shuttle bay and the wax pellet is inadvertently actuated by CDS, the cover may break loose and, like the broken antenna pieces, cause the Shuttle bay doors to jam.

C19.3.13.2 RTG Cooling - Spacecraft electrical power is provided by the two radioisotope thermoelectric generators (RTGs). While in the Shuttle, each RTG generates 225W of electrical power and radiates almost 4200W of thermal energy. To prevent overheating of the Shuttle while on the launch pad and during flight, the Shuttle is equipped with an RTG active cooling system (ACS) to cool the RTGs.

The RTG ACS is a water coolant system capable of removing 85-90% of the thermal energy generated by the RTGs (up to 15,000 Btu/hr/RTG). The RTG ACS maximum operating pressure is about 85 psi, and the nominal flow rate is about 930 lb/hr. Figure C19-11 shows a schematic of the RTG cooling system.

The RTG cooling loop is filled and activated following installation of the RTGs, about five days before lift-off. At lift-off the cooling supply will switch from ground support equipment (GSE) to internal Shuttle supply using the Shuttle flash evaporators which will be fully activated within four minutes after lift-off. About six minutes before spacecraft/Centaur separation from Shuttle, the ACS cooling pump will be stopped and the cooling lines purged with dry nitrogen.

Two pyro-activated separation devices (one for each RTG) and the cooling lines connecting the RTGs to the ACS are mounted on the spacecraft-to-Centaur adapter. The separation devices, which are actuated by spacecraft pyrotechnics before spacecraft separation from Centaur, will disconnect the cooling lines at the RTGs.

Failure of the RTG cooling system at lift-off will not by itself cause damage to the Shuttle or require the use of STS abort/emergency procedures. Furthermore, the spacecraft is designed to accommodate a worst-case hot abort condition with loss of RTG cooling at lift-off, without causing a hazardous situation to the Shuttle and the crew.

C19.3.13.3 RTG Shorting - One of the most significant safety-related design features on the Galileo spacecraft is the ability to short the RTGs in case of a mission abort and thereby eliminate power to all spacecraft loads except CDS and AACS memories. This ability is required to protect against possible overheating and rupture of the RPM propellant tanks for a worst-case hot abort condition.

Heating of the RPM tanks is provided by a 30-Vdc shunt regulator that regulates the spacecraft power and dissipates all excess power into a set of shut heaters mounted near the RPM propellant tanks. However, in case of a failure, all switchable spacecraft power loads can be turned off. The RPM shunt heaters would then dissipate about 300W into the RPM. Thermal analysis for a worst-case hot abort condition shows that at this (maximum) rate of heating, the RPM propellant tanks could exceed their rupture temperature of 62°C (based on elastic tank model) within 48 hours after touchdown. RTG shorting was incorporated to divert all power from the RPM shunt heater in case of an abort, thus preventing this unsafe condition.

The RTG shorting function is provided by a set of four latching relays located in the MPS and actuated by Centaur. RTG shorting will be performed by the Centaur as part of the Centaur abort sequence, which also dumps the Centaur propellant load overboard the Shuttle in case of an abort.

To ensure that post-landing RPM propellant tank temperature will remain below 62°C indefinitely, air-conditioned purge of the cargo bay after touchdown is required at all primary and contingency landing sites.

For nominal operations, STS safety requires a factor of safety equal to or greater than 1.5 for the RPM propellants tanks. For an abort, the factor of safety is required to remain above 1.0. The RPM tank temperatures corresponding to factors of safety equal to 1.0 and 1.5 are about 62°C (elastic)/55°C (rigid) and 45°C, respectively. Under nominal launch conditions the RPM tank temperature is expected to be between 20°C and 30°C.

C19.3.13.4 <u>Structural</u> - All structural elements of the Galileo spacecraft are designed with an ultimate factor of safety greater than or equal to 1.4, as required by STS safety. The structural loads and thermal environment used in the design of the spacecraft structure represent worst-case launch and abort landing conditions. The design loads are derived from simulated Shuttle/Centaur launch and landing dynamics adjusted at some frequencies to reflect STS-2 flight data. The design thermal environment reflects a worst-case hot abort condition with loss of RTG cooling at lift-off and no opening of the payload bay doors.

In addition, STS safety requires that all fracture-critical elements are reviewed specifically to prevent structural failures due to crack formation and growth. This task is accomplished through an extensive fracture mechanics and stress corrosion cracking review process involving all structural parts of the spacecraft.

C19.3.13.5 <u>Flammability</u> - STS safety requires minimum use of flammable materials. Wherever flammable material cannot be avoided, spacing from potential ignition sources (e.g., electrical wiring) must be provided to prevent flame propagation.

To meet these requirements, extensive use of nonflammable materials has been emphasized in the spacecraft design process. For example, almost all electrical wiring is insulated with Kapton, which is considered nonflammable (flash temperature is about 900°C). In a very few cases, exclusively within the RPM and the probe, Raychem-insulated wires (with a lower flash temperature) are used in place of the more rigid Kapton-insulated wire type. The Raychem-insulated wires are either not powered while in STS or fused and current limited to prevent ignition in case of a short.

In general, all switchable power loads on the spacecraft are turned off while in the Shuttle unless they are required to ensure proper functioning of the spacecraft, e.g., to maintain thermal control.

C19.3.13.6 Nuclear and Electromagnetic Radiation - Use of radioactive materials is limited to the following three areas of the spacecraft design: radioisotope thermoelectric generators (RTGs), radioisotope heater units (RHUs), and radioisotope calibration targets. Both the RHUs and the calibration targets are vacuum encapsulated to prevent leakage, and have radiation levels not hazardous to the STS crew. The total number of RHUs is 105. The RHUs are used in place of electrical heaters, where applicable, to remain within spacecraft electrical power constraints.

The only major radioactive sources are the RTGs. Special handling and installation procedures have been developed for the RTGs to limit radiation exposure.

While in the Shuttle bay, STS safety requires that electromagnetic radiation from spacecraft RF sources does not exceed certain power levels. The two-fault-tolerant electrical inhibits, which preclude power to the S-Band exciters accommodate this requirement and hence ensure crew safety.

C19.4 ULYSSES SPACECRAFT SYSTEMS DESCRIPTION, HAZARDOUS, MATERIALS, SCHEMATICS

Although the original plans included both a European and a NASA spacecraft, subsequent cancellation of the latter has left the European entry as the sole surviving spacecraft. This spacecraft is illustrated in Figure C19-4.

Prominent characteristics of the spin-stabilized spacecraft are the large-diameter (1.65m) parabolic high gain antenna (HGA) on top of the spacecraft, the RTG, the 5.5-meter radial boom that provides an appropriate environment in terms of electromagnetic cleanliness for certain experiments, and the 7.5-meter-long axial boom. The body of the spacecraft contains all subsystems and science instruments with the exception of those experiment sensors mounted on the 5.5-meter boom. All of the internally mounted units are carried on a honeycomb center panel that also supports the hydrazine fuel tank. In general, the experiments that are more sensitive to nuclear radiation are mounted on the portion of the spacecraft farthest from the RTG, while the less sensitive subsystems are nearer to it.

Mass properties and balance of the spacecraft have been drivers in the design. Requirements on the launch configuration and for the HGA pointing are met. The spin axis of the launch configuration is the geometric-centerline Z-axis of the spacecraft. The electrical axis of the HGA will be aligned with the theoretical orbital spin axis. The tilt angle of the principal axis with maximum inertia has been minimized (0.10 degrees half cone). In order to ensure quasi-omnidirectional coverage for at least near-Earth operations, two low gain antennae (LGA) have been chosen to complement the HGA.

C19.4.1 Structure Subsystems

The spacecraft structure is a box structure with two overhanging balconies and a single aluminum honeycomb platform. Sidewalls are also aluminum honeycomb covered externally with multilayer insulation. The platform provides sufficient area to mount all electronic units of the scientific and spacecraft subsystems, most of the sensors, and the propellant tank.

The axial boom is mounted on the bottom plate, which serves as a thermal radiator. The redundant TWTAs are also mounted on the bottom on a thermally decoupled bracket.

Four vertical longerons provide the box cover supports and the interface attachment points to the adapter. They also form the hoist points for hoisting the spacecraft.

C19.4.2 Mechanism Subsystem

The mechanism subsystem consists of:

a. A radial boom to deploy the experiment sensors to a distance from the spacecraft body compatible with magnetic and radioactive requirements; there are two separated holddown points for support of the boom during launch. Deployment will take place after separation from the Centaur by activating a pyrotechnic cable cutter at each position.

b. A pair of radial extending wire booms to form a dipole for the plasma experiment; they are stowed during all launch phases; deployment will occur after TCM-2.

c. An axial boom, deployed on the -z axis, to form a monopole antenna for the plasma experiment; deployment will take place after TCM-2.

C19.4.3 Thermal Subsystem

Thermal control is achieved by passive means and the operation, by commands, of internal/external power dump system and heaters applied to individual spacecraft units, such as reaction control elements and some experiments. The most stringent requirements on the thermal subsystem are to guarantee at all times a temperature above 5°C for the hydrazine in the reaction control equipment and a temperature lower than 35°C for all solid-state detectors used by experiments.

All insulated spacecraft walls are covered with blankets closely fitting around sensor apertures, and all units external to the spacecraft are thermally decoupled from the interior. Heat rejection is performed by the -Z radiator panel, where the TWTAs are located.

The thermal blankets consist of 20 or 12 layers of aluminized mylar or Kapton; the outermost layer is Kapton-coated with a transparent conductive coating (TCC).

C19.4.4 Power Subsystem

The energy source is an RTG. The power subsystem conditions this power and delivers it to the subsystems and experiments at 28V, plus or minus 2%. Regulation is achieved by an active shunt regulator with linear and fixed internal/external dumpers. Secondary voltages for the data handling subsystem and the Attitude and Orbit Control Subsystems (AOCS) are generated by DC/DC converters with regulated output voltages. The power subsystem distributes the power through parallel redundant main switches to the experiments and through latching current limiters to the S/C subsystems. The main switch is associated with an over-current protection that provides an OFF command after a few milliseconds of excess current. Power to S- and X-Band transmitters; pyro subsystem; and all AOCS, Reaction Control Equipment (RCE) components, and axial and wire booms are inhibited by separation switches or break connectors during the launch phase.

C19.4.5 Pyro Subsystem

The pyro subsystem provides the circuits for radial boom and experiment cover release. There are a total of 10 redundant firing circuits. The subsystem is inhibited during launch by the separation switch. After separation from the Centaur, power and arming of the subsystem is performed by ground commands.

C19.4.6 Attitude and Orbital Control Subsystem

The primary operational functions of the AOCS are to maintain the S/C spin axis Earth-pointing and to control the spin-rate. Additional operational functions are dictated by TCMS, nutation damping, experiment-required measurements, and control of the spin-rate and spin-phase reference signals.

The main components are:

a. Attitude measurement units consisting of redundant X-beam sun sensors that allow measurement of spin-rate and solar aspect angle in the range of 1.5 degrees to 110 degrees solar aspect angle; meridian sensors to determine the spin-rate in the range of 90 degrees to 150 degrees solar aspect angle; and a CONSCAN system to measure the angle between the axis of the HGA and Earth.

b. Control electronics that perform closed-loop control via CONSCAN, spin, and solar aspect angle. Open loop maneuvers can also be executed by ground command.

c. The reaction control equipment includes a single hydrazine bladder tank, mounted on the launch center-of-gravity, feeding eight two-newton catalytic decomposition thrusters arranged on two manifolds and isolated by latch valves.

d. Three nutation dampers are carried for reasons of redundancy. The dampers are of the fluid-in-tube type and are mounted with the damper axis in a plane normal to the spin axis.

The AOCS power is inhibited during the STS-flight by separation switches until separation from the Centaur.

C19.5 GALILEO MISSION SCENARIO

The mission goal is to obtain more detailed scientific data on Jupiter and its satellites. To obtain such data, a Jovian Orbiter and a Jovian atmospheric probe will be used, both being parts of the Galileo spacecraft.

The overall scientific objectives of the Galileo mission are to conduct comprehensive investigations of the Jupiter planetary system by making in-situ and remote measurements of the planet, its environment, and its satellites. Investigations of the Galilean satellites of Jupiter will constitute a major objective of the mission. Close-up study of the planet and its principal satellites will greatly extend our knowledge of the role of the Jovian system in the complex and analogous relationships existing between the sun and its planetary system.

The Orbiter mission encompasses an equatorial tour of the planet system and multiple encounters with the satellites Europa, Ganymede, and Callisto. An Io encounter is possible before orbit insertion. The Orbiter will conduct a synoptic study of the Jovian atmosphere, determine the distribution and stability of trapped radiation, and define the topology and dynamics of the outer magnetosphere, magnetosheath, and bowshock.

The probe will seek to determine the physical structure and chemical composition of the Jovian atmosphere to a pressure depth of at least 10 bars, the location and structure of the Jovian clouds in the troposhere, and the thermal balance of the planet. It also will characterize the upper atmosphere and the nature and extent of the cloud particles.

C19.5.1 Mission Phases

Figure C19-12 shows the major phases of the Shuttle missions that include RTGs in their payloads. At T-45 hours to T-8 hours, the RTGs are installed on the payload in the Orbiter payload bay. This presents certain hazards related to the installation and the ground servicing equipment. From T-8 hours to T-31 seconds the ET and the Centaur propellant tanks are filled and the Centaur leakage test is performed. At T-31 seconds, the STS launch sequence becomes automatic and controlled by the onboard general purpose computers (GPCs). Problems arising with the propellant storage, propulsion, or control systems during the period of T-31 seconds to T-0 would be controlled in a predetermined manner. Depending on the timing and the event, serious failure modes are possible. During this period, the SSMEs are ignited, the SRBs are ignited, the final ground connections are broken, and the launch pad tiedowns are released. Safe mission abort is possible up to SRB ignition and the release of the tiedowns. From T-0 through SRB separation (first-stage ascent), the following critical events occur: (1) clearing the launch tower, (2) passing the max (dynamic pressure), (3) separating and avoiding the SRBs and (4) remaining within the flight termination boundary as specified by range safety.

The second-stage ascent phase is from SRB separation through main engine cut-off (MECO). The critical events during this phase are heeding range safety boundaries and accomplishing MECO. Following MECO, the ET is separated from the Orbiter, thereby engendering the critical event of possible recontact. The OMS-1 and OMS-2 burns respectively provide the thrust for orbit insertion and orbit circularization. Each of these creates occasion for propellant and propulsion hazardous events. The preparation for the payload deployment involves opening the payload bay door and erecting the Centaur to launch position. The Centaur, with its payload containing the RTGs, is deployed during orbit number five. A successful deployment from the Orbiter terminates the NSTS responsibility for the hazardous event of releasing nuclear material from the RTGs.

C19.5.2 Launch to Jupiter Arrival

The STS launch initiating the Galileo mission is illustrated in Figure C19-13. The Centaur G-Prime vehicle will be used to boost the spacecraft from the low-altitude Earth orbit attained by the Shuttle. Although direct trajectories to Jupiter in 1986 require launch energy (C_3) in excess of 84 km^2/s^2, broken plane trajectories can be flown with lower values of C_3 with the spacecraft supplying additional delta V to change the heliocentric orbit plane. The Centaur will accelerate the Galileo spacecraft from the Shuttle deployment orbit (i.e., parking orbit) to a C_3 of about 80 km^2/s^2.

Figure C19-14 illustrates the time of the opening of the launch window for each launch date during the primary launch period for Galileo. Curves are shown for several possible C_3's (launch energies). The baseline launch energy for Galileo is 80 km^2/s^2; however, more optimistic estimates of the launch energy have also been examined to provide more comprehensive data.

C19.5.3 Separation from Shuttle and Centaur

Galileo launch mode telemetry will be available in near-real time via the STS-TDRS communications link from lift-off to Shuttle-Centaur separation. Based on the telemetry data, a decision to continue with the planned flight to Jupiter must be made by launch plus approximately 6 hours so that the Centaur will be rotated; a second decision must also be made by Launch plus approximately 6.5 hours so that the Centaur will be separated from the Shuttle. If a "go" decision's made, the Shuttle-Centaur separation should occur during the fifth Shuttle orbit, at about 6.75 hours after launch. On the more favorable days of the launch period, the separation could be delayed until approximately the 14th orbit without jeopardizing the objectives of the Galileo mission. After separation, a Centaur S-Band link can be used to route data between the spacecraft and Shuttle or TDRS. The maximum useful range of the Centaur-to-Shuttle link is 10 kilometers.

About 45 minutes after Shuttle-Centaur separation, the Centaur main engine will burn for approximately 10 minutes. After MECO, the Centaur will initiate a slow thermal roll of 0.1 rpm, and the Galileo spacecraft will start deployment of the RTG and science and magnetometer booms. The spacecraft transmitters will then be turned on to provide a downlink through the deep space network (DSN) just prior to Centaur-spacecraft separation. Centaur will turn the spacecraft to point 9 degrees to the Earth side of the Sun and will spin-up the spacecraft to 2.9 rpm. The spacecraft will then separate from Centaur, which will maneuver to avoid the same trajectory path as the Galileo spacecraft. The spacecraft transponder will now be the only means of exchanging data between the Earth and the spacecraft.

Table C19-5 briefly describes events for the Galileo spacecraft while on-orbit.

Table C19-5 Galileo Spacecraft On-Orbit Activities

On-Orbit Events (Nominal Time)*

1 001:23:00 MET. Open payload bay doors.

2 001:45:30 MET. Send Centaur Discrete Command Set A to start S/C thermal control sequence. The 'S/C Thermal Control Seq' will:

- Enable selected fault protection algorithms for temperature control.
- Turn on selected heaters to maintain thermal control and prevent instrument contamination after cargo bay doors open. Nominal 3-min-36-sec duration.

3 05:17:30 MET. Activate Deployment Adapter Rotation.

Table C19-5 Galileo Spacecraft On-Orbit Activities - Concl

	On-Orbit Events (Nominal Time)*
4	06:19:00 MET. Send CTR discrete CMD Set B to start S/C PWR sharing sequence. The 'S/C POWER SHARING SEQ' will turn off selected heaters to meet RTG power constraints as the RTG coolant pump is turned off and the RTG power decreases. (Nominal 14-sec duration).
5	06:29:00 MET. JPL Go/No-Go for separation.
6	06:29:24 MET. Stop RTG cooling pump and start purge of RTG cooling lines. RTG power output will decrease to a minimum within 20 min. After that, RTG power output will increase gradually.
7	06:41:24 MET. Fire Super*Zip. S/C-Centaur payload separates from Shuttle Orbiter.
8	07:26:02 MET. ARM and UNSHORT S/C PSU 1 and PSU 2. The two independent safety inhibits (ARM and UNSHORT PSU 1 & 2) are removed 45 min after separation. Centaur Main Engine is also enabled at this time.
9	07:31:24 MET. Centaur Main Engine Start (nominal burn of 9 min 36 sec).
10	07:41:10 MET. Send Centaur Discrete Command Set C to start S/C deployment Phase 1 Sequence (nominal 24-min-38-sec duration).
11	07:42:28 MET. Deploy RTG & Science Booms.
12	07:49:10 MET. Deploy Mag Boom.
13	08:01:50 MET. S-Band TWTA on.
14	08:06:42 MET. Centaur turn to Sun and spin-up to 2.9 rpm.
15	08:08:42 MET. Begin acquisition S/C DSN S-Band Downlink (nominal).
16	08:11:12 MET. Send Centaur Discrete Command Set D to start S/C deployment Phase 2 sequence.
17	08:11:24 MET. S/C-Centaur separation.
18	08:36:28 MET. Deploy HGA.
19	08:36:28 MET. Pressurize RPM.
20	09:05:16 MET. S/C back-up Sun acquisition.

* Reference: For details, see JPL PD 625-205, GLL 3-120, Launch Sequence of Events

C19.5.4 Post-Separation to Jupiter Arrival - Immediately after separation, the high-gain antenna will be deployed and the retropropulsion module (RPM) pressurized. After RPM pressurization, the Galileo spacecraft will perform a sun acquisition (approximately one hour after Centaur-spacecraft separation).

About eight months after launch, the spacecraft propulsion system will impart a delta V of about 200 to 300 m/s using its RPM 10-N thrusters. The purpose of this maneuver is to change slightly the heliocentric inclination so that the spacecraft will intercept Jupiter at the desired arrival date. The position of Jupiter at the nominal arrival date is about one degree below the ecliptic. Ballistic trajectories without this plane change maneuver require large heliocentric inclinations at launch. The plane change maneuver thus reduces the C_3 significantly, because the spacecraft trajectory departing the Earth lies nearly in the ecliptic.

The interplanetary cruise operations activities (e.g., cruise science, navigation, spacecraft monitoring, etc) on the Earth-to-Jupiter trajectory will be comparable to those planned for the previous Galileo missions. The trajectory must be targeted for a Jupiter arrival no earlier than August 1988 in order to satisfy the probe-to-Orbiter relay link geometry. Arrival dates between mid-September and late October are excluded so that probe separation does not occur near the time of solar conjunction. Therefore, arrival at Jupiter will probably occur sometime between early August and mid-September.

During the cruise to Jupiter, the probe will be in a non-energized condition with both power and command capability inhibited except for periods of probed checkout. After a cruise period of about 675 days, and about 150 days prior to arrival at Jupiter, the Orbiter and probe will be spun up to 10 rpm and the probe separated from the Orbiter. Except for a timer, the probe is passive during the 150 day coast towards Jupiter. The probe is targeted to a daylight entry within a few degrees from the equator. At an altitude of 450 km above 1-bar pressure, it will have a nominal speed of less than 47.8 km/s and flight path angle of -8.6° relative to the atmosphere. Deceleration from this point to sonic velocity will take place over a period of about two minutes, during which the probe will experience a nominal peak deceleration of about 250g and peak heating rates approaching 500 Mw/m^2. At approximately Mach 1, a pilot parachute will be deployed by a mortar whose pyrotechnics firing signal is derived from measurements of the deceleration profile. The pilot parachute, in turn, will remove the deceleration module aft cover and deploy a main parachute. The main parachute then will separate the descent module from the deceleration module and control the speed of the descent module throughout the rest of the mission. Those instruments and subsystems which had not been actuated prior to entry will be turned on at the time of descent module separation. Instrument data acquired prior to time of descent module separation will be transmitted to the Orbiter together with data acquired during the remainder of the descent through the atmosphere. The probe is designed to operate to a pressure level of at least 10 bars.

C19.5.5 Abort Operations

In the event of problems incurred while on-orbit, it would be possible to abort the Galileo mission and still return with the spacecraft in the Shuttle payload bay. Table C19-6 gives an overview of the operations planned for after landing should such an abort be necessary.

In the event of a Centaur main engine failure, the main engine will automatically attempt to fire a second time. Should the second attempt to fire fail, the Centaur will automatically step through its normal post-MECO sequence of events on a speeded-up time scale so that the Galileo spacecraft will separate from the Centaur in the event of a main engine failure.

This would leave the Galileo spacecraft in a low Earth orbit, the Shuttle's 130-nm parking orbit, with a low lifetime on the order of 10 to 20 days. At this point, Galileo is fully loaded with propellant that would be used to boost the spacecraft into a higher, longer-lifetime orbit. The available propellant in the spacecraft will provide a delta V of approximately 1 km/s. This delta V will allow the spacecraft to achieve a circular orbit of approximately 2000-km altitude which has a lifetime on the order of thousands of years. This maneuver would most likely be achieved with the 10-Newton thrusters because the 400-Newton engine is blocked by the probe. Two sets of multiple burns would be required: the first set to achieve an elliptical orbit with an apogee of approximately 2000 km altitude and the second set to circularize this orbit.

C19.5.6 Jovian System Exploration

About 150 days after separating from the probe, the Orbiter will fire its retro engine, thereby entering orbit around Jupiter. Following orbit insertion and the perijove raise (PJR) maneuver necessary to reduce the radiation dose the Orbiter will receive, a series of close encounters with the Galilean satellites will be targeted. These encounters will not only permit close-in scientific investigations, they will also provide gravity assists to the Orbiter, providing the necessary trajectory shaping required to reach subsequent satellites in the tour. The remaining Orbiter propulsive capability is used primarily for navigational purposes.

Multiple encounters with the same satellite over the duration of the satellite tour will allow exploration of both equatorial and polar regions. In addition, the satellite tour will allow intensive scientific measurements of Jupiter and its magnetosphere. The nominal mission will end 20 months after orbit insertion.

C19.5.7 Asteroid Encounter Option

NASA is considering the option of having the Galileo spacecraft encounter the asteroid Amphitrite on the way to Jupiter. If a trajectory to enable this option is selected, the arrival date at Jupiter will change from

Table C19-6 Galileo Spacecraft Abort Post-Landing Activities

1 Spacecraft in Shuttle Bay (doors closed):

- Verify RTGs condition and radiation levels around Orbiter bay.
- Monitor propulsion condition for evidence of leaks.
- Monitor temperature and humidity levels in Orbiter bay.

Note: S/C can be without air conditioning for about 30 hours. RTGs are shorted in abort case.

2 Spacecraft in Shuttle:

- Inspect S/C for evidence of physical damage.
- Safe pyrotechnics super-zip.
- Remove RTGs and place in shipping container.
- Move RTG to a storage area and secure away from all personnel.

Note: Capability must exist to remove RTGs (KSC stands).

3 Centaur/Spacecraft removed and placed on Payload Special Handling Stand:

- Vent propulsion helium tanks
- Initiate science purge and probe cooling.
- Safe pyrotechnics (Galileo Orbiter).
- Remove spacecraft from Centaur and place on transporter.
- Remove SXA from spacecraft.
- Remove lower adapter from spacecraft.
- Depressurize and defuel RPM.
- Make preparations to transport spacecraft and equipment.
- Transport to KSC.

August to December 1988, and a slight change in the launch geometry will therefore be required. The launch period will be up to two hours later than the baseline times.

C19.5.8 Galileo Mission Functions

The mission functions of the Galileo Jovian Orbiter are as follows:

a. Deliver the probe to the designated release point near Jupiter and relay the probe measurements.

1. Provide radio relay link/command/power functions during the launch-to-injection phase.

2. Conduct periodic status checks (every six months) of the probe system and adjust S/C attitude when required during cruise phase (which is about 660 days).

3. Perform final probe check, provide required pointing or attitude at release, and release the probe approximately 150 to 100 days before encounter.

4. Perform a deflection maneuver after probe release.

5. Configure for data acquisition, receive probe data transmitted for a period of at least 60 minutes, simultaneously store and retransmit data in real time.

6. Playback stored data.

b. Accomplish a scientific investigation of Jupiter, its satellites, and its environment.

c. Perform:

1. Trajectory correction maneuvers (TCM)

2. The broken-plane TCM

3. Jupiter orbit insertion (JOI)

4. Orbit trim maneuvers

C19.5.9 Galileo Jovian Atmospheric Entry Probe

The probe will seek to determine the physical structure and chemical composition of the Jovian atmosphere to a pressure depth of 10 bars or more, the composition and structure of the Jovian clouds in the troposphere, and the thermal balance of the planet. It will also determine the nature of the ionosphere and upper atmosphere and the trapped radiation near the planet. The probe will separate from the Orbiter about 150 to 100 days before entering the Jovian atmosphere, where for 60 minutes the instruments will make measurements down to a pressure of about 10 Earth atmospheres, relaying data to Earth through the Orbiter communications system.

Before separation from the Orbiter, the probe will be spinning at 10 rpm in order to achieve stability. It will be released with an attitude for atmospheric entry at nominally zero degrees angle of attack and an initial entry angle of -8.5° relative to the atmosphere. These conditions are to assure that the probe will be aerodynamically stable during entry and descent.

The probe science payload has been selected to satisfy the scientific objectives of determining by in-situ measurements the physical structure and chemical composition of the atmosphere, the composition and location of the clouds in the troposphere, and the thermal balance of the planet. The basic complement of atmospheric-dedicated probe instruments was augmented by supplementary pre-entry science instruments to determine the nature of the ionosphere, upper atmosphere, and the trapped radiation near the planet.

The scientific payload consists of six scientific instruments with a combined weight of 64 lb and a volume of 1715 in^3. These instruments require 72 watts of electrical power and a total of six serialdigital, six analog, and eight bilevel telemetry channels. A total of 35 discrete commands from the sequence programmer and 35 discrete commands and one quantitative command from the Orbiter (for checkout during transit via the probe data/command processor) are provided for the instruments. Of the six instruments, three have sensors mounted external to the descent module, and three have inlets to admit atmospheric samples. The purpose of each instrument is given in Table C19-7.

C19.6 ULYSSES MISSION SCENARIO

The Ulysses (ULS) mission is a joint effort by European Space Agency (ESA) and NASA. The European contribution to the ILS consists of the provision and operation of the spacecraft and about half of the experiments. NASA will be responsible for providing the launch by the STS/Centaur, the remaining experiments along with the RTG, and mission support using the DSN. The primary scientific objectives of the ULS are to investigate, as a function of solar latitude, the properties of the solar corona, the solar wind, the structure of the sun-wind interaction, the heliospheric magnetic field, solar and non-solar cosmic rays, solar radio bursts and plasma waves, and the interstellar/interplanetary neutral gas and dust. Secondary science objectives of the mission include interplanetary physics during the initial Earth-Jupiter or ecliptic phase and measurements of the Jovian magnetosphere during the Jupiter flyby phase.

More specifically, the primary scientific objectives are to perform investigations as a function of solar latitude of the following:

a. The physics of the inner and outer corona of the sun.

b. The origin and acceleration of the solar wind.

c. The internal dynamics of the solar wind, of waves of shock, and of other discontinuities.

d. The propagation and acceleration of energetic particles.

e. Source locations of solar radio and x-ray bursts.

f. The acceleration, transport, and storage of energetic particles in the solar atmosphere.

g. The energy spectra, composition, and anisotropies of galactic cosmic rays.

h. Source locations of interstellar gamma rays.

i. The neutral component of interstellar gas/solar wind interaction.

Table C19-7 Probe Science Investigations

Instrument Name	Objectives
Atmospheric Structure Instrument (ASI)	Determine state properties (temperature, pressure, density, and molecular weight(of Jovian atmosphere
Neutral Mass Spectrometer (NMS)	Determine chemical and isotopic composition of Jovian atmosphere.
Helium Abundance Detector (HAD)	Perform precision determination of (HAD) helium abundance measurement in Jovian atmosphere.
Nephelometer (NEP)	Determine microphysical characteristics (particle size distribution number, density, and physical structure) of Jovian clouds.
Net Flux Radiometer	Measure vertical distribution of net flux of solar energy and planetary emissions.
Lightning and Radio Emission/ Energetic Particle Detector (LRD/EPD)	Study lightning in the Jovian atmosphere and energetic particles near Jupiter.
Radio Science	Study composition and structure of Jovian atmosphere.

j. The origin and behavior of interplanetary dust.

k. Radio science and gravitational wave investigation.

There is good reason to believe that the conditions found in the narrow band of heliographic latitudes sampled by spacecraft confined to the ecliptic plane are not representative of the inner heliosphere as a whole, and yet attempts to understand the basic physical processes occurring within this environment have so far been based on observations made in the ecliptic plane. ULS will for the first time permit measurements to be made in-situ away from the plane of the ecliptic and over the poles of the sun, its unique trajectory taking the spacecraft into the uncharted third dimension of the heliosphere.

C19.6.1 Nominal Mission

The Ulysses spacecraft will be launched at the ELS by the STS with a Centaur upper stage and will then travel nearly in the ecliptic plane to Jupiter. The gravitational field of Jupiter will be used to deflect the spacecraft into an out-of-ecliptic trajectory. After Jupiter flyby, the spacecraft will travel in a heliocentric out-of-ecliptic orbit, passing over the polar regions of the Sun. The choice whether to travel north or south after Jupiter shall be made no later than June 1986, so that as much information about the new solar cycle as is possible to collect may be considered in the decision. Figure C19-15 depicts the nominal ULS trajectory.

C19.6.2 Launch and Injection

The launch and injection phase begins with Shuttle lift-off and continues until spacecraft separation from the Centaur. Deep space network (DSN) acquisition occurs within 10 minutes of spacecraft separation (nominal deployment on Rev 5).

The daily launch window is one hour from 15 May 1986 to 7 June 1986 for injections from 15 May 1986 to 8 June 1986, respectively. The baseline C_3 is 133 km^2/sec^2, but it could be as high as 136 or as low as 130 to maximize performance. Launch times for these C_3s are within eight minutes of baseline C_3 launch time versus launch data curve.

An additional contingency launch period after the Ulysses secondary launch period is being considered. It is preliminary at this time.

Major activities in this phase include launch, Shuttle ascent, Shuttle orbit, Centaur/CISS/Ulysses checkout, Centaur deployment and separation, Centaur coast, Centaur burn, reorientation to separation attitude, spin-up Ulysses separation, Centaur collision and contamination avoidance maneuver (CCAM), and DSN acquisition.

A more detailed summary of these events is listed in Tables C19-8 and C19-9.

Table C19-8 Ulysses Launch-to-Injection Sequence of Events (Centaur)

Centaur Events			
Item	Time, h:m:s, h:m	Event	Duration, h:m
1	01:51:10 MET	CISS/Centaur Checkout	03:00
2	02:30:00 MET	Accelerometer Bias Calibration	15:00
	03:43:56 MET	Gyro Platform Rotation	02:00
3	06:40:00 MET	Centaur Separation	0 to 06:00
4	SEP + 05:00	Arm Centaur RCS	
5	SEP + 05:25	Tail-to-Sun Attitude Maneuver	04:30
6	SEP + 45:00	Arm Centaur Main Engine	
7	MES - 05:00	Reorient to MES Attitude	03:00
8	MES + 00:00	Centaur Burn	10:00
9	MECO + 05:00	LH2 Tank Venting Phase	05:00
10	MECO + 05:00	ULS Separation Attitude Maneuver	01:30
11	MECO + 07:30	Spin up to 4.7 rpm	01:40
12	MECO + 09:11	Ulysses Separation	
13	MECO + 09:16	Centaur Despin	00:54
14	MECO + 10:10	Reorient to CCAM Attitude	01:10
15	MECO + 11:20	CCAM Phase, LH2-LOX Blow Down	06:00
16	MECO + 17:20	Centaur Coast	

Table C19-9 Ulysses Launch-to-Injection Sequence of Events

Ulysses Events				
Item	Time, h:m:s, h:m	Event	Duration, h:m	Ref.
1	02:00:00 MET	Spacecraft Checkout, Reconfigure RTG Power Dump 75 Minutes Total, interrupted	(75:[illegible])	
2	06:40:00 MET	Centaur Separation	0 to 06:00	
3	MECO + 09:11	Ulysses Separation (USEP)		
4	USEP + 10:00	Ulysses Coast, DSN Acquisition		

Note: Item 1 depends on TDRS coverage. Checkout is not continuous but in blocks of time whenever available.

Until Centaur separation from the Shuttle, spacecraft-to-ground communications will be through the STS. Predicted and actual Centaur MECO state vectors will be sent from the Centaur Payload Operations Control Center (CPOCC) to JPL, and JPL will generate DSN predicts.

CPOCC will send the time of lift-off to JPL via voice and FAX so that acquisition predicts can be generated from launch polynomials supplied by LeRC (Centaur).

C19.6.3 Near-Earth

The near-Earth phase is from DSN acquisition until approximately day 30. During this phase, two trajectory correction maneuvers (TCM) will take place. The TCM-1 will occur nominally 10 days after separation from the Centaur and will be followed by TCM-2, which should occur about 30 days after launch. During this phase, spacecraft and science instrument checkouts and calibrations will be performed. DSN support tracking for navigation will be continuous. Orbit determination (OD), maneuver strategy, and control software verification will be emphasized during this phase. Tracking will be 24 hours per day with 64-m antenna coverage during TCMs.

C19.6.4 Earth-Jupiter

The Earth-Jupiter phase is post-TCM-2 to just prior to Jupiter encounter (J-25). After TCM-2, DSN support will be maintained at the nominal mission schedule of eight hours per day, seven days per week. A superior conjunction will occur nominally on day 310 of the mission, with potential loss of command/telemetry capability of up to 15 days around conjunction. At this time, enhanced DSN support is required for radio science.

C19.6.5 Jupiter Flyby

The Jupiter flyby is J minus 25 to J plus 1. About 15 days prior to TCM-3 (J minus 10), navigation will be emphasized in support of the TCM. At close approach to Jupiter, radiation-sensitive science instruments may be put in a safe state. DSN coverage will increase with required 64-m coverage. Passage through the Jovian radiation belts will be a major environmental hazard during the mission. This factor has been taken into account in both spacecraft subsystem and experiment design. The distance of closest approach to Jupiter will be 6-7 Jovian radii, and the radiation dose is expected to be slightly lower than that encountered by Pioneers 10 and 11 and by the two Voyager spacecraft.

C19.6.6 Out-of-Ecliptic

There are four out-of ecliptic phases:

a. From post-Jupiter flyby to the beginning of the first solar pass above 70 degrees heliographic latitude;

b. The first solar polar over-flight phase;

c. From the end of the first solar over-flight, through the ecliptic, to the beginning of the second solar pass above 70 degrees;

d. The second solar polar over-flight phase.

Although all four phases are of high scientific importance, the two polar over-flight phases have been formally declared as being of the most significant value.

C19.6.7 Mission Summary

A precis of the mission has been given as follows. The inecliptic mission phases are particularized by:

a. Launch Period: May 15 - June 8, 1986

b. Launch Energy: 133.0 km^2/s^2

c. Jupiter Encounter: July 18 - August 23, 1987

d. Closest Approach to Jupiter: 6.0 Jovian radii

The out-of-ecliptic mission phases are characterized by:

a. Perihelion Date: 13 June 1990

b. Perihelion Distance: 1.2 AU

c. Maximum Solar Latitude: 85°

d. Total Time above 70° (South First): 235 days

e. Total Time above 70° (North First): 172 days

f. Heliocentric Range at Maximum Latitude: 1.9 - 2.0 AU

g. Mission Termination: March 31, 1991

C19.6.8 Mission Abort Operations

In the event of problems incurred while on orbit, it would be possible to abort the Ulysses mission and return with the spacecraft in the Shuttle payload bay.

In the event of a Centaur main engine failure, the main engine will automatically attempt to fire a second time. Should the second attempt to fire fail, the Centaur will automatically step through its normal post-MECO sequence of events on a speeded-up time scale so that the Ulysses spacecraft will separate from the Centaur.

This would leave the ULS spacecraft in a low-Earth orbit (the Shuttle's 130-nm parking orbit) with a low lifetime on the order of 4 days. The spacecraft would have an available propellant mass that could provide a delta V of approximately 214 m/sec. Preliminary analysis indicates that this delta V would allow the spacecraft to achieve a circular orbit altitude of 330 nm, which is sufficient to provide an orbit lifetime of approximately 3 years.

The spacecraft propellant could also be used during spacecraft reentry to modify the reentry velocity to control the point of reentry, such as over a large body of water, if it can be demonstrated that reentry calculations can be accurately predicted. No study of this application of the spacecraft propellant has been initiated.

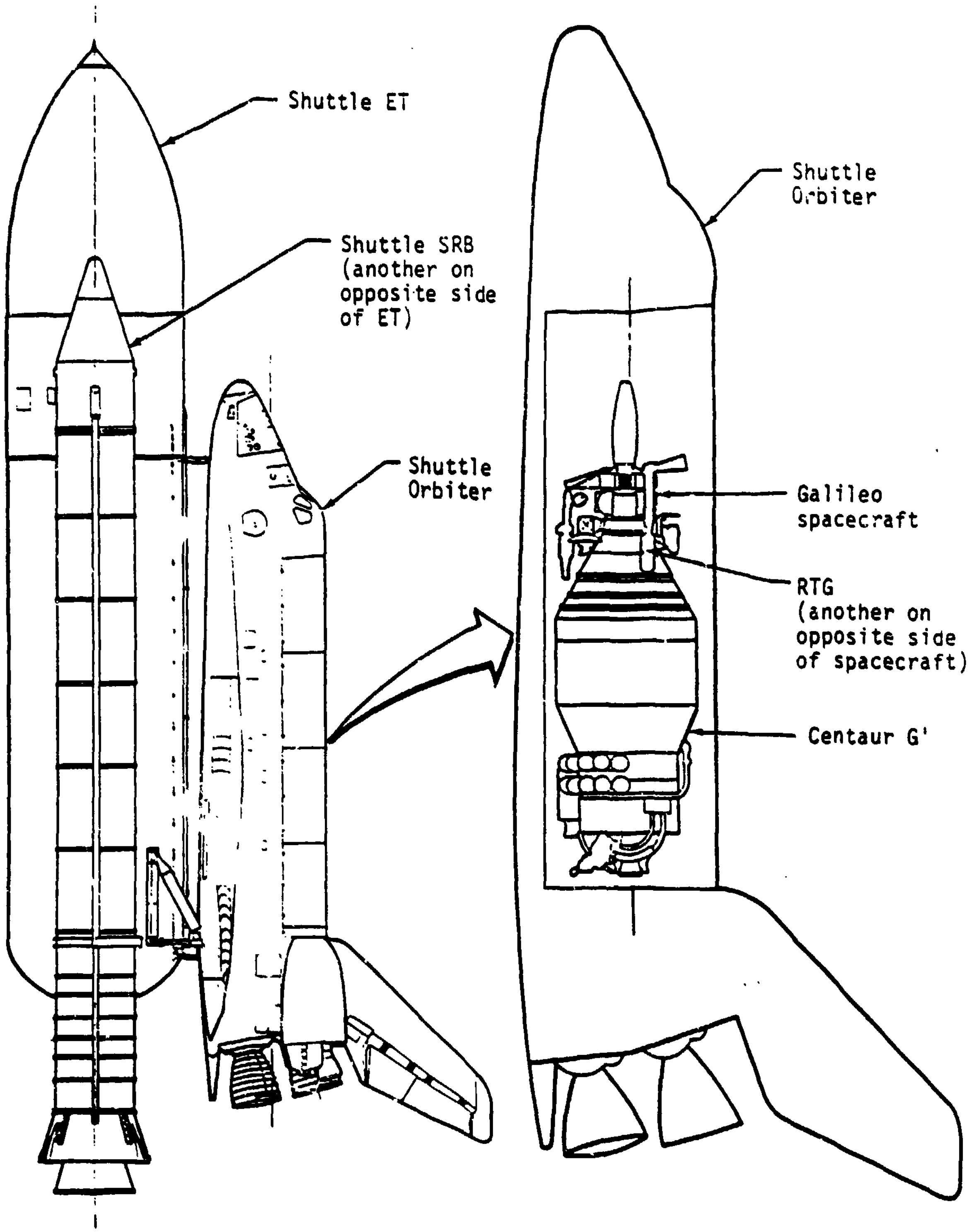

Figure C19-1 Galileo Launch Configuration

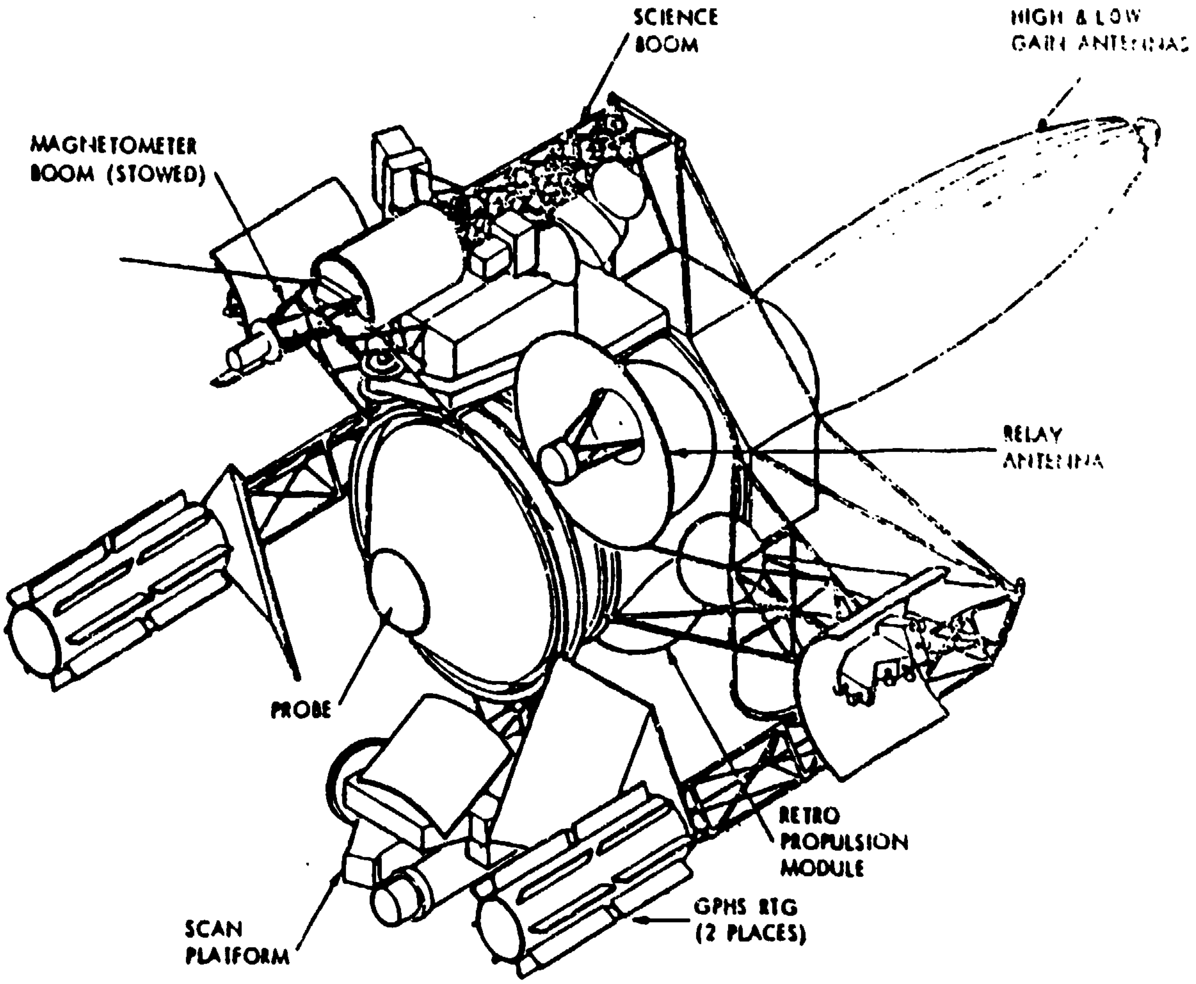

Figure C19-2 Galileo Spacecraft: Stowed Configuration

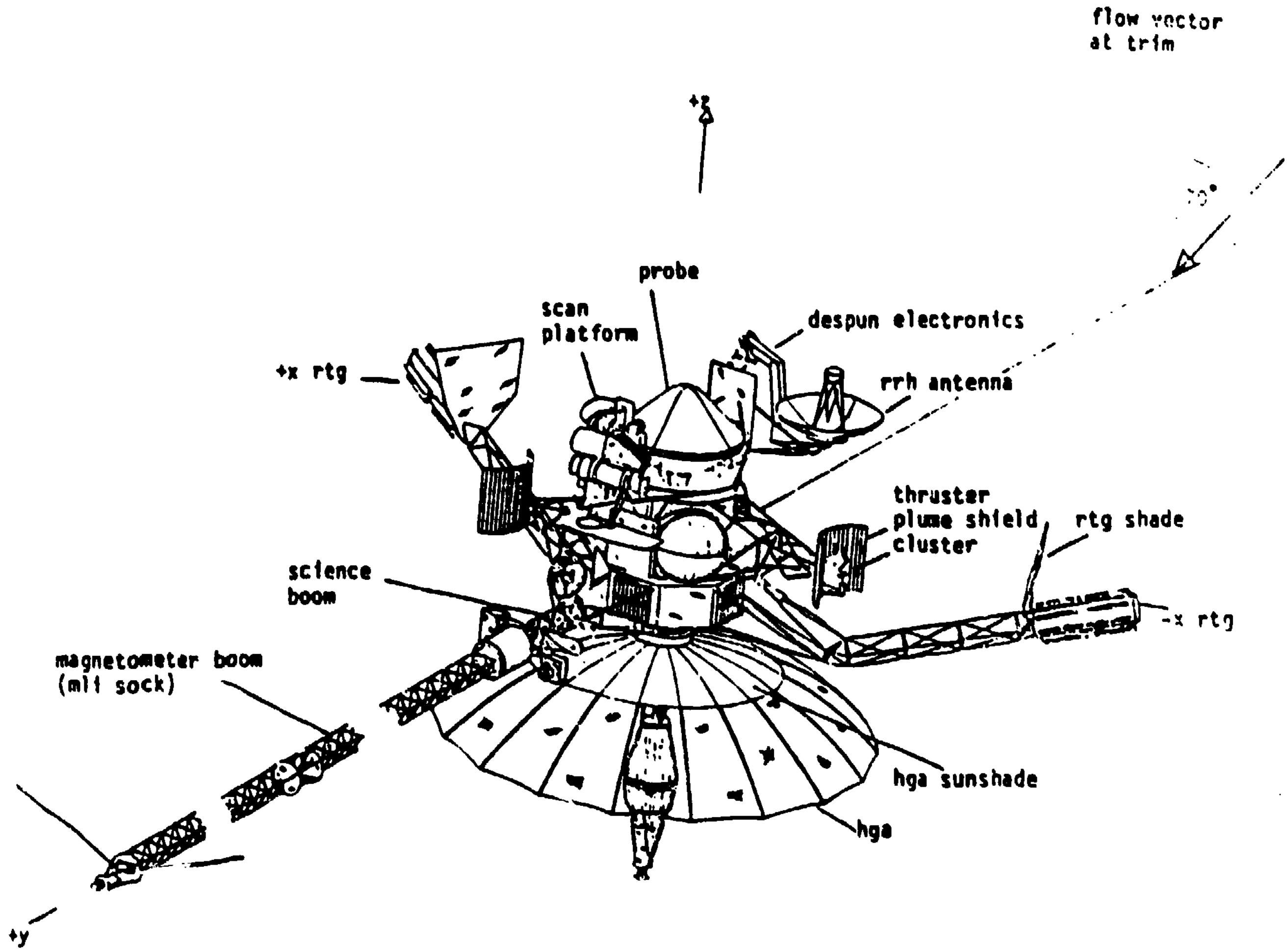

Figure C19-3 Galileo Spacecraft: Cruise Configuration

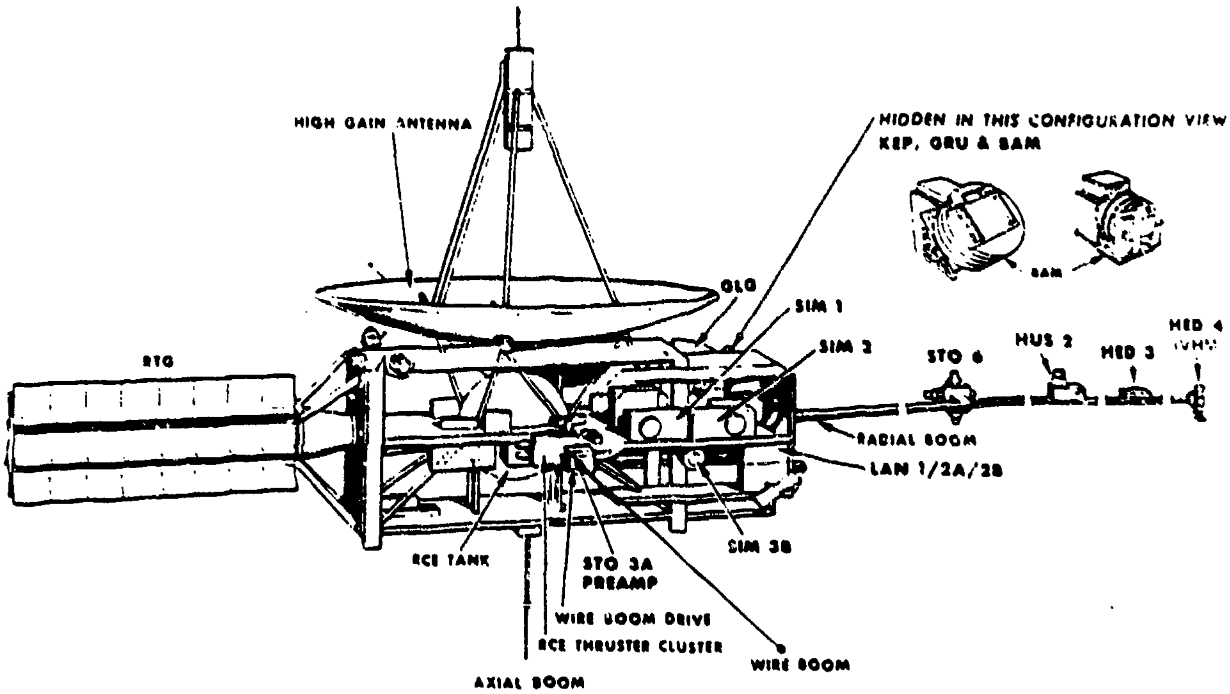

Figure C19-4 ESA Spacecraft Configuration for Ulysses

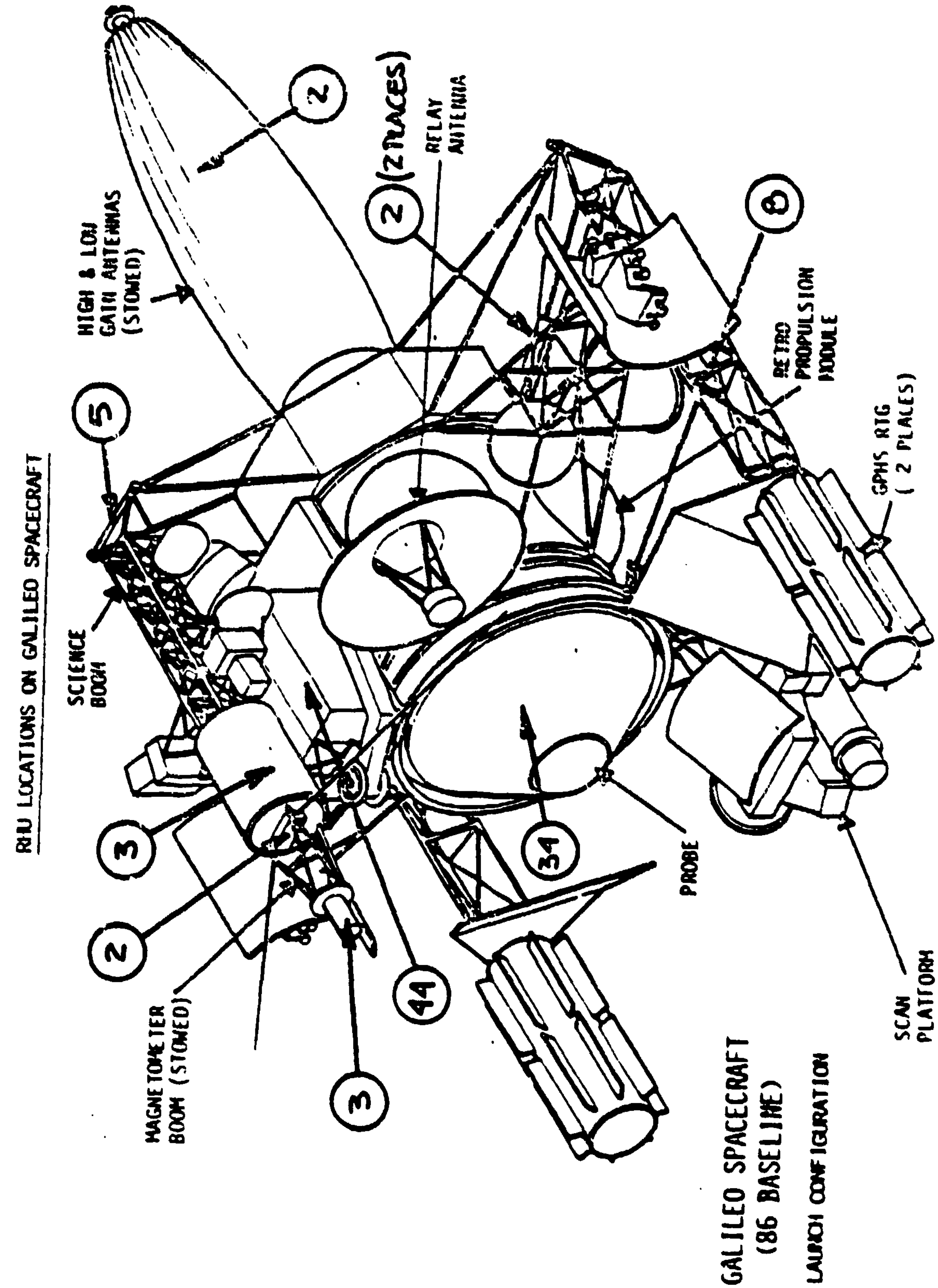

Figure C19-5 Galileo RHU Location

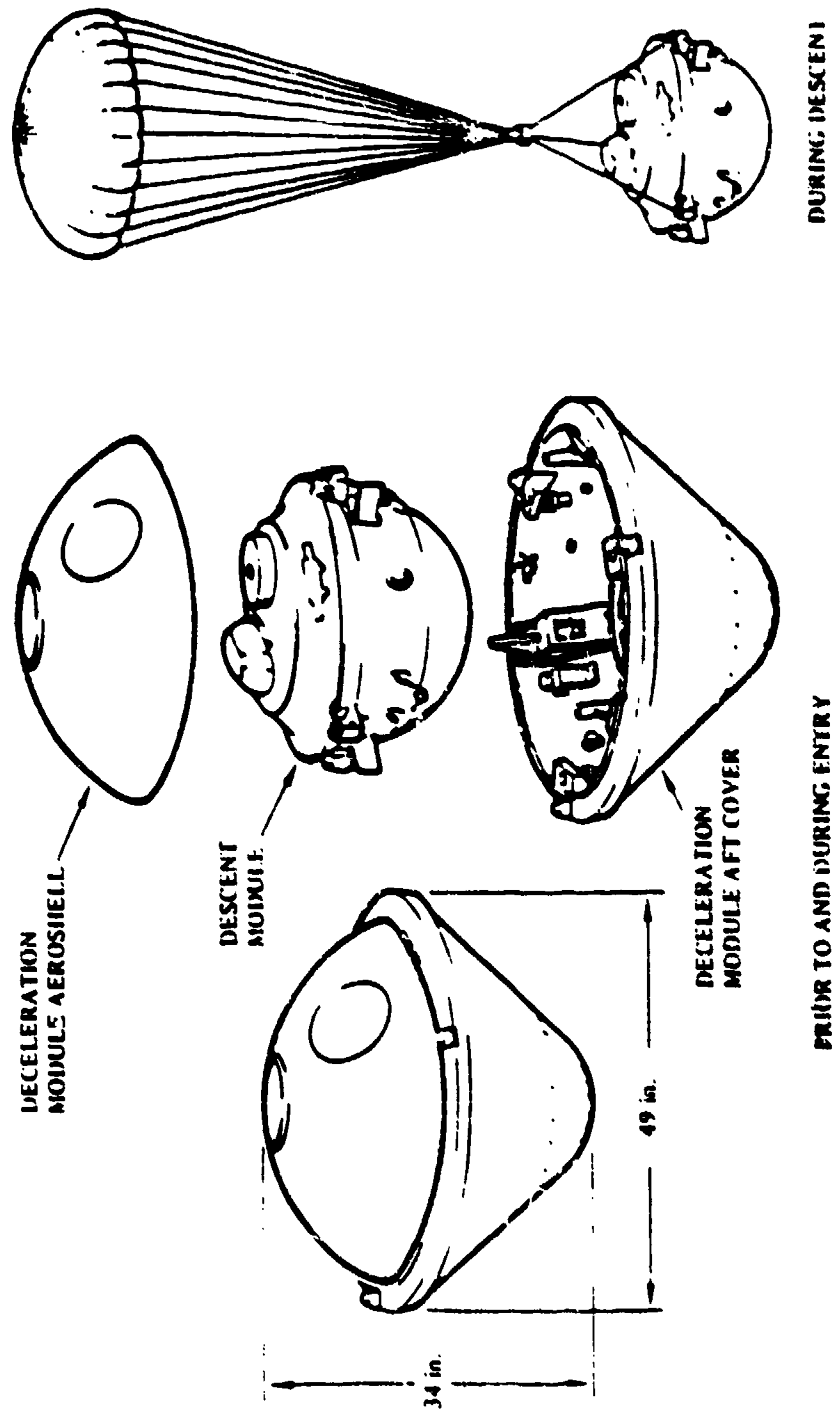

Figure C19-6 Galileo Probe

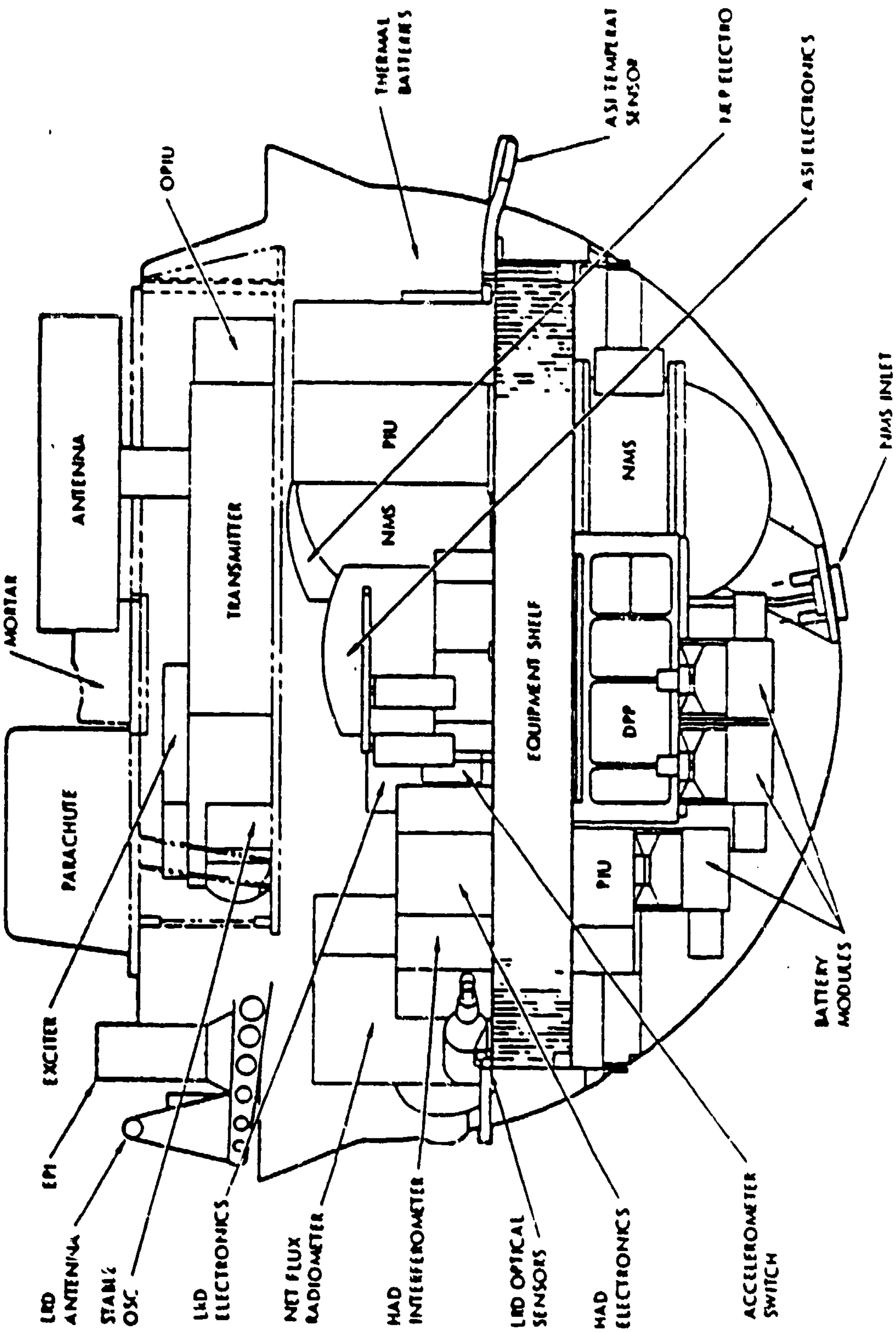

Figure C19-7 Galileo Probe Descent Module (Cross Section)

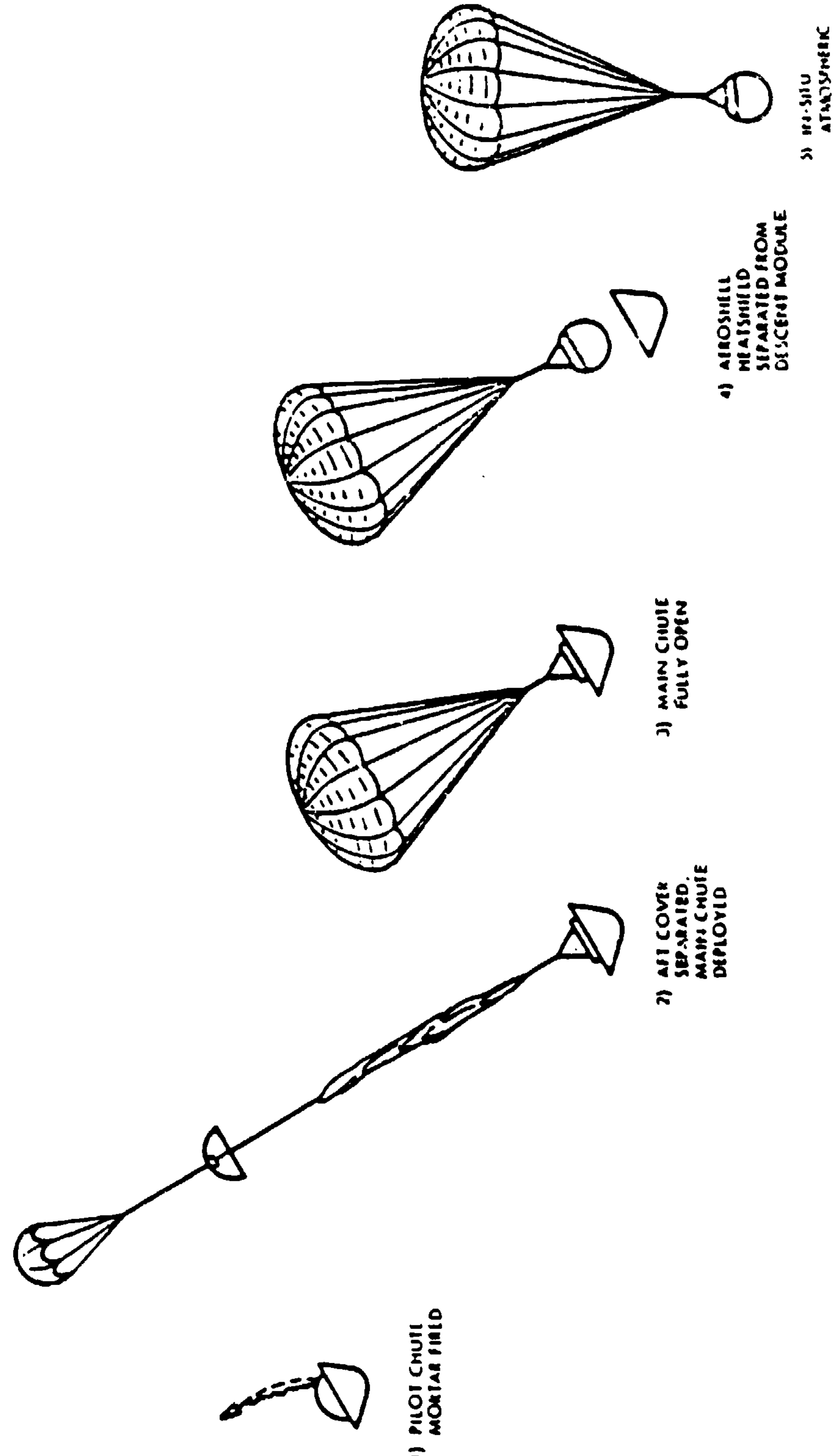

Figure C19-8 Galileo Probe System Concept

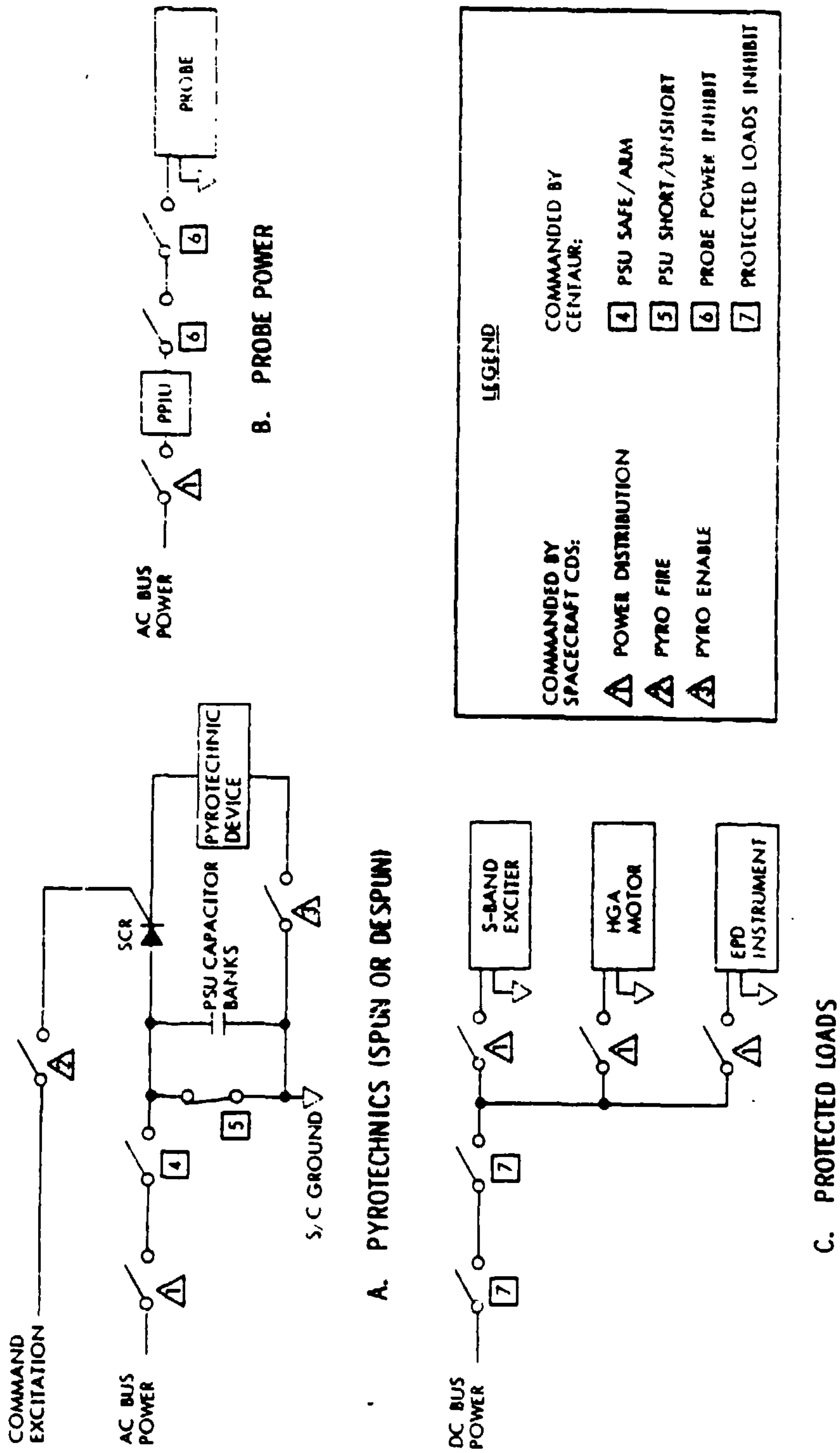

Figure C19-9 Galileo Two-Fault-Tolerant Safety Inhibits (Detail)

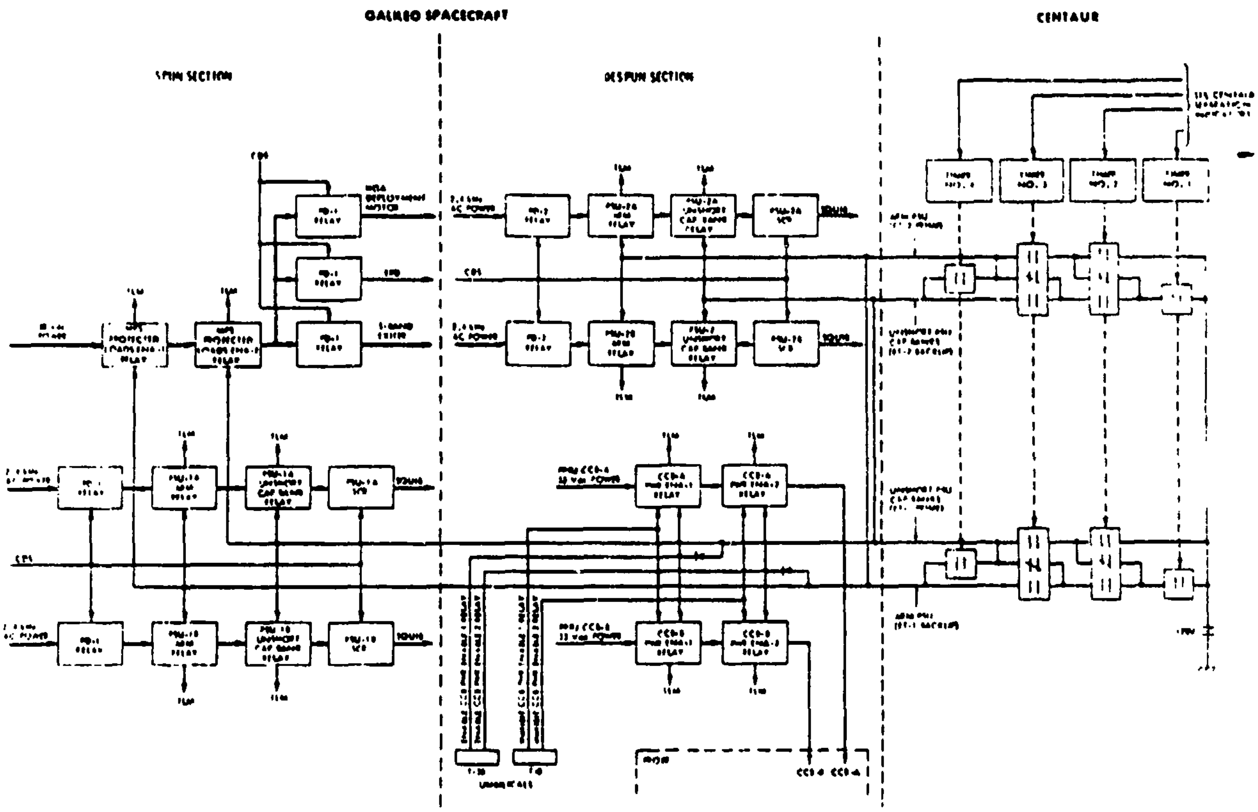

Figure C19-10 Galileo Two-Fault-Tolerant Safety Inhibits (Overview)

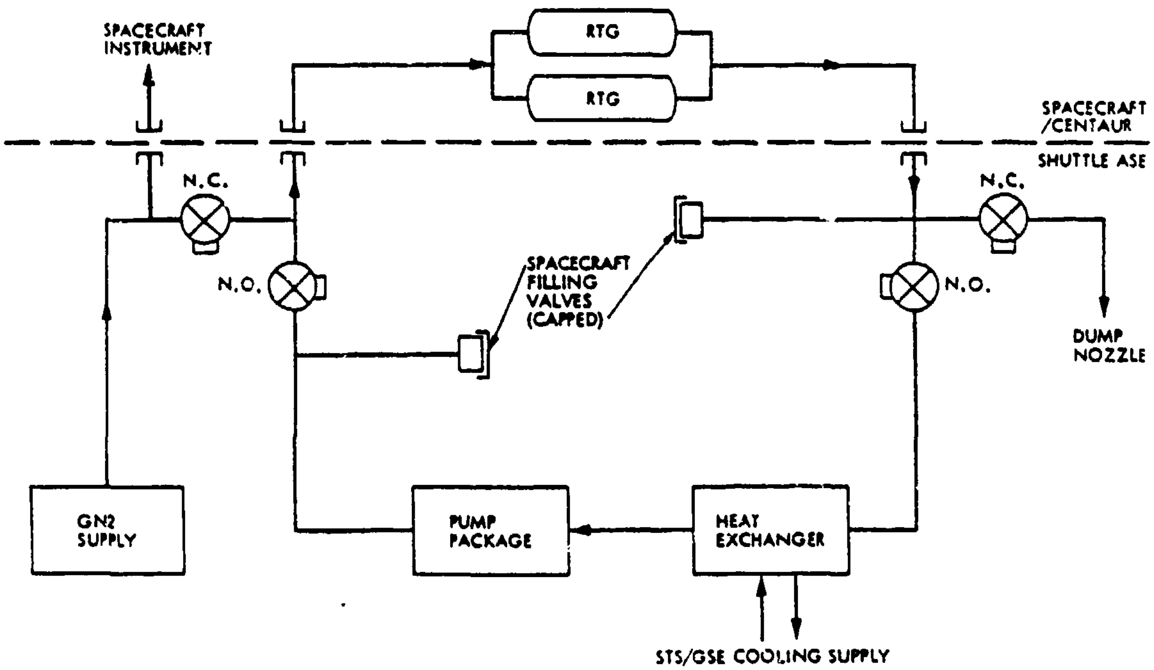

Figure C19-11 Galileo RTG Active Cooling System

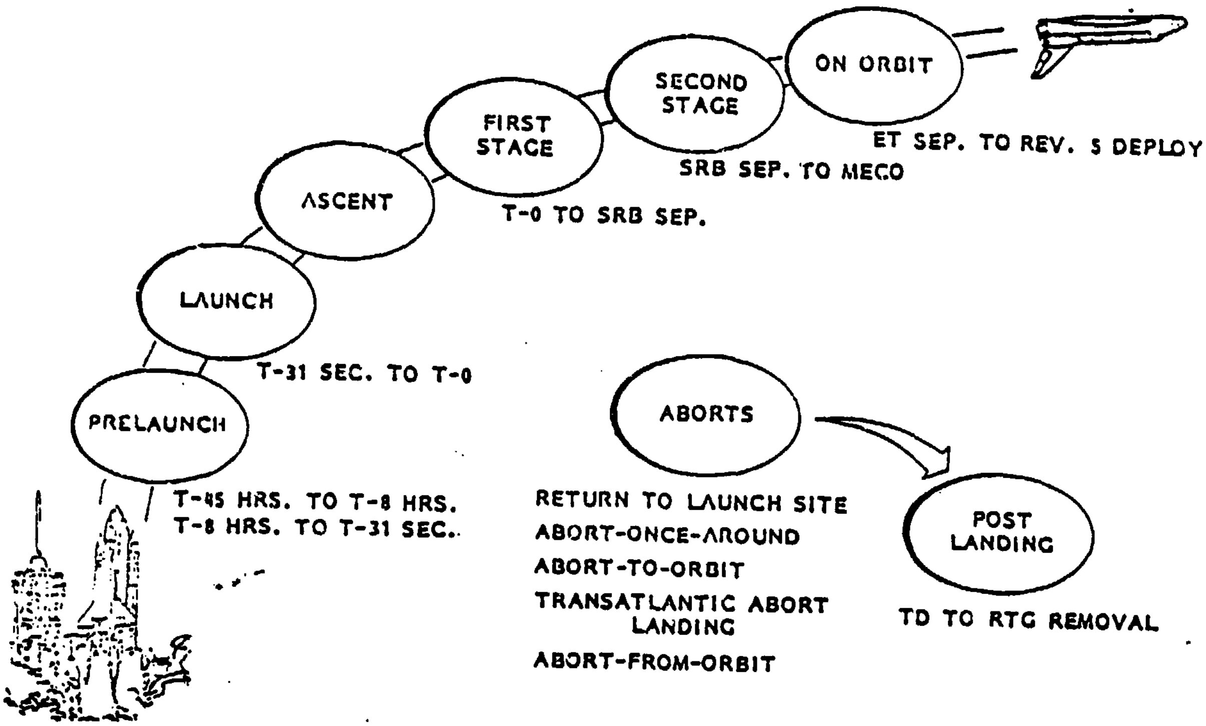

Figure C19-12 Major Phases of Shuttle/Centaur Missions

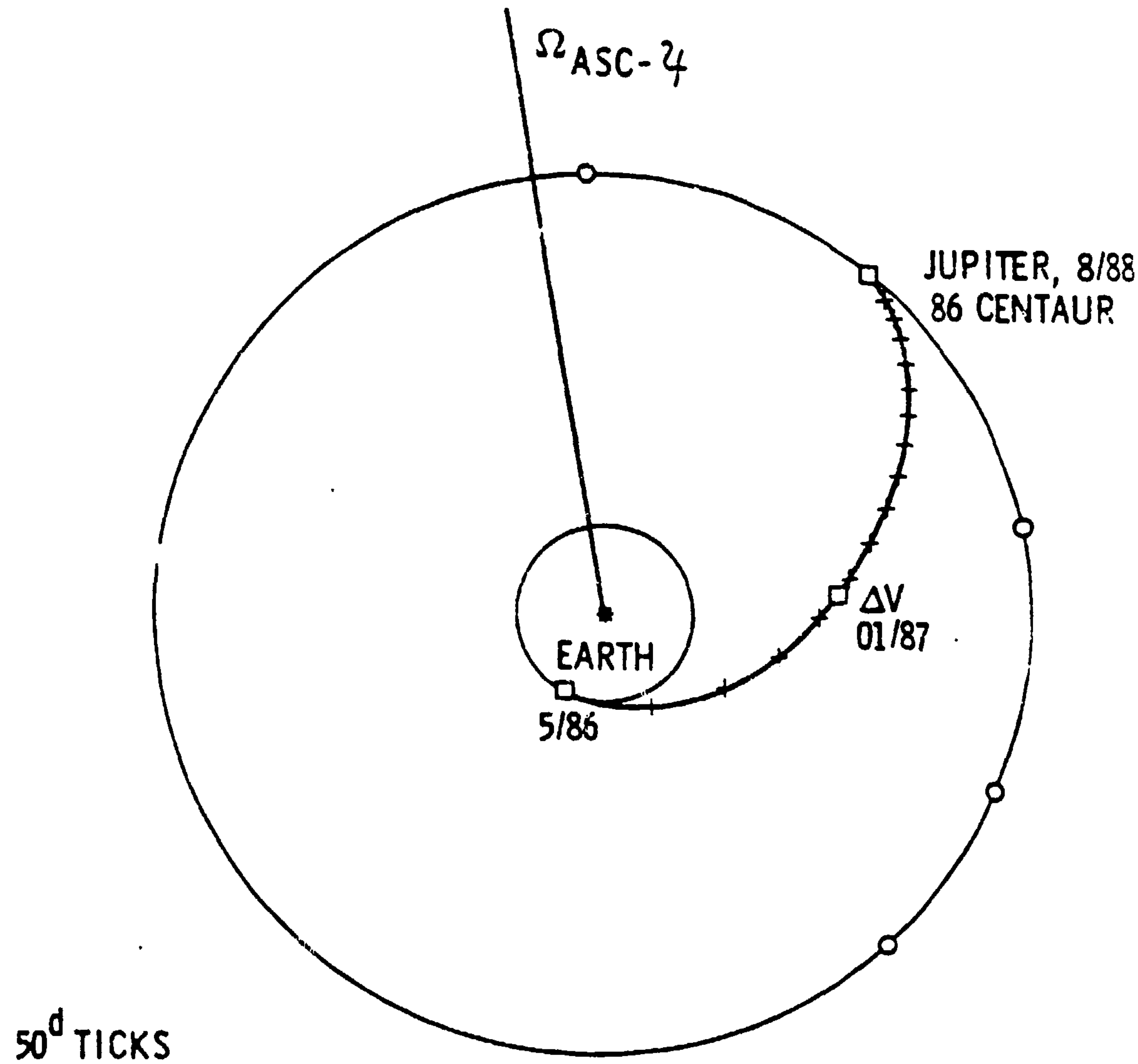

Figure C19-13 Interplanetary Trajectory for 1986 Galileo Mission

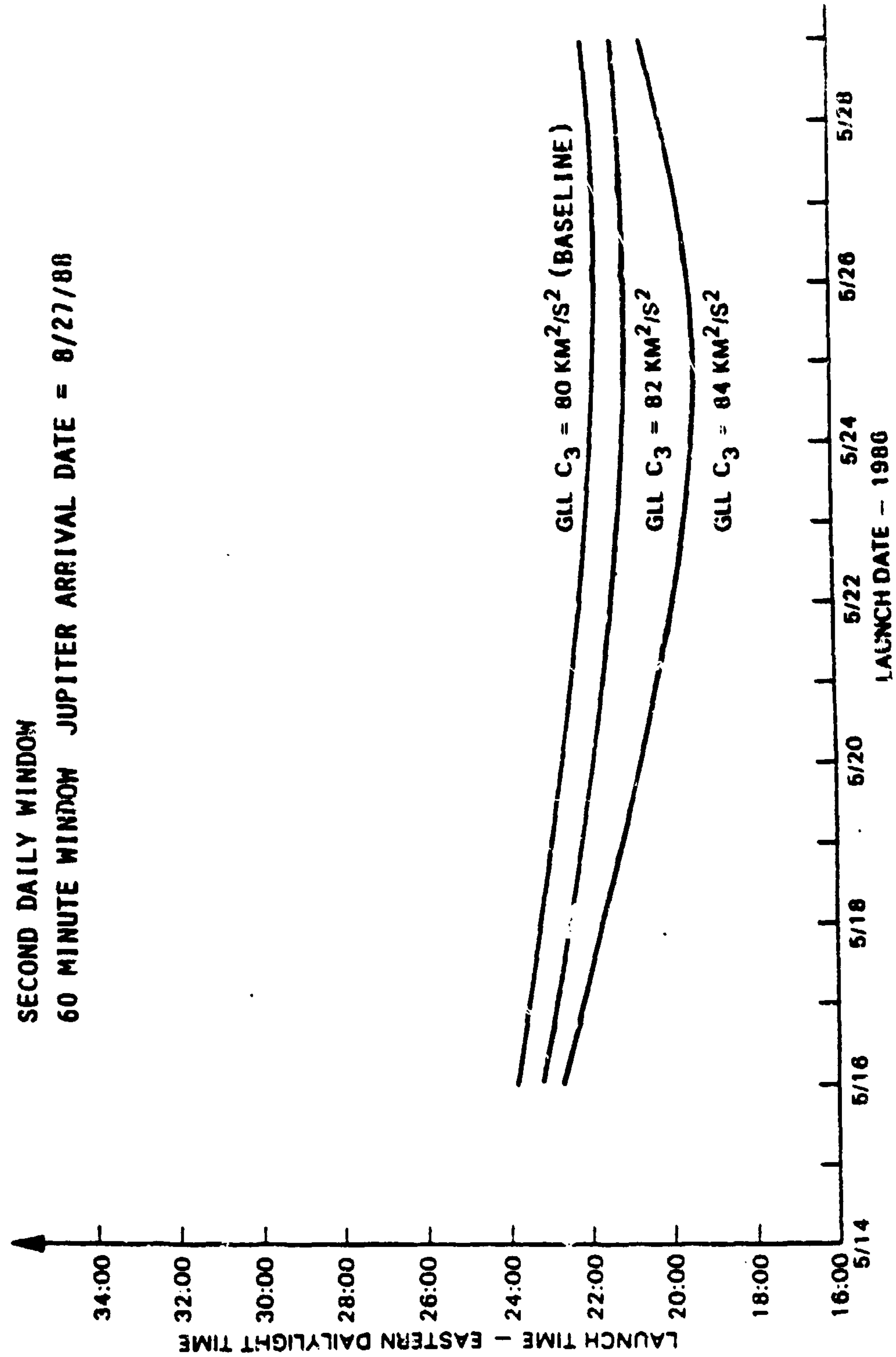

Figure C19-14 Opening of Launch Window vs Launch Date

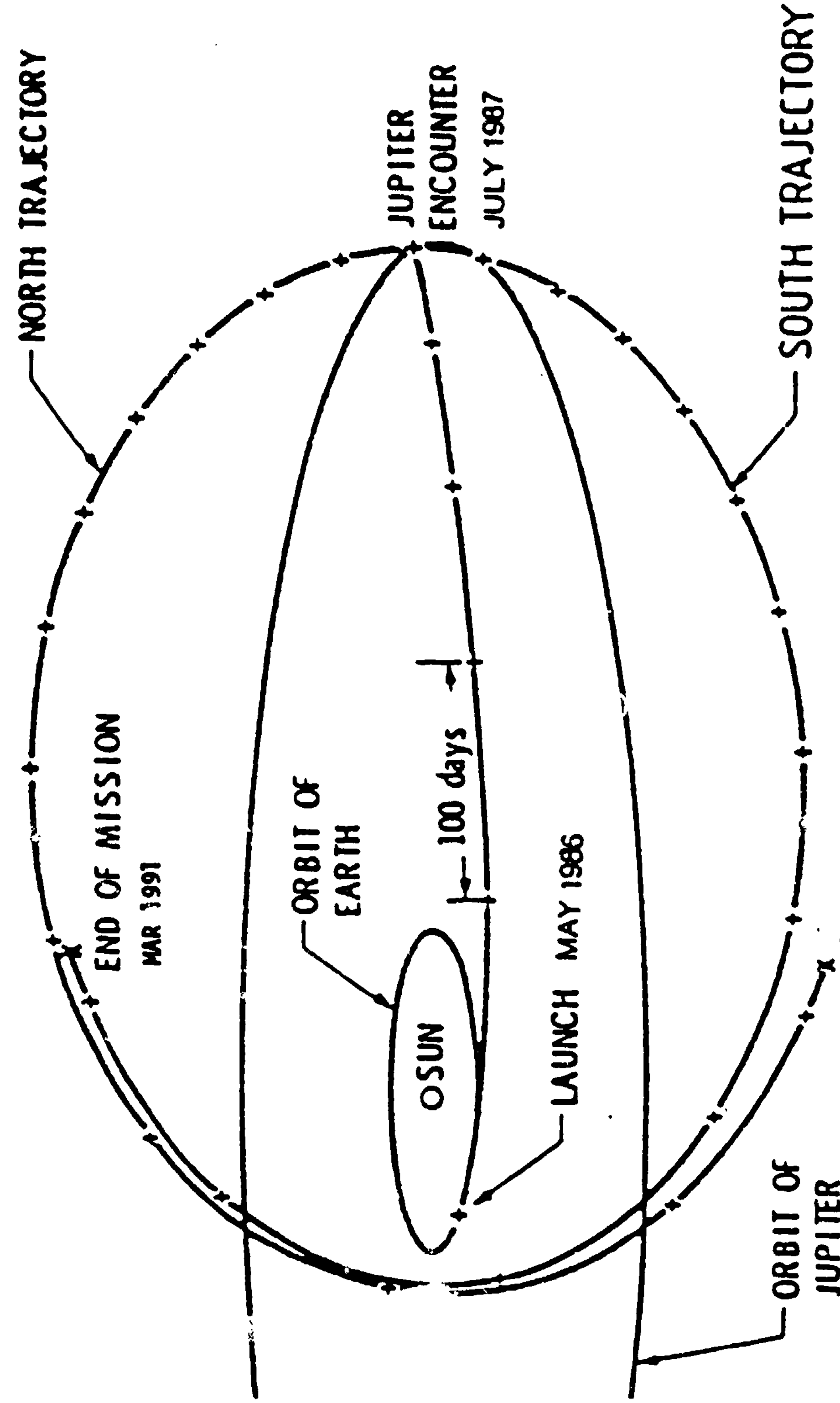

Figure C19-15 Nominal Ulysses Trajectory